<u>İSTANBUL TECHNICAL UNIVERSITY ★ INSTITUTE OF SCIENCE AND TECHNOLOGY</u>

UNMANNED HELICOPTER PRE-DESIGN AND ANALYSIS

M.Sc. Thesis by
Aykut CEYHAN

Department : **Aeronautical&Astronautical Engineering**

Programme : **Interdisciplenary programme**

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Aykut CEYHAN
(511091124)

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Supervisor (Chairman) : Assoc. Prof. Dr. Vedat Z. DOĞAN (ITU
Members of the Examining Committee : Prof. Dr. Zahit MECİTOĞLU(ITU)
Prof. Dr. Tuncer TOPRAK(ITU)

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<u>İSTANBUL TEKNİK ÜNİVERSİTESİ ★ FEN BİLİMLERİ ENSTİTÜSÜ</u>

İNSANSIZ HELİKOPTER ÖN TASARIMI VE ANALİZİ

YÜKSEK LİSANS TEZİ

Aykut CEYHAN
(511091124)

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Tez Danışmanı : Doç. Dr. Vedat Ziya DOĞAN (İTÜ)
Diğer Jüri Üyeleri : Prof. Dr. Zahit MECİTOĞLU (İTÜ)
Prof. Dr. Tuncer TOPRAK (İTÜ)

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FOREWORD

I would like to express my deep appreciation and thanks for my supervisor Assoc. Prof. Dr. Vedat Ziya Doğan and my family for their support.

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Aykut Ceyhan

Aeronatical&Astronautical Engineering

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ABBREVIATIONS

BEMT **:** Blade element momentum theory
BET : Blade element theory
BL **:** Blade loading
DL **:** Disk loading
EO **:** Electro-optical
GW **:** Gross weight
IGE **:** In ground effect
IR **:** Infrared
NOTAR **:** No tail rotor
OGE **:** Out of ground effect
PL **:** Payload
RPM **:** Revolution per minute
UL **:** Useful load

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UNMANNED HELICOPTER PRE-DESIGN AND ANALYSIS

SUMMARY

Unmanned helicopter conceptual design can be done by different methods. These methods change with manufacturer by manufacturer. At this thesis, an estimated value of gross weight is calculated using ideal helicopter formula which states that useful load per gross weight should be between 0.5 and 0.55. This estimated gross weight value is used to obtain a disk loading value which depends on statistical data taken from similar helicopters. With respect to this disk loading value, blade loading and some aerodynamic data are obtained. Aspect ratio of main rotor and tail rotor is wanted to be kept in specified intervals. Weight iteration formulas has been gotton from references which has been stated at relevant chapters. Main rotor power configurations is obtained and by using these data, tail rotor is designed with certain assumptions. By using these processes, a design is constructed and an engine is selected. All calculations are remade with respect to selected engine. Power values at different configurations were determined and performance analysis of designed helicopter has been done. At this thesis, conceptual and preliminary design of a helicopter is explained in detail. At last chapter under the light of determined dimensions, an example helicopter is constructed by a commercial computer aided drawing programme. While preparing drawings, a visual design process has been done. Fuselage is designed by considering aerodynamic efficiency. For future works, with selected materials considering the weight approximations used in the thesis, blade dynamic analysis, computational fluid dynamics analysis, hub strenght analysis can be done.

İNSANSIZ HELİKOPTER ÖN TASARIMI VE ANALİZİ

ÖZET

İnsansız helikopter tasarım süreci farklı metotlarla işleyebilir. Bu metotlar üreticiden üreticiye değişir. Bu tezde maksimum kalkış ağırlığı için ilk yaklaşık değer, faydalı yük maksimum kalkış ağırlığı oranının 0.5 ile 0.55 arasında olması gerektiğini belirten ideal helikopter yaklaşımı kullanılarak bulunur. Bu bulunan maksimum kalkış ağırlığı benzer helikopterlerden çıkarılan istastiki veriler yardımı ile disk yüklemesini bulmakta kullanılır. Bu disk yüklemesi değerine göre pala yüklemesi ve bazı aerodinamik veriler elde edilir. Ana ve kuyruk rotoru için en boy oranı tanımlanan aralıklar içinde tutulmak istenerek bir tasarım yapılır. Ağırlık yaklaşımları kullanılarak yeni bir maksimum kalkış ağırlığı elde edilir. Ana rotorun güç konfigurasyonları belirlenerek kuyruk rotoru belirli kabuller ile tasarlanır. Tasarlanan yeni kongürasyon için yeni bir motor seçilir ve bu seçilen motor için bütün işlemler yenilenir. Farklı durumlar için güç değerleri belirlenir ve tasarımın performans analizi yapılır. Bu tezde bir helikopterin ön tasarımı işlenmiştir. Son bölümde belirlenen boyutlar için örnek bir helikopter ticari bir çizim programı aracılığı ile çizilir ve görsel tasarım yapılır. Gövde tasarımı aerodinamik verim göz önüne alınarak görsel olarak tamamlanır. Malzeme atanması, yapısal ve akış analizleri detaylı tasarımın bünyesinde olduğundan incelenmemiştir. İleriki çalışmalar için ağırlık yaklaşımları gözönüne alınarak seçilecek malzemeler eşliğinde pala dinamik analizi, hesaplamalı akışkanlar dinamiği, hub dayanım analizleri gibi çalışmalar yapılabilir.

1. INTRODUCTION

Helicopter design process is very complicated and can be defined in three steps; conceptual, preliminary and detailed design. At conceptual design, mission profile is obtained and the elements of aircraft are determined by trade-off studies. At this work during conceptual design process, trade-off studies have been done and some specifications as number of blades, engine, interval of dimensions, mission requirements,etc. have been stated at Chapter 2.

Preliminary design determines the helicopter preliminary dimensions, performance, ceilings,etc. This process is just the start point of the helicopter design process and gives an overview of aimed helicopter. This work only contains ceonceptual and preliminary design process. The drawings of designed helicopter can be seen at Chapter 8.

1.1. Purpose

The purpose of this work is to demonstrate the conceptual and preliminary design process of a single rotor conventional unmanned helicopter. The assumptions during the design process can not be used in detailed design and the calculations which have been done here, just gives an overview of the aimed design. After material selection and several analyses the iterative design process should be redone and while doing this iteration new approaches should be used such as new weight iterations, disk loading data, variable profile drag coefficients,etc. The defined mission is achieved and the aimed mission profile is obtained. During the design process, calculations have been made for two altitudes, namely sea level and 2000 meters, in order to maintain the operational altitude defined as 2000 meters. The ceilings which has been obtained at the end of Chapter 7, are not true values; because during calculations engine's performance with respect to altitude change is neglected. The expected values for hover ceiling and service ceilings 2500 meters and 3500 meters respectively. Already, the engine's maximum operational altitude is 4000 meters. Assuming constant profile drag coefficients gives misvalued profile power values for detailed design but these calculations are sufficient enough for preliminary design process. A more detailed Reynolds number and profile drag coefficient approach will be needed for detailed design.

The purpose of this work is to obtain a preliminary design of an unmanned helicopter for the defined payloads. The empty performance of the helicopter has not been considered.

1.2. Similar design

A single rotor unmanned helicopter database was constructed to be able to use statistical data to start the design. Disk loading data was obtained from there. The considered helicopters can be stated as; Atech-pro Foucade, Survey-copter, DSS Scorpio 6, Flycam Flycam, Nrist Z-2, Black eagle50/STD-5 Helivision, Dragonfly DP 3, Techno Sud Vigilant, Robochopper, Scandicraft APID, Yamaha R-50, Yamaha R-max, NRL Dragon Warrior, Camcopter, Soar Bird, Fuji Rph-2, Cac systemes/ED Heliot.

2. CONCEPTUAL DESIGN

2.1. Introduction

At this chapter,mission requirements and a payload which is capable of handling the requirements for that mission and main configurations of helicopter like main rotor, tail rotor, hub and landing gear configurations will be selected by trade-off studies. An engine is selected primitively. This choice can easily be changed by user at the later stages.

2.2. Mission requirements

The helicopter will carry a FLIR camera which has thermal/normal imaging lens and laser pointer on itself. Various types of payloads are considered. These payloads will be explained under the payload title.

Primary mission of this unmanned helicopter is search and rescue. During search and rescue missions there is a need to cover as much of the area as rapidly as possible. Use of unmanned helicopters in such missions together with rescue teams may save time and lives. Helicopters can scan difficult terrain with various sensors and day/night cameras. The system is suitable for mobile search and rescue units because of its compactness. Vertical take-off and landing abilities makes helicopters useful in such missions. The ability to hover and move in all directions gives the user more time and ease to scan objects and track the required object. Also laser pointer is a good way of tracking objects.

Secondary mission of this rotorcraft is law enforcement. Often the best method of law enforcement is done by viewing from above. Many police forces use manned helicopter units both in emergency and routine situations.

Nearly all airborne tasks are done by manned helicopters equipped with visual cameras and IR sensors. But the use of manned helicopters has disadvantages like high maintenance costs, pilot necessity and danger of lives of crew during violent events. Unmanned vehicles are efficient and very useful for every law enforcement tasks. Rotary unmanned air vehicles have great advantages and can be used for stealth operations or regular patrols. It can send live video from the situation area to the user and it can't be detected easily compared with like more bigger and noisy manned ones. And also with a laser pointer, the helicopter can be used for targetting systems for aircraft and missiles.

Another mission of this unmanned conventional helicopter is power line inspection. Power line inspection involves examining the pylons and their high voltage insulators [6]. This process is usually being performed by helicopters. Typically the smallest team is made up of an observer using necessary equipment and a pilot flying at about 5-25 km/h [6]. The manned helicopter usually hovers at a horizontal distance close enough for observation, approximately 25-200 meters and at a height of about 5-25 meters from the ground. This means that the noise produced may be a problem due to noise abatement laws and disturbance to livestock. With the current economic situation where cost reduction is much more important than ever, replacing this method of inspection should be considered. But the dimensions of this unmanned helicopter will be too big for this type of mission in cities and noise will become a problem. But its level of noise is still too low when it is compared with manned helicopters. So; it can be still used for this mission.

All of these missions can be achieved by only one FLIR camera. It will be selected later on this chapter. Estimated mission time for this helicopter is one hour.

2.3. Payload

Cobalt 190 turret provides seven payload in one. It carries four cameras, laser rangefinder, laser designator and laser pointer [7]. Cobalt 190 can be used in reconnasissance, surveillance, target acquisition and target identification. It is a compact sensor suite which can be defined a 19 cm diameter gyro-stabilized turret that carries seven payloads simultaneously. It has a mid-range IR sensor with continuous zoom optics, a long wavelength infrared sensor and two EO sensors [7]. All these daylight/night sensors provide wide field of view. It also has eyesafe laser range finder which supports high accuracy to range to target, target geolocation and Geo-LockTM capability [7].

IR IMAGING PERFORMANCE	
IR Sensor #1	Midwave IR Sensor with 640 x 480 InSb FPA
Fields of View	1.7° to 28° with 14.5X Continuous Zoom
IR Sensor #2	Longwave IR Sensor with 320 x 240 uncooled microbolometer FPA
Field of View	34° Fixed Wide FOV
DAYLIGHT IMAGING PERFORMANCE	
EO Sensor #1	Continuous zoom Color CCD-TV
Fields of View	1.7° to 28° with 14.5X Continuous Zoom
EO Sensor #2	Miniature color CCD TV
Field fo View	35° Fixed Wide FOV
LASER PAYLOADS	
Rangefinder	1.54 µm, eyesafe, supports >20 km range capability
Designator (optional)	1.06 µm diode pumped
Laser Pointer	850 nm, 30 mW output power
AUTOMATIC VIDEO TRACKING	
	Two independent trackers
TARGET GEOLOCATION/GEOPOINTING/GEO-CUEING	
	Using External GPS/IMU
SYSTEM PERFORMANCE	
Gimbal type	2-axis gyro-stabilization
Azimuth coverage	360° continuous
Elevation coverage	+20° to -60°
SYSTEM INTERFACES	
Video	Dual RS-170; split screen capable
Control	Open architecture to simplify ground station integration
Data	IP over Ethernet
POWER REQUIREMENTS	
18-32V DC	Designed to MIL-STD-704E
DIMENSIONS, WEIGHT & MOUNTING	
Size	7.5 in x 10.6 in (190 mm x 270 mm)
Weight	~18 lbs (8.2 kg)
ENVIRONMENTAL	
Standards	MIL-STD-810 & MIL-STD-461

Figure 2.1 : Specifications of turret camera [7]

Cobalt 190 includes a laser designator supports precision targeting. The last sensor it has is a laser pointer which pinpoints targets for observers using night vision equipment, while remaining invisible to others. It simultaneously detect targets with both IR and EO sensors. Targets acquired with one sensor can be handed off to the other. Its design provides two simultaneous video output that viewing two sensors simultaneously in separate display or split screen is possible. The camera will be located at the most suitable forward position in the airframe as the center of weight remains in the possible center of weight envelope. The drawings and location can be seen from Catia drawings.

Figure 2.2 : The selected turret camera [7]

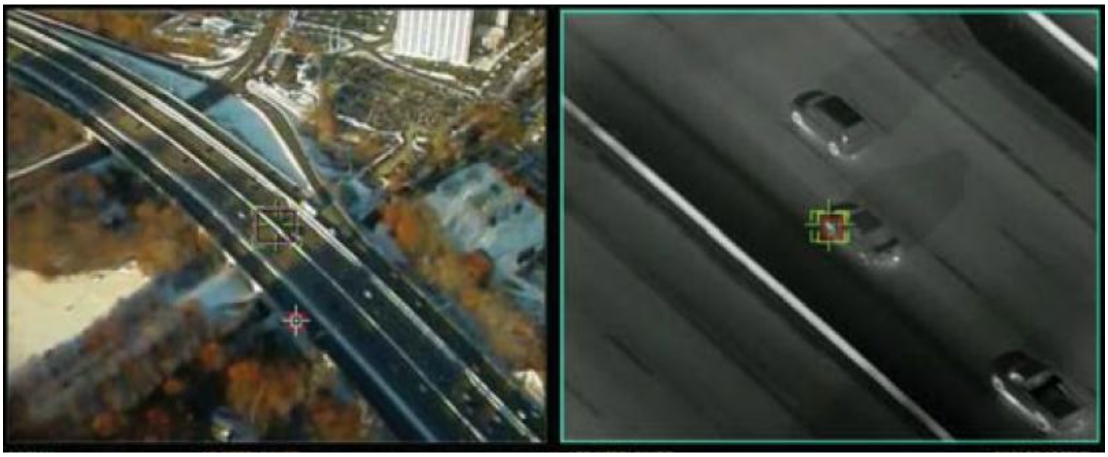

Figure 2.3 : Images from Cobalt 190 [7]

A mission computer is loaded into vehicle in order to be able to use it in different mission types with the same payload. Parvus DuraCOR 820 computer is selected. This computer is proper for MIL-STD-704E standarts [8].

The DuraCOR 820 is a mission processor system, optimally designed for military/aerospace ground mobile and airborne deployments. Targeting unmanned applications where reliable high performance computing is required, the DuraCOR 820 can be suitable for harsh environmental conditions such as high altitude, extreme temperatures and high vibration levels. It also has a autopilot circuit for unmanned systems.

Figure 2.4 : Mission computer [8]

Low-Power x86 CPU	▪ Intel Pentium M738, 2048k L2 cache ▪ 1.4 GHz Clockspeed w/Speedstepping (Equivalent to a 2.8GHz Pentium 4 Performance)
Board Architecture	▪ Compliant with PC/104-Plus Standard (PCI / ISA Bus)
RAM Memory	1024MB DDR-SODIMM
Solid State Disk	16 GB / 8 GB / 2 GB Non-Volatile CompactFlash
Operating System	▪ Pre-installed Linux, Windows 7, or Windows Embedded (120 day eval license) ▪ Hardware Compatible with x86 embedded and Real-Time Operating Systems
Network	▪ 2x Fast Ethernet Network Interfaces (10/100Mbps)
Serial	▪ 2x RS-232 serial ports, 115Kbps max
USB	▪ 3x USB 2.0 Ports
Video	▪ VGA Analog Video Output
PS2	▪ PS2 Keyboard and Mouse Ports
DIO	▪ Eight General-Purpose Digital I/O (GPDIO) lines - 4 Parallel Port Control Lines, 4 Parallel Port Data Lines; Capable of Sinking/Sourcing 12mA
Physical	▪ Weight: Approx. 2.9lbs (1.32kg) ▪ Dimensions (H x W x L): 3.03" (7.70cm) x 4.30"(10.92cm) x 7.05" (17.91cm) - Including Connectors and Baseplate ▪ Chassis: Aluminium Alloy, Resistant to Corrosion, Dust, Water ▪ Connectors: Glenair Series 801 Mighty Mouse (MIL-38999-like) ▪ Cooling: No Moving Parts. Passive Conductive Cooling ▪ Installation: Flange Mount Baseplate ▪ Finish: Anodized per MIL-A-8625, Type II, Class 2

Figure 2.5 : Specifications of mission computer [8]

Power	▪ 9-32 VDC Input (28VDC Nominal) ▪ Reverse, Over Voltage, Surge Protected ▪ MIL-STD-704F Compliance ▪ <24 Watts Power Dissipation (max) ▪ Ground: Grounding Lug for Connection to System Chassis Ground ▪ Battery for Real-Time Clock Maintains Time/Day for 30 Days+
Environmental	Tested and Qualified per MIL-STD-810G: ▪ Operating Altitude: Up to 60,000 feet (18,288 meters) w/ derating ▪ Operating Temp: -40°C to +71°C ambient (-40°F to +160°F) ▪ Storage Temp: -40°C to +85°C (-40°F to +185°F) ▪ Humidity: per MIL-STD-810G, Method 507.5, Procedure II; Conformal Coated ▪ Water Ingress: 1 Meter Submersion, 30 Minutes (Similar to IP67) ▪ Operating Shock: 15g, 15ms, ½ Sine Wave, 3 Pos/Neg per Axis, Total 18 Pulses ▪ Random Vibration: Combined Jet-Helo Profile per MIL-STD-810G, Method 514
EMI/EMC	Tested and Qualified to MIL-STD-461E: ▪ CS101, Power Leads, 30 Hz to 150 KHz, Curve 2 (28V and below) ▪ RE102, Electric Field, 10 KHz to 18 GHz, Figure RE102-3 for Fixed Wing Shorter than 25m ▪ RS103, Electric Field, 30 MHz to 18 GHz
Warranty	1 Year RTF Warranty (Extended Service Contracts Available)
Reliability (MTBF)	Calculated per MIL-HDBK-217F @ 25°C / @ 71°C ▪ 476,542 Hours / 119,574 Hours (Ground Benign, Controlled GB, GC) ▪ 74,932 Hours / 38,161 Hours (Airborne Inhabit Fighter) ▪ 95,299 Hours / 43,558 Hours (Ground Mobile)
Special Order Options	Optional Integrated PC104(+) I/O / Comm Module (rather than 2nd Ethernet NIC): ▪ *Dual-Redundant MIL-STD-1553 Interfaces* (1-4 Channels, DDC BC/RT/MT Architecture, IRIG-B Time Code Inputs) ▪ *RS-232/422/485 Asynchronous Serial Ports* (115 Kbaud max speed; 1-6 channels, depending on protocol and flow control support) ▪ *12-Channel GPS* (Fastrax iTrax03 Receiver: NMEA and Binary GPS Protocols; L1 frequency & C/A code (SPS)) ▪ Other Custom Configurations Possible. Consult Sales@parvus.com for more information.

Figure 2.6 : Other specifications of mission computer [8]

2.4. Main rotor configuration

Under this title, main rotor configurations are explained and one of them will be selected by trade-off.

2.4.1. Single rotor

The single rotor configuration or conventional helicopters needs tail rotors to provide anti torque. The conventional helicopters are the most preferred helicopters and the lift is maintained by only one rotor. Their good hover performance and their more stabilized structures compared with other types make them a good choice during the trade-off process.

Figure 2.7 : Conventional helicopter [2]

2.4.2. Coaxial rotor

This system has two counter rotating rotors to balance anti torque without tail rotor. So, all power is used for lift only. The efficient usage of motor power, flight safety and low structural weight are the advantages of co-axial rotors compared with conventional ones. Besides, worse autorotation ability, control difficulties and aerodynamic inefficiencies for high disk loadings are the disadvantages of co-axial helicopters [5].

Figure 2.8 : Coaxial helicopter [2]

2.4.3. Tandem rotor

They are the most preferred rotor configurations after single rotor configuration. The rotors are rotating counter like co-axial rotors but here the second rotor is positioned on the tail. They do not need anti torque devices as in the co-axial ones. They preferred for heavy transportation helicopters. Tandem helicopters can carry external loads because their center of gravity envelope is more wide than the others. They have smaller blades than conventional rotors because they have two rotor system. So, the rotors are rotating at higher RPM values than the conventional ones. This gives smaller reduction ratio between engine. And smaller ratios means lighter transmissions. Tandem helicopters have disadvantages like their control problem, slow stabilities and higher weights [5].

Figure 2.9 : Tandem helicopter [2]

2.4.4. Selection of main rotor configuration

A trade-off table is shown below. The meaning of points are also given in another table.

Table 2.1: Trade study for rotors

	%	Single	Co-axial	Tandem
Velocity	8	4	3	4
Noise	10	4	3	2
Range	12	3	4	5
Hover efficiency	10	4	5	4
Autorotation	8	4	3	2
Dimension	10	4	5	3
Weight	10	4	5	2
Manoeuvrability	8	2	2	2
Productibility	8	5	4	4
Maintenance	8	4	2	4
Reliability	8	5	4	4
		3.88	3.72	3.3

1	Very bad
2	Bad
3	Normal
4	Good
5	Very good

As a result of trade-off study single rotor conguration is selected.

2.5. Tail rotor configuration

Main rotor is selected and there is an anti torque device needed. Types of tail rotors are described below and then one of them is selected by trade-off.

2.5.1. Conventional tail rotor

They are the most used anti torque systems which are very simple and cheap. The purpose of a conventional tail rotor is to generate a thrust force in the opposite direction of the thrust which is generated in main rotor by means of rotation effect. Most conventional tail rotors have two to five blades whose tips are exposed to the external air.

Figure 2.10 : Conventional tail rotor [2]

2.5.2. Notar

NOTAR (NO TAil Rotor), is a different anti torque system makes use of compressed air that is forced out of slots inside the tail boom [9]. This jet of air changes the direction of the air flow to create an aerodynamic force that balance the aircraft.

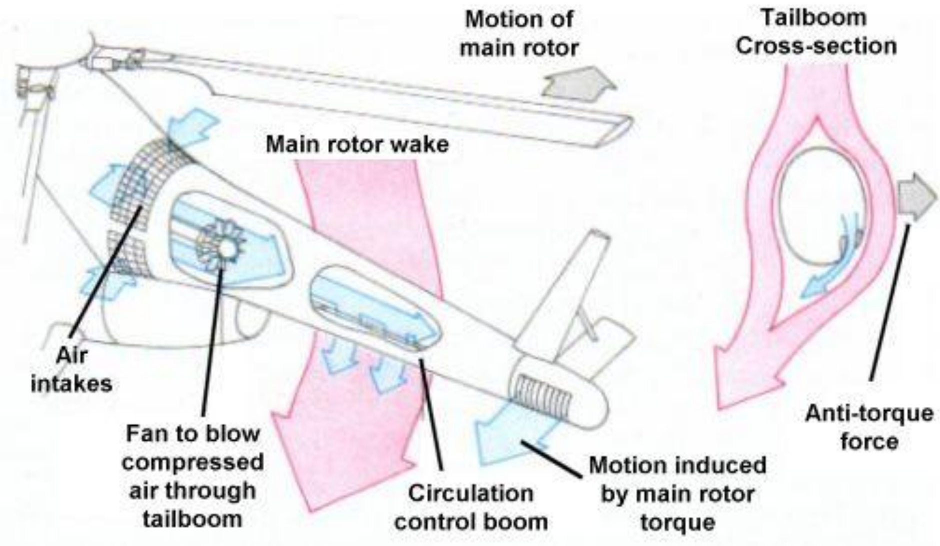

Figure 2.11 : Notar [9]

2.5.3. Fan-in-fin

The advantages of fenestron system with respect to conventional tail rotor can be summarized as increased safety, being less vulnerable to foreign object damage and reduced noise. Disadvantages can be stated as higher weight and higher air resistance due to the enclosure thickness, higher consruction cost and higher power requirement for a given thrust value. The fenestron can produce the same total thrust for the same power as a tail rotor nearly 30% larger and fan diameter can be small as approximately 30% of the tail rotor it is replacing [5]. The be effective, the depth of the duct should be at least 20% of the fan diameter [5].

At this thesis, fenestron was introduced and it is confirmed that it is not necessary. Fenestron should be selected for much more heavier aircrafts.

Figure 2.12 : Fenestron [2]

2.5.4. Selection of tail rotor configuration

Most of the unmanned helicopters use conventional tail rotors because of easiness of manufacturing. Notar and Fenestron designs is too complicated for a small size unmanned helicopter. Fenestron blades will be too small and this gives the manufacturer a problem. Notar selection is never been efficient like conventional ones. So; Notar takes the minimum point in the trade-off study.

Because of these, the conventional tail rotor configuration is selected. Due to the design process, this selection could be aborted and fenestron design will be able to selected for example more smaller blade radii will be able to be necessary; so this makes fenestron choice more logical. But for now; conventional tail rotor configuration is selected. The pusher configuration is selected. Therefore,for a main rotor rotating clokwise, tair rotor is on the left side of the tail rotating clockwise. Tractor configuration is not selected because at tractor configuration the inflow is going into the tail and this gives inefficiency of tail rotor performance. Tail rotor can be seen from the Catia drawings.

Table 2.2: Trade study for tail rotor

	%	Conventional	Notar	Fenestron
Maneuverability	10	5	2	3
Noise	15	3	5	4
Efficiency	15	4	3	5
Productibility	15	5	2	3
Reliability	15	4	4	4
Maintenance	15	5	3	3
Safety	15	3	5	5
		4.1	3.5	3.9

1	Very bad
2	Bad
3	Normal
4	Good
5	Very good

2.6. Hub configuration

Different configurations was considered and a hub configuration was selected. In order to select most suitable hub type, four hub types were investigated: teetering/gimbaled, articulated, hingeless and bearingless. In order to understand the basics clearly a figure is given below.

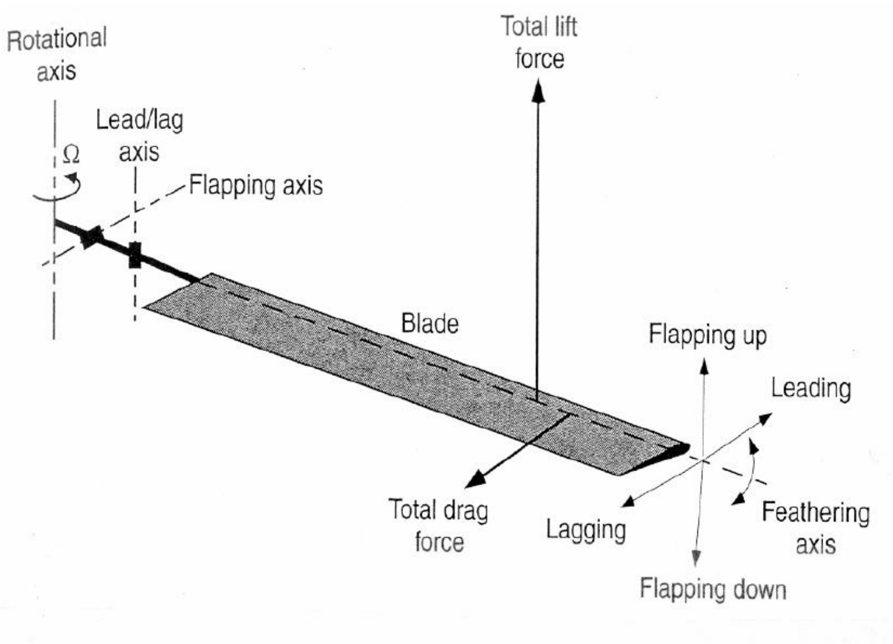

Figure 2.13 : Axes of a blade [4]

2.6.1. Teetering rotor

It does not use independent lead-lag or flapping hinges. When one blade flaps up the other flaps down. Also seperate feathering bearing gives cyclic or collective pitching ability to system. It needs a stabilizer bar which acts like a gyroscope and introduce flapping cyclic pitch feedback. It is mechanically simple but because of stabilizer bar this type of hubs have high parasitic drag in forward flight. On later teetering rotor designs,this problem is solved.

2.6.2. Fully-articulated rotor

It contains flap and lead-lag hinges both and a feathering bearing. Because of low drag and aerodynamic damping in the lead-lag plane,mechanical dampers are located at the lag hinges. It is mechanically complicated structure and its maintenance cost is higher. These hinges and its mechanical complication makes this rotor disadvantageous because of high parasitic drag in forward flight.

2.6.3. Hingeless rotor

It does not have lead/lag or flapping hinges. It contains a feathering bearing. It uses elastic flexing of a structural beam instead of hinges; so it is mechanically much more simple than articulated systems. Its design process is so complicated than the teetering or articulated systems. But it gives a advantage in forward flight because of its lower parasitic drag. The configuration has better maneuvering capability because its stiff hub design makes the helicopter is more sensitive to control inputs.

2.6.4. Bearingless rotor

Different from hingeless design, bearingless rotor also eliminates the feathering bearing. Three degree of freedom is obtained by bending,flexing and twisting of the hub. Its design process is complicated because of the structural beams should be made from new technology born composite materials such as Kevlar.

2.6.5. Selection of hub

Teetering design is not considered because of its necessity of a stabilizer bar. Bearingless rotor hub configuration is selected with the trade-off table shown below.

Table 2.3: Trade study for hub

	%	Articulated	Hingeless	Bearingless	1 Very bad
Maintenance	35	1	3	5	2 Bad
Maneuverability	15	1	3	3	3 Normal
Cost	20	5	4	3	4 Good
Productibility	30	4	3	2	5 Very good
		2.7	3.2	3.4	

2.7. Landing gear configuration

The aircraft will be relatively small in size and weight and it does not have to achieve high forward speeds. So, a rectractable landing gear does not needed. Tricycle-type landing gear is not needed also. Skid type landing gear is selected here.

2.8. Empennage

At the tail rotor a vertical stabilizer will be used. The primary purpose of of vertical stabilizer is to provide stability in yaw. While the tail rotor itself provides considerable yaw stability, the vertical stabilizer may also be required to provide suffcient anti-torque to allow continued flight in the event of the loss of the tail rotor. This side force can be provided by using an airfoil section with a relatively large amount of camber.NACA 63421 airfoil has been used in design. At flybar, NACA 63418 airfoil is used for paddles.

2.9. Engine

To find an estimated gross weight value, at first an engine must be selected. Various types of engines has been considered. This selection can be changed at later stages due to the power needed by helicopter. Momentum theory will be used and the error rate of theory will be considered.

DA-85 engine which is produced by Desert Aircraft Company, is selected at first stage.Various types of engines were considered like Nitto Manufacturing (NRG-20EH and NR-20EH), AMT Engines (Mercury), Wankel rotary engine (49 PI), Kavan engine (4-stroke,50cc model), Wren 44 helicopter engine, Zenoah (G26/G231 Heli), Radne Motor AB (Raket 120), Mecoa (.32 Heli, .46 Heli, .46 Heli-swamp buggy with pull starter), Toki (.40 Heli engine with muffer), HB (.25 Heli, .40 Heli,

.61 PDP Heli), O.S. Engines (37SZ-H Ringed heli engine, 50-SX-H Ringed hyper heli engine, 55 HZ-R DRS Heli engine, 55HZ-H Hyper Ringed Heli engine w/40L, 55HZ Limited Heli engine w/Powerboost pipe, 70SZ-H Ring 3D Heli engine, 91HZ Ringed, 91HZ, 91HZ-R 3D Helicopter engine, 91RZ-H Rear Exhaust Ringed heli engine)

Table 2.4 : Specifications of first engine [13]

Displacement	5.24ci (85.9cc)
Output	8.5 hp(6.3kW)
Weight	4.3 lbs (1.95 kilos)
Bore	2.047 in (52 mm)
Stroke	1.59 in (40.49 mm)
Length	5.9 in (150 mm)
RPM Range	1200 to 7500
RPM Max	9500
Fuel Consumption	2.2 oz/min @ 6,000 RPM
Warranty	3 years

The characteristics of selected engine can be seen above. It has Walbro carburator and 7075 aluminum alloy crankcase. Its fuel consumption 2.2 oz/min (0.062 kg/min).

2.10. Mission overview and pre-work on design

For this selected engine an estimated gross weight can be calculated from the given formula below which is suitable for an ideal helicopter

$$\frac{UL}{GW} = 0.55 \quad \textbf{(2.1)}$$

For this helicopter this ratio will be selected 0.5

$$UL = Payload + Fuel \quad \textbf{(2.2)}$$

Payload mass is 8.2 kg and a mission computer which is 1.32 kg. Payload is approximately 10 kg. And mission time is defined one hour, the required fuel is;

$$0.062 \; x \; 60 = 3.72 \; kg \quad \textbf{(2.3)}$$

So the useful load is approximately 14 kg. By using these data, gross weight is calculated as approximately 28 kg. This is an estimated value and it only shows the order of the size of the helicopter. By using estimations, the motor can be checked if it is suitable for this gross weight with momentum theory by considering the worst scenario. Before doing this, radius and chord value interval must be defined.

Table 2.5: First approach to geometric quantities

Main rotor radius	r	0.6<r<1.5
Main rotor chord	c	0.03<c<0.1
Endurance	t	60 min
Payload	PL	14 kg
Gross weight	GW	28 kg
Power output of motor	Peng	6.3 kW
Transmission losses		app. 20%
Root cut out ratio<1	r_0	0.2

Main rotor radius and chord intervals are taken from similar size helicopters and transmission losses of transmitted power is an approximated value for general helicopters [5].

$$A_e = \pi r^2 (B^2 - r_0{}^2) \tag{2.4}$$

where r_0 is root cut out ratio and must be small from one. Here, it is selected 0.2 as the worst scenario. B is tip loss factor and can be defined by Gessow and Myers as;

$$B = 1 - \frac{c}{2r} \tag{2.5}$$

The c defined above is tip chord. Here it is taken as 0.03 as the worst scenario for lift and radius is selected as r=0.6 m and the density of air is taken 0.84 at 11000 ft. The induced velocity for hover which is the most power consuming scenario is;

$$v_i = \sqrt{\frac{T}{2\rho A_e}} \tag{2.6}$$

And for hover the rotor thrust is;

$$T = GW \tag{2.7}$$

And the power consumed by rotor is

$$P = Tv_i \tag{2.8}$$

This power value will be compared with the power output of engine selected considering the transmisson losses as 20% and the 30% error rate of momentum theory

```
r=input('radius');                  %Main rotor radius
c=input('chord');                   %Main rotor chord
GW=input('Gross Weight');           %Gross weight
r0=input('Root cutout');            %Root cutout ratio
ro=input('Density');                %Density
pi=22/7
T=GW                                %For hover
B=1-(c/(2*r))                       %Tip-loss effect
Ae=pi*(r^2)*((B^2)-(r0^2))          %Effective rotor area
vi=sqrt(T/(2*ro*Ae))                %Induced velocity for hover
P_rotor=(T*vi)/1000                 %Power consumed by rotor(kW)
P_engine=P_rotor*1.20               %Power trasmitted by engine (kW)
P_engine_safety=P_engine*1.30       %Consideration of error rate
```

Figure 2.14 : A basic computer code for power

The final engine output for worst scenario is 5.55 kW. This means the selected engine is suitable for this helicopter by means of power needs.

The calculations above is not valid actually. This is done only because to have the first approximation of selecting the engine and determining an estimated value for gross weight. At similar design processes empty weight value will be an input for starting the design.But here an engine is selected to find useful load and gross weight. But it should not be forgotten that this selection is not suitable actually for design processes. The engine is a rubber engine, so it can be changed at later steps of work.The next chapter will focus on main rotor design and for this engine blades will be designed.

An example of mission profile needs to be constructed in order to understand the concept. This profile is just an example and calculations are not done for this situation.

2.11. Mission requirements

Table 2.6: Mission requirements

Range	**Best range**
Endurance	60 min
Payload weight	8.2 kg FLIR + 1.34 kg mission processor
Payload camera volume	19cm x 19 cm x 27 cm
Payload computer volume	7.7 cm x 10.92cm x 17.91cm
Start up	0.5 min
Climb	5 min
Hover	5 min
Forward flight	10 min + 10 min at maximum speed
Loiter	24 min at best endurance speed
Descent	5 min
Landing/shut down	0.5 min
Cruise speed	80 km/h
Operation altitude	2000 m

The mission requirement can be seen from the table above. An example mission profile is sketched below.

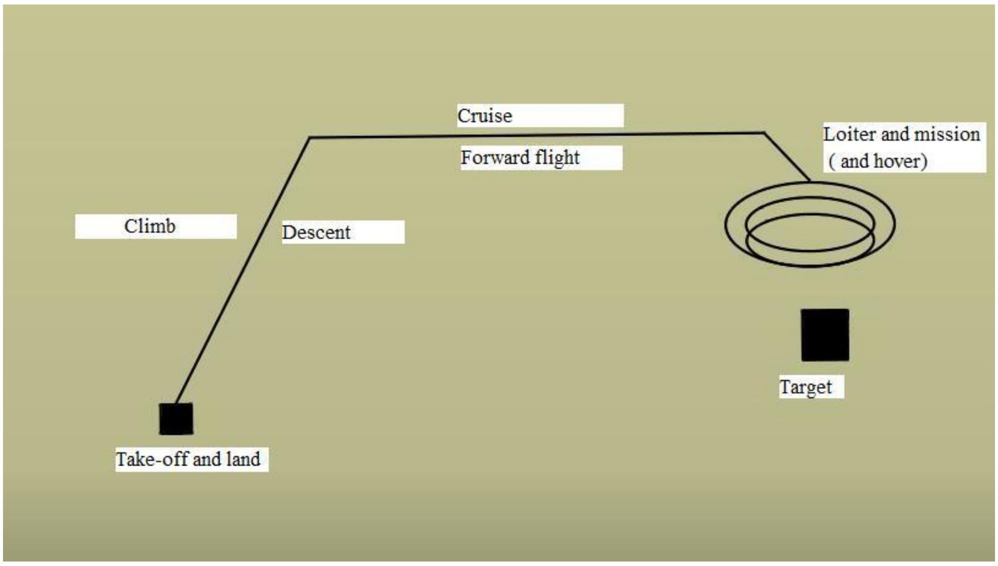

Figure 2.15 : A mission profile

Mission profile is an ordinary mission's profile. The design is expected to counter all these requirements.

3. MAIN ROTOR DESIGN

3.1. First estimation of gross weight

The first step of design is determining an estimated value for helicopter's gross weight which has already be done at Chapter 2 by a rubber engine. Usually the gross weight is an input variable at design processes. It can be calculated from manufacturer's empty weight-gross weight charts or from historical data for typical missions.

GW=28 kg **(3.1)**

The second step is to calculating the maximum tip velocity by an assumption that at the tip, any section has a Mach number below 0.65. [3] The target altitude of the helicopter is 2000 meters. Rotor tip speed will be maximum at sea level so the calculations for tip speed will be calculated for sea level.

The speed of sound can be calculated by

$a = 331.5 + (0.6T)$=340 m/s at sea level **(3.2)**

Where T is temperature in Celsius and a is the speed of sound. At sea level the speed of sound is 340 m/s. The maximum tip velocity $V_{tip,max}$ is;

$V_{tip,max} = M_{tip,max} . a$ = 221m/s **(3.3)**

Here a=340 m/s and $M_{tip,max}$=0.65. So; maximum tip velocity is calculated as 221m/s

Rotor radius will be calculated from the equation below by using disk loading-gross weight charts for similar helicopters.

$$R = \sqrt{\frac{GW}{\pi DL}} \tag{3.4}$$

Data mentioned above are listed at the table below.

Table 3.1: Disk loading data for similar helicopters**[2]**

Helicopter	**Radius** (m)	**GW** (kg)	**DL** (N/m^2)
Atech-pro Foucade	0.900	7.2	27.735
Copter 1	0.910	10.4	39.184
DSS Scorpio 6	0.900	13	50.078
Flycam Flycam(Medium)	0.900	15.2	58.553
Copter 2	1.100	18	46.408
Flycam Flycam(Industrial)	1.100	20.7	53.370
Flycam Flycam(Giant)	1.500	31	42.980
Nrist Z-2	1.625	35	41.350
Black eagle50/STD-5 Helivision	1.009	35	107.254
DSS Scorpio 30	1.100	38	97.974
Dragonfly DP 3	1.120	38.1	94.718
Techno Sud Vigilant	0.975	39	127.954
Robochopper	1.120	50	124.302
Scandicraft APID	1.490	55	77.220
Dragonfly DP4	1.295	63.5	118.08
Camcopter	1.545	66	86.250
Yamaha R-50	1.535	67	88.700
Yamaha R-max	1.560	88	113.170
Yanmar YH-300	1.690	88	96.078
NRL Dragon Warrior	1.220	113	237.280
Soar Bird	2.920	280	102.440
Fuji Rph-2	2.400	325	176.015
Cac systemes/ED Heliot	3.350	450	125.240
SAIC Vigilante 600	3.505	499	126.650
Orka-1200	3.600	680	163.600
Robocopter 300/Argus	4.090	794	140.070
RQ-8 Fire Scout	4.190	1157	205.197

Disk loading and gross weight values are of some of the helicopters listed above is shown below. Here the variance is 0.9266. Estimated disk loading value can be calculated by the equation below

$$DL \cong 16{,}932GW^{0.3694} \tag{3.5}$$

For GW= 28 kg disk loading is 57.98. At this work T is defined as thrust means T=GW(9.81) until another definiton will be made.

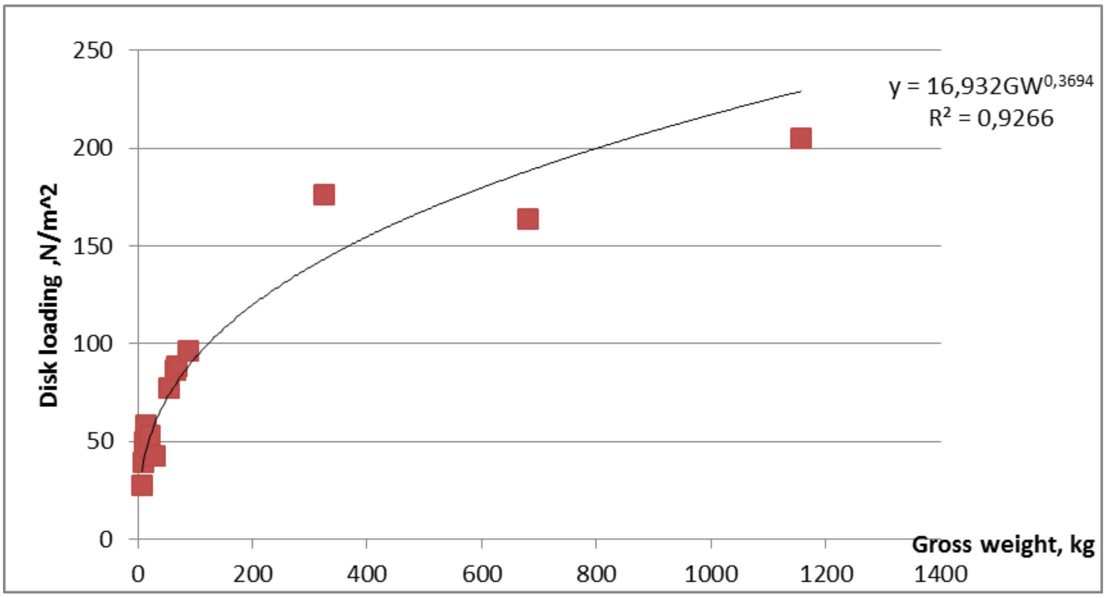

Figure 3.1 : Gross weight disk loading variation

The optimum radius of the main rotor is

$$R = \sqrt{\frac{T}{\pi.DL}} = \sqrt{\frac{28(9.81)}{\pi(57.98)}} = 1.228m \tag{3.6}$$

Maximum rotational velocity can be calculated since the rotor radius and maximum tip velocity are known from the equation given below:

$$\Omega_{max} = \frac{V_{tip,max}}{R} = \frac{(221)}{(1.228)} = 146.5526\, rad/s \cong 1399 \text{ rpm} \tag{3.7}$$

The density value at 2000 meters altitude will be calculated as the helicopter will operate at that altitude.

$$\rho = \rho_0 e^{\frac{-0.0296h}{304.8}} \tag{3.8}$$

where h is altitude in meters and ρ_0 is the density at sea level which equals to $1.225\ kg/m^3$[4] . So; the density value at 2000 meters is 1.0087 kg/m^3.

At first all calculations will be done for sea level condition,later they will be done operating altitude,2000 meters,also.

The first estimation of thrust coefficient can be found by using the equation below:

$$C_T = \frac{T}{\rho A V_{tip}{}^2} = \frac{(28)(9.81)}{(1.225)(\pi(1.228)^2)(221)^2} = 9.69 \text{ x } 10^{-4} \tag{3.9}$$

Maximum forward speed can be seen from mission requirements. The required maximum forward speed is 130 km/h. So the maximum advance ratio can be found

$$\mu_{max} = \frac{V_{f,max}}{V_{tip,max}} = \frac{(130)(1000)}{(221)(60)(60)} = 0.163399 \tag{3.10}$$

Blade loading-advance ratio variation can be seen from the table given below[3].

Table 3.2 : Advance ratio blade loading coefficient variation

Advance Ratio	Blade Loading
0.10	0.126
0.15	0.122
0.20	0.117
0.25	0.112
0.30	0.106
0.35	0.100
0.40	0.091
0.45	0.085
0.50	0.075
0.55	0.065
0.60	0.054

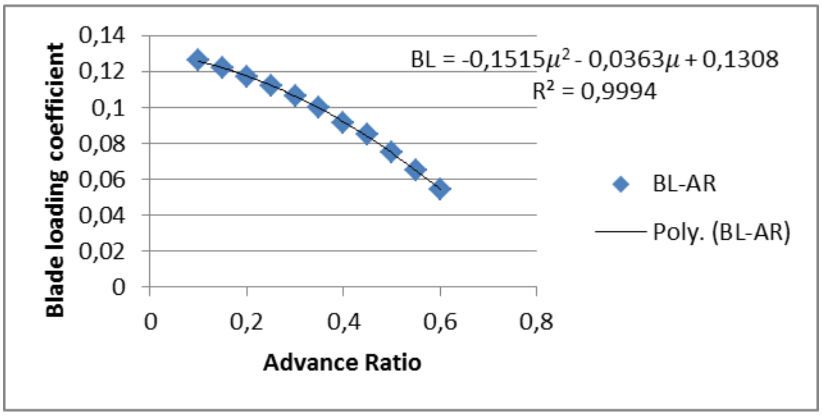

Figure 3.2 : Blade loading coefficient advance ratio variation

As it can be seen from above blade loading can be written with an acceptable variation 0.9994;

$$BL = -0.1515\mu^2 - 0.0363\mu + 0.1308 \tag{3.11}$$

BL=0.1208240

The first aproximation for solidity can be calculated by using

$$BL = \frac{C_T}{\sigma} \tag{3.12}$$

Solidity is

$$\sigma = \frac{C_T}{BL} = \frac{9{,}69 \text{ x } 10^{-4}}{0.12824} = 0.0080207 \tag{3.13}$$

Two rotor blades will be used. Number of blades is a function of radius, solidity, vibration and weight. The blade chord can be determined by

$$c = \frac{\pi R \sigma}{N_b} = \frac{\pi(1.228)(0.0080207)}{2} = 0.01547 \tag{3.14}$$

For a helicopter rotor, the aspect ratio can be defined as the radius over chord. Historically, the main rotor aspect ratio has between 15 to 20 [3] .

$$AR = \frac{R}{c} = \frac{1.228}{0.01547} = 79.37 \tag{3.15}$$

This value is not valid since the interval is 15<AR<20. The rotational speed will be decreased in order to obtain an acceptable aspect ratio.

Define as tip speed equals to V_{tip}=118 m/s. The new thrust coefficient is

$$C_T = \frac{T}{\rho A V_{tip}{}^2} = \frac{(28)(9.81)}{(1.225)\pi(1.228)^2(118)^2} = 3.399x10^{-3} \tag{3.16}$$

Advance ratio

$$\mu = \frac{V_{f,max}}{V_{tip}} = \frac{(130)(1000)}{(118)(60)(60)} = 0.3060 \tag{3.17}$$

Blade loading coefficient;

$$BL = -0.1515\mu^2 - 0.0363\mu + 0.1308 = 0.105502 \tag{3.18}$$

Solidity;

$$\sigma = \frac{C_T}{BL} = \frac{3.399x10^{-3}}{0.105502} = 0.03221 \tag{3.19}$$

Aspect ratio;

$$AR = \frac{R}{c} = \frac{N_b}{\pi\sigma} = \frac{2}{\pi(0.03221)} = 19.75 \quad \textbf{(3.20)}$$

This aspect ratio value can be accepted.

The new chord value is;

$$c = \frac{\pi R\sigma}{N_b} = \frac{\pi(1.228)(0.03221)}{2} = 0.0621\ m \quad \textbf{(3.21)}$$

Mean rotor lift coefficient can be calculated by the equation that's given below;

$$\overline{C_L} = 6\frac{C_T}{\sigma} = 6.BL = 0.633017 \quad \textbf{(3.22)}$$

Typical values of mean lift coefficient for helicopters range from about 0.4 to 0.7. The value found above is acceptable.

While selecting an airfoil there are important points needs to be taken into account.

These are high stall angle of attack to avoid stall on the retreating side of the blade,high lift curve slope to avoid operation at high angles of attack, high maximum lift coefficient to provide the necessary lift, high drag divergence Mach number to avoid comppresibility effects on the advancing side of the blade, low drag at combinations of angles of attack and Mach numbers representing conditions at hover and cruise and low pitching moments to avoid high control loads excessice twisting of the blades [1].

BoeingVR-7 airfoil is selected for a acceptable hovering performance with a reasonable forward flight performance and profile drag coefficient and lift curve slope will be determined.This airfoil is a high-speed airfoil but it has also has a good lifting capability. Other more efficient for hovering high-lift airfoils like OA-214,0A 212 RC(4)-10 was not selected because also a reasonable higher speed is required. On the other hand high-speed airfoils like Bell FX-69-H-083, RC(5)-10, VR-15,OA-206 was not prefferred in order to maintain a good hovering performance.

To determine which Reynolds number is selected in order to use airfoil charts listed by the manufacturer, kinematic viscosity at sea level (later, 2000 meters altitude), cruise speed ,the retreating blade's velocity at r=0.2 and chord values are necessary.

Required cruise speed is 80 km/h. Chord value is c=0.0621m and kinematic viscosity which can be defined as $\nu = \frac{dynamic\ viscosity\ \mu}{density\ \rho_{@alt}}$ at 2000 meters altitude is $1.348x10^{-5}$[10] . At sea level,it is $1.466x10^{-5}\ m^2/_s$

$$Re = \frac{V_{cr}c}{\nu} for\ a\ fixed - wing \quad \textbf{(3.23)}$$

The airfoil data at 75000 and 100000 Reynolds number was found and the figures below was sketched [11].

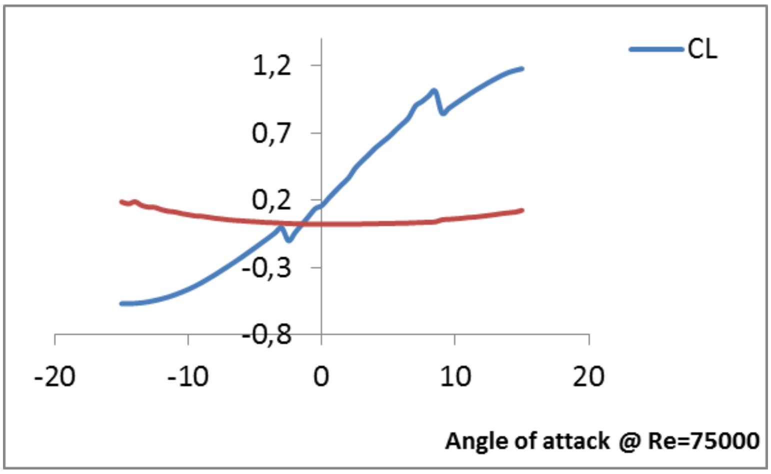

Figure 3.3 : Lift and drag coefficients of VR7 at 75000 Reynolds

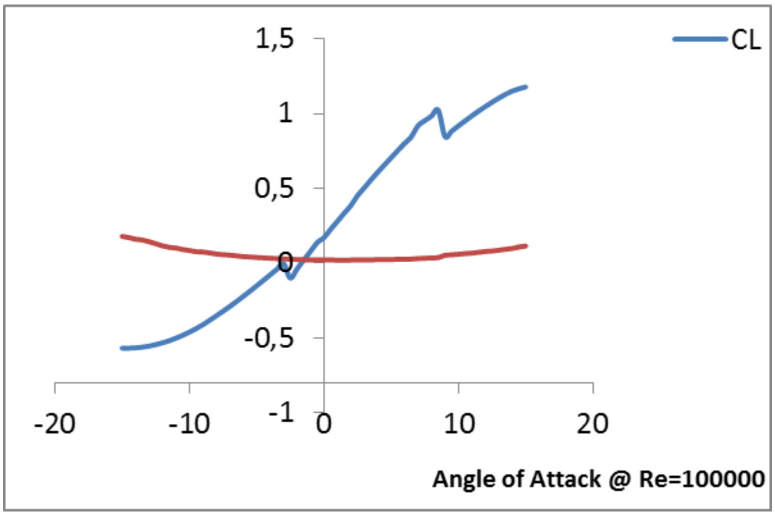

Figure 3.4 : Lift and drag coefficients of VR7 at 100000 Reynolds

For a rotary-wing vehicle,the lowest value of Reynolds number will be on the retreating blades. The blades velocity distibution for hover and forward flight can be seen at the figure below.

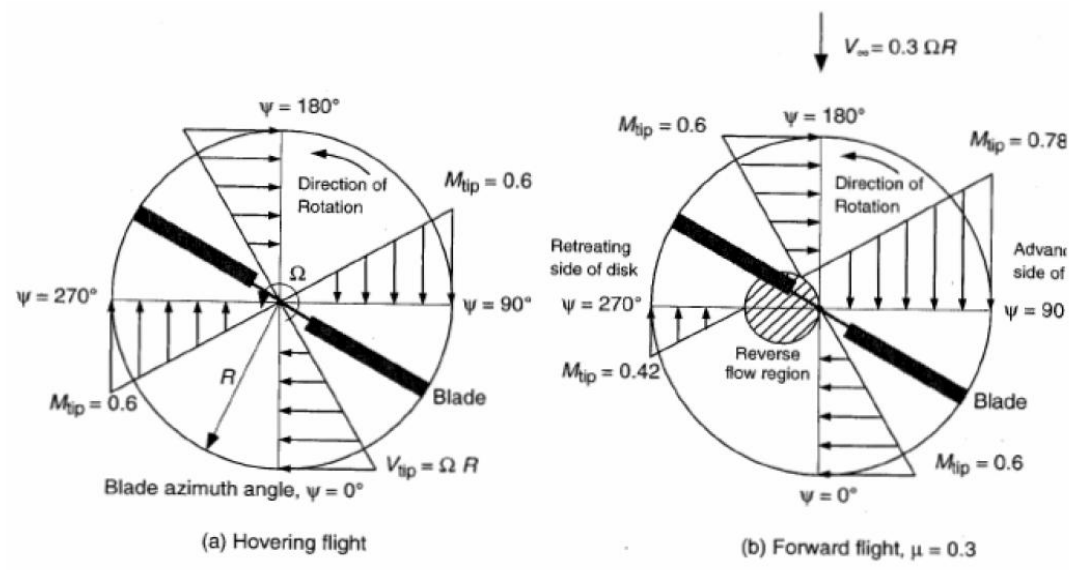

Figure 3.5 : Velocity distribution over the blade [4]

The lowest value of Reynolds during hover is on the retreating blade. At this design there will be a root cut-out at ratio r=0.2. So for r=0.2 it is necessary to calculate the Mach numbers and corresponding Reynolds numbers.

r is the ratio of location on the blade versus radius and it changes between 0 and 1.

Reynolds number can be stated as;

$$Re = \left(\frac{ac}{v}\right) M \tag{3.24}$$

where M is the Mach number and a is the speed of sound.

During forward flight there is a reverse flow region. This area's effect is neglected during this design process and to lower the effects of this region a root cut out is constructed.

The retreating side of the blade is the critical part and during forward flight an approximation needs to be done for deciding which section is selected for calculating the Reynolds number and corresponding drag coefficient. But profile drag coefficients are obtained from hovering rotor in this work and it is assumed that the value is not changed with increasing speed.

At retreating side, the lowest mach number and lowest Reynolds number is on the blade's root-cut out's end point(r=0.2) At this point the speed of blade's section is zero. The speed of sound can be defined as; [4]

$$a = 331.5 + (0.6T) \quad \textbf{(3.25)}$$

And the temperature change can be modelled with [4]

$$T = 15 - 0.000981\ x\ altitude\ in\ meters \quad \textbf{(3.26)}$$

Table 3.3: Inputs for Reynolds calculator

Tip speed	INPUT
Speed of sound at sea level	340.5 m/s
Speed of sound at 2000 meters	338.1228 m/s
Blade's speed at r=0.2	-
Mach number at hover for r=0.2 of retreating blade at sea level	OUTPUT
Mach number at hover for r=0.2 of retreating blade at 2000m	OUTPUT
Chord	INPUT
Kinematic viscosity at sea level	0.00001466
Kinematic viscosity at 2000 meters	1.35E-01
Reynolds number of retreating blade r=0.2 at sea level hover	OUTPUT
Reynolds number of retreating blade r=0.2 at 2000m hover	OUTPUT

At sea level speed of sound is 340.5 m/s and at 2000 meters the speed of sound is 338.12 m/

By using these calculator,for hover Reynolds numbers can be obtained and the power curve given before, can be used to estimate values for profile drag coefficients.

An assumption has to be made for VR7 airfoil for Reynolds number and profile drag coefficient. It is expected that for low Reynolds number, the drag coefficient is higher and decreasing with increasing Reynolds number. For relatively higher Reynolds number profile drag coefficient is approached as a constant.At this work,after the obtaining of the profile drag coefficients, it is assumed that they do not change with forward speed and only depends on profile and density.

The Reynolds number and the profile drag coefficients with respect to that Reynolds number can be seen at the table below.

Table 3.4 : Profile drag coefficients with respect to Reynolds numbers for VR7 [11]

Reynolds number	Cd (VR7)
25000	0.02819
50000	0.01981
75000	0.01668
100000	0.01464

For that table an equation can be constructed in order to determine the profile drag coefficients with respect to the Reynolds numbers that are worked on.

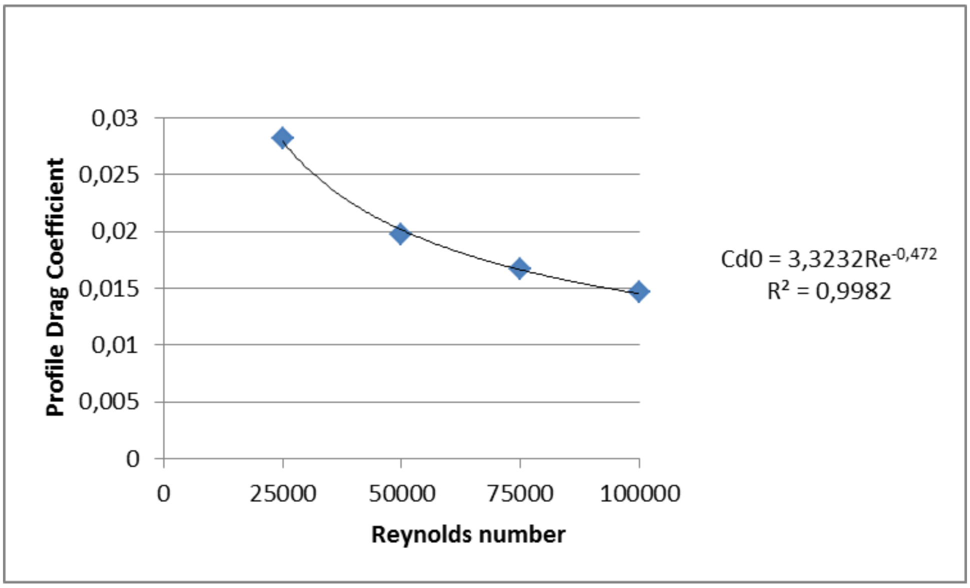

Figure 3.6 : Profile drag coefficient Reynolds variation of VR7

This assumption can not be made for detailed design of the helicopter but for conceptual and preliminary design process it is enough for modelling the Reynolds number effect on profile drag coefficients which will be used in process.

The hover values can be seen at the table below.

Table 3.5: Determination of Reynolds number at r=0.2

	Mach number at hover for r=0.2 of blade	Reynolds number	Profile drag coefficient
Sea level	0.068722467	97458.39533	0.014733
2000 meters	0.069205626	105241.8463	0.014161

The obtained drag coefficient by this method gives an over designed helicopter. Another approach is necessary. In detailed design process, the drag coefficient distribution on blades must be constructed. But in preliminary design, drag coefficient can be approached as a constant. The value which will be used for power calculations at hover will be assumed as the arithmetic mean of value of drag coefficients at stations that are constructed on the blades by r=0.1 interval. The average values of profile drag coefficients at sea level and 2000 meters altitude at hover were obtained as 0.009533 and 0.009164, respectively. The variation of profile drag coefficients can be seen at the figure below with respect to stations.

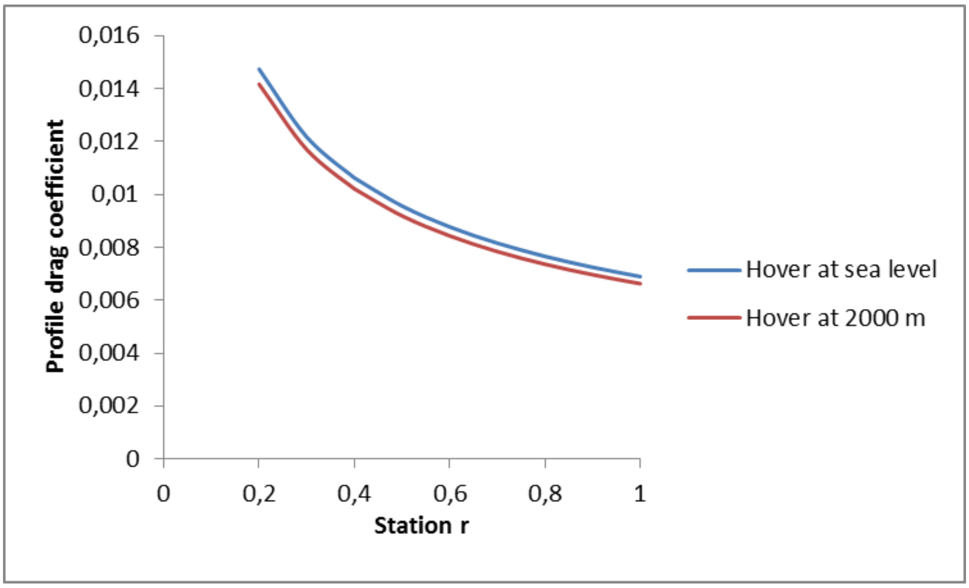

Figure 3.7 : Variation of profile drag coefficient with respect to blade stations

Mean lift coefficients which will be calculated during the design process will be used to determine the mean angle of attack or effective angle of attack from the tables for airfoil selected.

$$\overline{C_L} = 0.633017 \tag{3.27}$$

For this value of lift coefficent,mean angle of attack is between 4.5 and 5 degrees.It can be approximated as 4.3 degrees because at these values of angle of attack and lift coefficient the curve can be approximated linearly by interpolation with a small rate of error.This error is said to be relatively small for preliminary and conceptual design process but as for the profile drag coefficient, during the detailed design process this assumptions can not be used.

$$\alpha = 4.3\ deg = 0.07509\ rad \tag{3.28}$$

Lift curve slope can be calculated as;

$$C_L = C_{l,\alpha}\alpha \tag{3.29}$$

$$C_{l,\alpha} = 8.43/rad \tag{3.30}$$

Profile drag coefficient at hover at $\alpha = 0$ obtained from the average value of stations' values and can be shown below.

$$C_{d0} = 0.009533 \text{ for sea level} \tag{3.31}$$

$$C_{d0} = 0.009164 \text{ for 2000 meters} \tag{3.32}$$

The airfoil's shape is illustrated below.

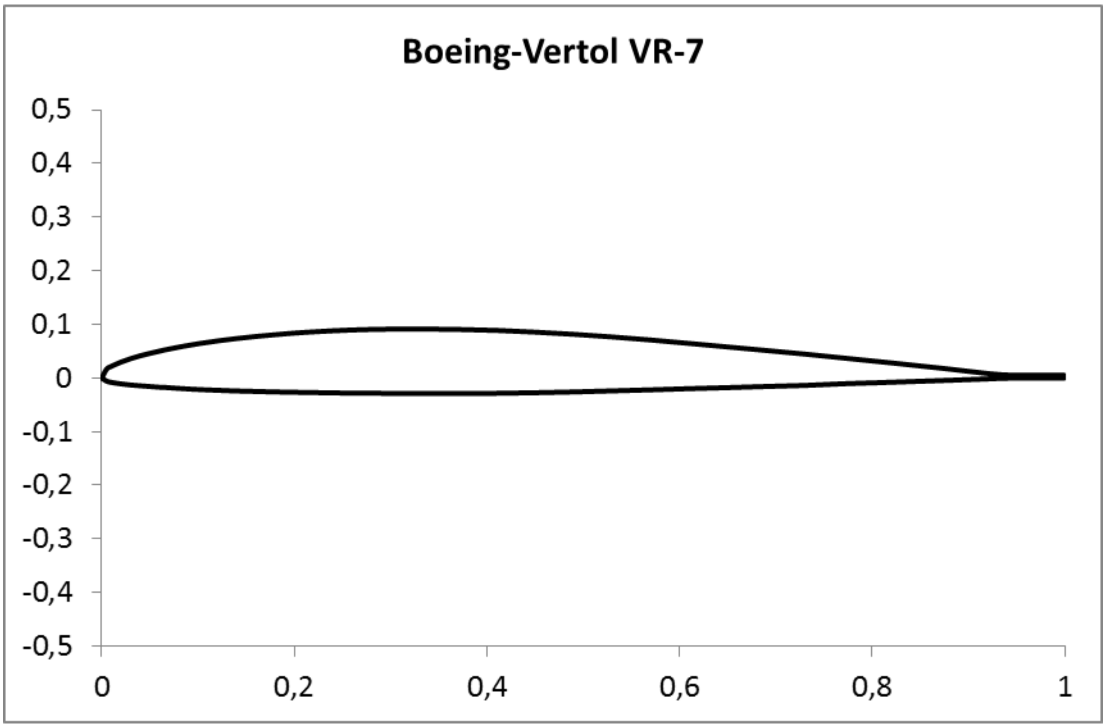

Figure 3.8 : VR7 airfoil's geometric shape

The total power required to hover OGE at sea level is the sum of induces power and profile power. A tip-loss effect factor is defined;

$$B = 1 - \frac{\sqrt{2C_T}}{N_b} \tag{3.33}$$

Gesgow and Myers suggest an empirical tip-loss factor based on blade geometry alone where [4]

$$B = 1 - \frac{c}{2R} \tag{3.34}$$

The last tip-loss function will be used at calculations.

$$B = 1 - \frac{c}{2R} = 1 - \frac{0.062149521}{2x1.227988} = 0.974694 \tag{3.35}$$

Also root-cut ratio should be defined. $r_0 = 0.2$

Effective rotor area is defined as;

$$A_e = \pi R^2 (B^2 - r_0{}^2) = 4.3111\ m^2 \tag{3.36}$$

Induced power factor can be defined approximately as $\kappa = 1.15$ [4]

$$P = P_i + P_0 = \frac{\kappa T^{\frac{3}{2}}}{\sqrt{2\rho A_e}} + \rho A_e V_{tip}^3 \left(\frac{\sigma C_{d0}}{8}\right) \tag{3.37}$$

$P_i = 1.490105\ kW$ at sea level **(3.38)**

$P_0 = 0.310390\ kW$ at sea level **(3.39)**

So the total power required to hover out-of-ground effect at sea level is;

$$P = P_i + P_0 = 1.800496\ kW \tag{3.40}$$

At 2000 meters altitude, for fixed design variables as radius,chord etc. ,the power needs of rotor is;

$$P = P_i + P_0 = 1.642116\ kW + 0.245691\ kW = 1.887808\ kW \tag{3.41}$$

3.2. Gross weight iteration

An iteration of gross weight will be done by using statistical data [3]. The unit conversion is done. This iteration process will go on until 5% error rate of gross weight will be reached.

$$W_{blades} = 0.0373 W_e R^{0.4} \sigma^{0.33} \tag{3.42}$$

$$W_{hub} = 8.169x10^{-3} W_e R^{0.42} \tag{3.43}$$

$$W_{main\ rotor} = W_{blades} + W_{hub} \quad \textbf{(3.44)}$$

$$Shaft\ Power = Engine\ output\ x\ 0.85 \quad \textbf{(3.45)}$$

$$W_{propulsion} = 0.8947\ x\ Shaft\ Power \quad \textbf{(3.46)}$$

$$W_{fuselage} = 0.21W_e \quad \textbf{(3.47)}$$

$$W_{controls} = W_{electrical} = 0.06W_e \quad \textbf{(3.48)}$$

$$W_{fixed\ equipment} = 0.28W_e = 3.032\ kg \quad \textbf{(3.49)}$$

$$W_e{}' = W_{main\ rotor} + W_{propulsion} + W_{fuselage} + W_{controls} + W_{electrical} + W_{fixed\ equipment} \quad \textbf{(3.50)}$$

$$GW' = W_e{}' + W_{fuel} + W_{payload} \quad \textbf{(3.51)}$$

Error percentage can be found by

$$error\% = \frac{GW - GW'}{GW} x100 \quad \textbf{(3.52)}$$

Table 3.6: First gross weight iteration

Previous gross weight (kg)	New gross weight (kg)	Error %	UL/GW
	28.00000000		0.47350
28.00000000	27.36589378	2.26	0.48452
27.36589378	25.77384568	1.46	0.49175
25.77384568	25.95563851	0.70	0.51087
25.95563851	26.07085898	0.40	0.50861
26.07085898	26.14389048	0.30	0.50719
26.14389048	26.19018272	0.20	0.50630
26.19018272	26.21952658	0.10	0.50573

The iteration process can be seen from the table above. Because the gross weight iteration needs geometrical data input for detecting new gross weight quantities like radius and solidity redesigned for all iteration steps. At last step a new gross weight is selected. Tip speed will need to be increase 117 to 118 m/s due to aspect ratio interval

$$GW=26.22\ kg \quad \textbf{(3.53)}$$

$$DL \cong DL \cong 16{,}932GW^{0.3694} = 56.59 \quad \textbf{(3.54)}$$

$$R = \sqrt{\frac{T}{\pi . DL}} = \sqrt{\frac{(26.22)(9.81)}{\pi(56.59)}} = 1.202\ m \tag{3.55}$$

$$\Omega = \frac{V_{tip,max}}{R} = \frac{(117)}{(1.202)} = 97.33\ rad/s \cong 929.43\ \text{rpm} \tag{3.56}$$

$$C_T = \frac{T}{\rho A V_{tip}{}^2} = \frac{(26.22)(9.81)}{(1.225)(\pi(1.202\ )^2)(117)^2} = 3.374x10^{-3} \tag{3.57}$$

Maximum advance ratio and blade loading coefficient are;

$$\mu = \frac{V_{f,max}}{V_{tip}} = \frac{(130)(1000)}{(117)(60)(60)} = 0.3086 \tag{3.58}$$

$$BL = -0.1515\mu^2 - 0.0363\mu + 0.1308 = 0.10516 \tag{3.59}$$

$$\sigma = \frac{C_T}{BL} = \frac{3.374x10^{-3}}{0.10516} = 0.03209 \tag{3.60}$$

$$AR = \frac{R}{c} = \frac{N_b}{\pi\sigma} = \frac{2}{\pi(0.03209)} = 19.83 \tag{3.61}$$

Aspect ratio should be between 15 and 20. It is an acceptable value

$$c = \frac{\pi R \sigma}{N_b} = \frac{\pi(1.202\ )(0.03209)}{2} = 0.0606 \tag{3.62}$$

$$\overline{C_L} = 6\frac{C_T}{\sigma} = 6.\,BL = 0.630986 \tag{3.63}$$

As it was explained at the first iteration step, the average values of profile drag coefficients will be used and for new design weight they need to determined again.

At hover the average drag coefficients of the stations located by $\Delta r = 0.1$ intervals at sea level and 2000 meters altitude are 0.0096445 and 0.0092702 respectively.

These average values can be seen at the tables given below. At first, the local blade station speeds are obtained in meters per second, then by using the speed of sound at the defined altitudes local Mach numbers are obtained. With tables of airfoil, local Reynolds numbers and corresponding local profile drag coefficients are defined. At last average values of these coefficients are obtained.

Table 3.7: Determination of drag coefficient at blade stations at sea level

r	Blade station speed (m/s)	Mach number	Reynolds number	Profile Drag coefficient
0.2	23.4	0.068722467	96728.51296	0.014735835
0.3	35.1	0.1030837	145092.7694	0.012169134
0.4	46.8	0.137444934	193457.0259	0.010624013
0.4	46.8	0.137444934	193457.0259	0.010624013
0.5	58.5	0.171806167	241821.2824	0.009561963
0.6	70.2	0.206167401	290185.5389	0.008773513
0.7	81.9	0.240528634	338549.7954	0.00815783
0.8	93.6	0.274889868	386914.0518	0.007659536
0.9	105.3	0.309251101	435278.3083	0.007245335
1	117	0.343612335	483642.5648	0.006893835
			Average	0.009644501

Table 3.8: Determination of drag coefficient at blade stations at 2000 meters

r	Blade station speed (m/s)	Mach number	Reynolds number	Profile Drag coefficient
0.2	23.4	0.069205626	105188.0424	0.014164079
0.3	35.1	0.103808439	157782.0636	0.011696967
0.4	46.8	0.138411252	210376.0849	0.010211797
0.4	46.8	0.138411252	210376.0849	0.010211797
0.5	58.5	0.173014065	262970.1061	0.009190956
0.6	70.2	0.207616878	315564.1273	0.008433097
0.7	81.9	0.242219691	368158.1485	0.007841303
0.8	93.6	0.276822504	420752.1697	0.007362343
0.9	105.3	0.311425316	473346.1909	0.006964213
1	117	0.346028129	525940.2122	0.006626352
			Average	0.009270291

Table 3.9: Average profile drag coefficients

	Profile drag coefficient
Sea level	0.0096445
2000 meters	0.0092702

At forward flight for all airspeeds independently, all stations profile drag coefficients have been found and the average values of these values are stated at the table below. At 2000 meters altitude all calculations were redone.

$$\overline{C_L} = 0.630986 \quad \textbf{(3.64)}$$

For this value of lift coefficent, mean angle of attack can be approximated as 4.5 degrees.

$$\alpha = 4.5\, deg = 0.0785\, rad \quad \textbf{(3.65)}$$

Lift curve slope can be calculated as;

$$C_L = C_{l,\alpha}\alpha \quad \textbf{(3.66)}$$

$$C_{l,\alpha} = 8.033\ /rad \quad \textbf{(3.67)}$$

Power required to hover OGE can be calculated as;

$$B = 1 - \frac{c}{2R} = 1 - \frac{0.0606}{2x1.202} = 0.974796 \quad \textbf{(3.68)}$$

Root cut out ratio is 0.2 and induced power factor is 1.15

$$A_e = \pi R^2 (B^2 - {r_0}^2) = 4.1371\ m^2 \quad \textbf{(3.69)}$$

$$P = P_i + P_0 = \frac{\kappa W^{\frac{3}{2}}}{\sqrt{2\rho A_e}} + \rho A_e V_{tip}^3 \left(\frac{\sigma C_{d0}}{8}\right) \quad \textbf{(3.70)}$$

$P_i = 1.4901\ kW$ at sea level **(3.71)**

$P_0 = 0.314\ kW$ at sea level **(3.72)**

So the total power required to hover out-of-ground effect at sea level is;

$$P = P_i + P_0 = 1.804\ kW \quad \textbf{(3.73)}$$

Induced power is approximately 75% of total power.

The power required for rotor at hover at 2000 meters altitude is;

$$P = P_i + P_0 = \frac{\kappa T^{\frac{3}{2}}}{\sqrt{2\rho A_e}} + \rho A_e V_{tip}^3 \frac{\sigma C_{d0}}{8} = 1.642 + 0.248 = 1.8906\ kW \quad \textbf{(3.74)}$$

3.3. Power required to hover IGE

The profile does not change for hover IGE, but the induced power will be smaller. In all cases ground effect will be negligible for rotors hovering greater than three rotor radii above the ground [4].

Hayden assumed that only the induced part of the power is influenced by the ground[4]. Assume that ground effect at 2 meters altitude will be calculated;

$$P = P_0 + k_G P_i \tag{3.75}$$

$$k_G = \frac{1}{A+B(\frac{2R}{z})^2} \tag{3.76}$$

Where A= 0.9926 and B= 0.0379

$$k_G = \frac{1}{0.9926+0.0379(\frac{2(1.202)}{2})^2} = 0.954783 \tag{3.77}$$

$$P = P_0 + k_G P_i = 0.314 + (0.954783)(1.4901) = 1.7367 \ kW \tag{3.78}$$

Total power required to hover in-ground effect at two meters altitude is $1.7367\ kW$ which is smaller than power required to hover out-of-ground effect $1.804\ kW$ as expected. It can be seen that above the altitude 3.6 meters there is no ground effect to provide advantages to helicopter.

3.4. Power calculations at forward flight

For advancing ratios larger than 0.1, induced power for forward flight can be approximated by Glauert high-speed formula which is given below; [4]

$$P_{i,f} = \frac{\kappa T^2}{2\rho A_e V_{forward}} \tag{3.79}$$

Because for high speeds, $V_{forward} \gg v_i$, thrust equation can be written as;

$$T = 2\rho A v_i \sqrt{V_{forward}^2 + 2V_{forward} v_i sin\alpha + v_i^2} = 2\rho A v_i V_{forward} \tag{3.80}$$

And the induced power is $P_{i,f} = \kappa T v_i$

Profile power for forward flight can be corrected with Stepniewski constant K=4.7 and it can be written as [4];

$$P_{0,f} = (1 + K\mu^2)P_0 \tag{3.81}$$

For climb speed occurance climb power can be defined as;

$$P_{cl} = T.V_c \tag{3.82}$$

where V_c is climb celocity and T is the thrust.

Parasitic power results from viscous shear effects and flow seperation on the airframe,rotor hub and so on. Parasitic power can be calculated by equivalent flat plate approach [4].

$$P_p = 0.5\,\rho\,{V_{forw}}^3\,f \tag{3.83}$$

where f is equivalent flat plate area. This value can be calculated from helicopter data history [3].

$$f = 0.00217\,GW^{0.8357} \tag{3.84}$$

Here gross weight's unit is kilogram and the exact formula from the reference shown above was changed into SI unit system.

$$f = 0.00217\,GW^{0.8357} = 0.00217\,(26.22)^{0.8357} = 0.03327\,m^2 \tag{3.85}$$

For example, with cruise level speed 80 km/h and 2000 meters altitude, the parasitic power is expected to be lower.

$$P_p = 0.5\,(1.0087)(22.22)^3(0.033327) = 184.4\,W = 0.1844\,kW \tag{3.86}$$

For high speed forward flight, parasitic power will be larger than the lower speed one.

Total main rotor power for forward flight will be calculated for two different altitudes (sea level and 2000 meters) and for a speed range 0-130 km/h. Chord and radius values will be fixed.

The calculator is constructed in Excel. The Matlab codes are not uısed at this stage. Two different density of air values can be seen. The inputs and outputs can be seen at the table given below.

Table 3.10: Forward flight power calculator

Gross weight	26.22 kg
Forward velocity in km/h	**INPUT**
Forward velocity in m/s	-------------------
Tip speed	117 m/s
Advance ratio	-------------------
Density	**1.225 or 1.0087 kg/m^3**
Main rotor radius	1.202819122 m
Number of blades	2
Main rotor chord	0.060630997 m
Profile drag coefficient	**INPUT**
Stepniewski constant	4.7
Equivalent flat plate area	0.033266803 m^2
Tip-loss effect	0.974796295
Root cut-out ratio	0.2
Effective rotor disk area	4.137143869 m^2
Induced power factor	1.15
Solidity	0.032090354
Profile power for hover	OUTPUT (kW)
Profile power for forward flight	OUTPUT (kW)
Induced power for hover	OUTPUT (kW)
Induced power in forward flight for nu>0.1	OUTPUT (kW)
Parasitic power for forward flight	OUTPUT (kW)
Total power for hover	**OUTPUT (kW)**
Total power for forward flight	**OUTPUT (kW)**

The important point in these calculations is remembering the induced power approximation of Glauert is only valid when advance ratio is higher than 0.1. For the speed values 10, 20, 30 and 40 km/h, the advance ratio is lower than 0.1; so Glauert's high-speed formula can not be used.

$$\mu = \frac{V_{forw}}{Vtip} > 0.1 \qquad \textbf{(3.87)}$$

By using this equation ,for forward speeds less than 42.12 km/h for defined tip speed 117 m/s, a new approximation for induced power is required.

$$P_{i,f} = \kappa T v_i \qquad \textbf{(3.88)}$$

where induced velocity can be stated as; [4]

$$(v_i)_{n+1} = \frac{{v_h}^2}{\sqrt{(V_{forw}\,Cos\alpha)^2 + (V_{forw}\,Sin\alpha + (v_i)_n)^2}} \tag{3.89}$$

A numerical approach is required. Here induced velocity at hover is;

$$v_h = \sqrt{\frac{T}{2\rho A}} \tag{3.90}$$

The error is acceptable and the convergence is said to occur if error rate which is stated below is smaller than 0.0005.

$$\varepsilon = \left\| \frac{(v_i)_{n+1} - (v_i)_n}{(v_i)_{n+1}} \right\| \tag{3.91}$$

These numerical approach can be approached by the Matlab code given above or by using Excel.

```
GW=input('Gross Weight');
ro=input('Density');
r=input('Radius');
c=input('chord');
rc=input('root cut-out ratio');
V_forw=input('Forward Flight Speed');
alfa=input('Effective angle of attack');
T=GW*(9.81)              %Thrust
B=1-(c/(2*r))            %Tip-loss factor
Ae=(pi*r^2)*((B^2)-(rc^2))  % Effective rotor disk area
v_h=sqrt(T/(2*ro*Ae))       % Induced velocity for hover
for v_i = 0.01:1:50
    vold=v_i;
    while (v_i-vold)/(v_i)<0.005
v_i=((v_h)^2)/(sqrt((V_forw*cos(alfa))^2+(V_forw*sin(alfa)+v_i)^2));
        v_i
    end
end
```

Figure 3.9 : Matlab code for induced velocity iteration

For 10 km/h speed at sea level the tables of iteration are given to show the process, for other speed values and altitude induced velocities are calculated and new forward flight tables are constructed.

Table 3.11: Induced velocity iteration

v_i (m/s)	v_i+1 (m/s)
0	9.135608661
2.601438526	6.42140156
3.5276232	5.447873909
4.023991762	5.009392535
4.28983731	4.796718253
4.429924769	4.690451048
4.502863435	4.636641925
4.540571423	4.609222682
4.559990113	4.595207292
4.569969685	4.588032117
4.575092844	4.584355882
4.577721432	4.582471585
4.579069721	4.581505567
4.579761202	4.58101027
4.580115806	4.580756306
4.580297647	4.580626083
4.580390893	4.580559308
4.580438708	4.580525068
4.580463227	4.58050751
4.5804758	4.580498507
4.580482247	4.580493891
4.580485553	4.580491523
4.580487248	4.580490309
4.580488117	4.580489687
4.580488563	4.580489368
4.580488791	4.580489204
4.580488909	4.58048912
4.580488969	4.580489077
4.580488999	4.580489055
4.580489015	4.580489044
4.580489023	4.580489038
4.580489028	4.580489035
4.58048903	4.580489033
4.580489031	4.580489033
4.580489031	4.580489032
4.580489032	4.580489032
4.580489032	4.580489032
4.580489032	4.580489032
4.580489032	4.580489032
4.580489032	4.580489032
4.580489032	4.580489032
4.580489032	4.580489032
4.580489032	4.580489032
4.580489032	4.580489032

Table 3.12 : Revised induced power values at sea level

Gross weight	**26.22**	**26.22**	**26.22**	**26.22**
Density(SEA LEVEL)	1.225	1.225	1.225	1.225
Radius	1.202819122	1.202819	1.202819	1.202819
Chord	0.060630997	0.060631	0.060631	0.060631
Root-cut out	0.2	0.2	0.2	0.2
Forward speed	2.778	5.556	8.334	11.112
Alfa(radian)	0.078539816	0.07854	0.07854	0.07854
Tip loss	0.974796295	0.974796	0.974796	0.974796
Effective disk area	4.137143867	4.137144	4.137144	4.137144
Thrust	257.2182	257.2182	257.2182	257.2182
Vh	5.037528236	5.037528	5.037528	5.037528
Forward speed km/h	**10**	**20**	**30**	**40**
Alfa(degrees)	4.5	4.5	4.5	4.5
Induced power factor	1.15	1.15	1.15	1.15
Obtained induced velocity	4.58048932	3.678179	2.818362	2.207275
Induced power at forward flight	**1.354913001**	**1.088009**	**0.833674**	**0.652914**

Table 3.13 : Revised induced power values 2000 meters

Gross weight	**26.22**	**26.22**	**26.22**	**26.22**
Density(2000 meters)	1.0087	1.0087	1.0087	1.0087
Radius	1.202819122	1.202819	1.202819	1.202819
Chord	0.060630997	0.060631	0.060631	0.060631
Root-cut out	0.2	0.2	0.2	0.2
Forward speed	2.778	5.556	8.334	11.112
Alfa(radian)	0.078539816	0.07854	0.07854	0.07854
Tip loss	0.974796295	0.974796	0.974796	0.974796
Effective disk area	4.137143867	4.137144	4.137144	4.137144
Thrust	257.2182	257.2182	257.2182	257.2182
Vh	5.551425645	5.551426	5.551426	5.551426
Forward speed km/h	**10**	**20**	**30**	**40**
Alfa(degrees)	4.5	4.5	4.5	4.5
Induced power factor	1.15	1.15	1.15	1.15
Obtained induced velocity	5.122765819	4.248555	3.342899	2.651348
Induced power at forward flight	**1.515318893**	1.256726	**0.988832**	**0.784271**

After the correction of induced power values below the advance ratio makes the Glauert's high speed formula wrong,new power tables can be revised. These revised forward flight power tables can be written as;

Table 3.14: Power values at forward flight at sea level

Forward Speed	Profile Power	Induced Power	Parasitic Power	Total Power at sea level
10	0.314852804	1.354913001	0.000436727	**1.670202532**
20	0.317348553	1.088008665	0.003493813	**1.408851031**
30	0.321508135	0.833674052	0.011791619	**1.166973806**
40	0.32733155	0.652914111	0.027950503	**1.008196164**
50	0.334818797	0.540464308	0.054590827	**0.929873932**
60	0.343969877	0.450386923	0.094332949	**0.888689749**
70	0.35478479	0.386045934	0.14979723	**0.890627954**
80	0.367263535	0.337790192	0.223604028	**0.928657755**
90	0.381406114	0.300257949	0.318373704	**1.000037767**
100	0.397212525	0.270232154	0.436726617	**1.104171296**
110	0.414682768	0.245665594	0.581283127	**1.241631489**
120	0.433816845	0.225193462	0.754663594	**1.413673901**
130	0.454614754	0.207870888	0.959488377	**1.621974019**

Table 3.15: Power values at forward flight at 2000 meters

	Profile Power	Induced Power	Parasitic Power	Total Power at 2000m
10	0.249197043	1.515318893	0.000359613	1.764875549
20	0.251172357	1.256726392	0.002876905	1.510775654
30	0.254464548	0.988832494	0.009709556	1.253006598
40	0.259073615	0.784271087	0.023015243	1.066359945
50	0.264999558	0.656358458	0.044951647	0.966309663
60	0.272242378	0.546965382	0.077676446	0.896884206
70	0.280802074	0.46882747	0.123347319	0.872976863
80	0.290678646	0.410224036	0.184121945	0.885024627
90	0.301872095	0.364643588	0.262158004	0.928673687
100	0.31438242	0.328179229	0.359613174	1.002174823
110	0.328209621	0.298344754	0.478645135	1.10519951
120	0.343353699	0.273482691	0.621411565	1.238247955
130	0.359814652	0.252445561	0.790070144	1.402330357

The variation of powers at sea level with respect to increasing forward speed can be seen at the figure below.

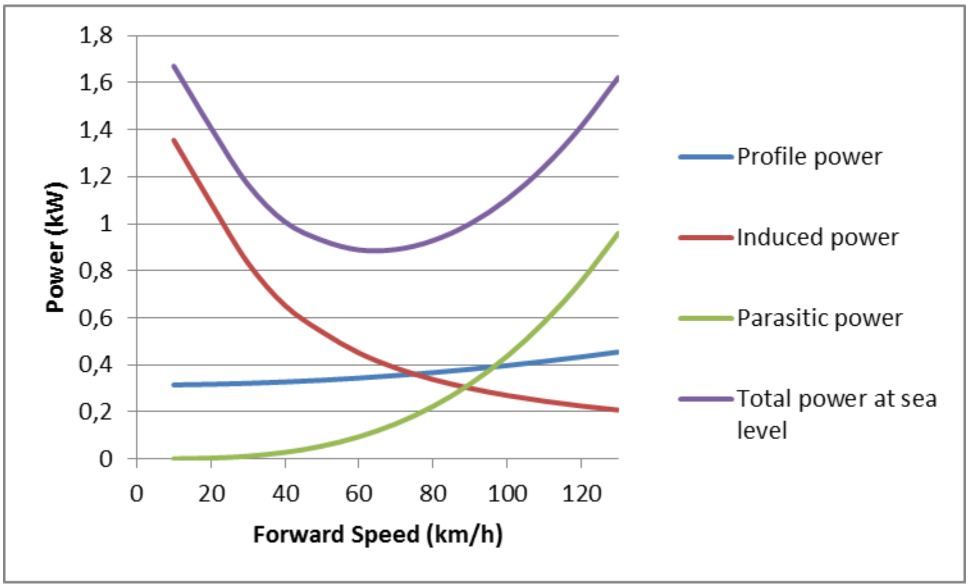

Figure 3.10 : Powers at forward flight at sea level

The tip Mach number of advancing side of the blade should also be checked.

$$M_{tip} = \frac{V_{forward}+V_{tip}}{Speed\ of\ sound} \tag{3.92}$$

Table 3.16: Tip mach number at sea level

Forward speed km/h	Forward speed m/s	Tip Mach number
10	2.777777778	0.351770272
20	5.555555556	0.35992821
30	8.333333333	0.368086148
40	11.11111111	0.376244085
50	13.88888889	0.384402023
60	16.66666667	0.392559961
70	19.44444444	0.400717899
80	22.22222222	0.408875836
90	25.00000000	0.417033774
100	27.77777778	0.425191712
110	30.55555556	0.433349649
120	33.33333333	0.441507587
130	36.11111111	0.449665525
Tip speed	117	At sea level
Speed of sound	340.5	

Table 3.17: Tip mach number at 2000 meters

Forward speed km/h	Forward speed m/s	Tip Mach number
10	2.777777778	0.354243422
20	5.555555556	0.362458715
30	8.333333333	0.370674008
40	11.11111111	0.3788893
50	13.88888889	0.387104593
60	16.66666667	0.395319886
70	19.44444444	0.403535178
80	22.22222222	0.411750471
90	25.00000000	0.419965764
100	27.77777778	0.428181057
110	30.55555556	0.436396349
120	33.33333333	0.444611642
130	36.11111111	0.452826935
Tip speed	117.0000000	At 2000 meters
Speed of sound	338.1228	

All these power calculations are done considering a zero twist blade. At this work, modified momentum theory is used for forward flight analysis and blade element momentum theory approach will be used at the end of this chapter considering twisted blades at hover. For the zero twist, results can be seen at the tables given before and for twisted blades, results can be seen the table at the end of this chapter.

3.5. Twist rate calculation with BEMT

At this section an estimation of twist rate will be done. All other geometric specifications of main rotor will be fixed.

Thrust coefficient taken from modified momentum theory will be used to determine which twist rate should be used from BET analysis. The twist angle which has obtained from BET, will be used in BEMT to determine inflow ratios and thrust coefficients at the stations respectively.

For twist angles between zero and fifteen; twenty iterations has been made by the formula given above [4].

$$C_T = \frac{1}{2}\sigma C_{l\alpha}\left[\frac{\theta_0}{3} - \frac{1}{2B}\sqrt{\frac{C_T}{2}}\right] \quad \textbf{(3.93)}$$

$C_{l\alpha}$ is 8.033 /rad, solidity σ is 0.032090354. Tip loss factor is 0.974796295. Twist angle values are given 0 to 15 and twenty iterations has been made in order to maintain the last thrust coefficient values.These values are given below. The thrust coefficient which has been obtained before is found by using disk area.At this work thrust coefficient will be found by using effective disk area. The procedure starts with momentum theory to obtain an estimated thrust coefficient value. This step and the results can be seen at the table below.

Table 3.18: Modified momentum theory output of BEMT calculator

Ideal Power (P)	**1.2957439**
Thrust (T)	257.2182000
Induced Velocity (vi)	5.0375282
Actual Power (P)	**1.8044372**
Power Coefficient (Cp)	0.0002223
Induced Power Coefficient (Cpi)	0.0001836
Profile Power Coefficient (Cp0)	0.0000387
Thrust Coefficient (Ct)	0.0037076

The momentum theory has been used to maintain a thrust coefficient value.The value obtained from modified momentum theory has been found as 0.0037076. At the previous steps it has been found differently.It is the result of modified momentum theory;it uses effective rotor disk area in calculations.

Table 3.19: BET output of BEMT calculator up to seven degrees

Teta(rad)	**0**	**0.01745**	**0.03491**	**0.05236**	**0.069813**	**0.087266**	**0.10472**	**0.12217**
Teta(deg)	0	1	2	3	4	5	6	7
1	-0.0005	0.00028	0.00103	0.001784	0.002534	0.003285	0.004036	0.00479
2		-4E-05	-3E-06	0.000275	0.000647	0.001071	0.001531	0.00202
3		0.00047	0.00142	0.001475	0.001813	0.002222	0.002673	0.00315
4		-0.0003	-0.00026	0.000454	0.00101	0.001547	0.002084	0.00263
5			0.00074	0.001254	0.001515	0.001912	0.002367	0.00286
6			0.00023	0.000595	0.001181	0.001707	0.002227	0.00275
7			0.0008	0.001111	0.001394	0.00182	0.002295	0.0028
8			0.00018	0.000692	0.001255	0.001757	0.002262	0.00278
9			0.00088	0.001021	0.001345	0.001792	0.002278	0.00279
10			0.00012	0.000757	0.001286	0.001772	0.00227	0.00278
11			0.001	0.000964	0.001324	0.001783	0.002274	0.00279
12			2.4E-05	0.000799	0.0013	0.001777	0.002272	0.00278
13			0.00127	0.000929	0.001315	0.00178	0.002273	0.00278
14			-0.00017	0.000825	0.001305	0.001779	0.002273	0.00278
15			0.0009	0.000908	0.001312	0.00178	0.002273	0.00278
16			0.0001	0.000842	0.001308	0.001779	0.002273	0.00278
17			0.00103	0.000894	0.00131	0.001779	0.002273	0.00278
18			-2.2E-06	0.000853	0.001309	0.001779	0.002273	0.00278
19			0.00143	0.000885	0.00131	0.001779	0.002273	0.00278
20			**-0.00027**	**0.000859**	**0.001309**	**0.001779**	**0.002273**	**0.00278**

Table 3.20: BET output of BEMT calculator between 7 and 15 degrees twist angle

Teta(rad)	0,139626	0,15708	0,174533	0,191986	0,20944	0,226893	0,244346	0,261799
Teta(deg)	8	9	10	11	12	13	14	15
1	0,005537	0,006287	0,007038	0,007789	0,008539	0,00929	0,01004	0,010791
2	0,002523	0,003045	0,00358	0,004127	0,004683	0,005247	0,005819	0,006398
3	0,003654	0,004173	0,004706	0,00525	0,005805	0,006368	0,006939	0,007516
4	0,003176	0,003732	0,004296	0,004866	0,005442	0,006024	0,00661	0,007202
5	0,003368	0,003897	0,004439	0,004992	0,005555	0,006126	0,006704	0,007288
6	0,003289	0,003834	0,004388	0,00495	0,005519	0,006095	0,006677	0,007264
7	0,003321	0,003858	0,004406	0,004964	0,005531	0,006104	0,006685	0,007271
8	0,003308	0,003849	0,0044	0,00496	0,005527	0,006102	0,006682	0,007269
9	0,003313	0,003852	0,004402	0,004961	0,005528	0,006102	0,006683	0,007269
10	0,003311	0,003851	0,004401	0,004961	0,005528	0,006102	0,006683	0,007269
11	0,003312	0,003851	0,004401	0,004961	0,005528	0,006102	0,006683	0,007269
12	0,003312	0,003851	0,004401	0,004961	0,005528	0,006102	0,006683	0,007269
13	0,003312	0,003851	0,004401	0,004961	0,005528	0,006102	0,006683	0,007269
14	0,003312	0,003851	0,004401	0,004961	0,005528	0,006102	0,006683	0,007269
15	0,003312	0,003851	0,004401	0,004961	0,005528	0,006102	0,006683	0,007269
16	0,003312	0,003851	0,004401	0,004961	0,005528	0,006102	0,006683	0,007269
17	0,003312	0,003851	0,004401	0,004961	0,005528	0,006102	0,006683	0,007269
18	0,003312	0,003851	0,004401	0,004961	0,005528	0,006102	0,006683	0,007269
19	0,003312	0,003851	0,004401	0,004961	0,005528	0,006102	0,006683	0,007269
20	**0,003312**	**0,003851**	**0,004401**	**0,004961**	**0,005528**	**0,006102**	**0,006683**	**0,007269**

The thrust coefficient value obtained from modified momentum theory has been found as 0.0037076. As it can be seen from the figure below, twist angle should be 8.8 degrees(actually by the figure below it is 8.8152 degrees,but it was approximated to 8.8 degrees) .

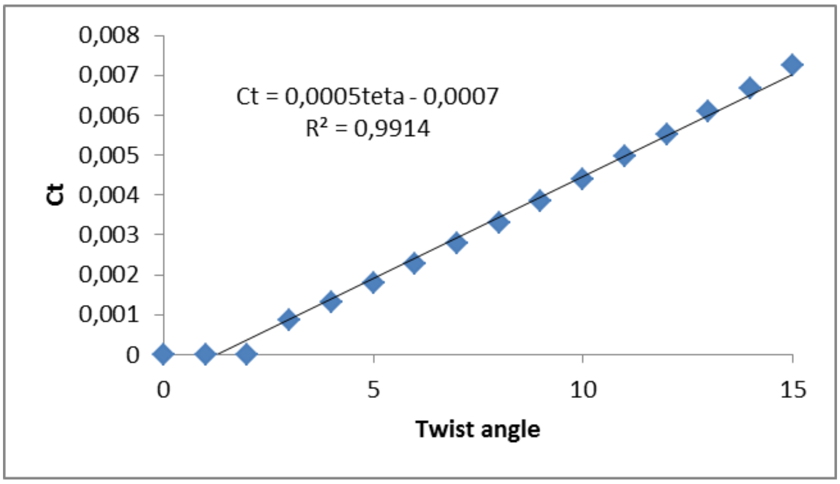

Figure 3.11 : Variation of thrust coefficient with respect to twist angle

Power coefficient can be defined as the formula given below by the uniform inflow and constant drag coefficient assumptions [4]

$$C_p = \lambda C_T + \frac{\sigma C_{do}}{8} \quad \textbf{(3.94)}$$

In hover with uniform inflow the equation given above, returns into simple momentum theory because for this assumptions $\lambda = \sqrt{C_T/2}$ .So;

$$C_p = \frac{\kappa C_T^{3/2}}{\sqrt{2}} + \frac{\sigma C_{do}}{8} \quad \textbf{(3.95)}$$

The second term is the extra power predicted by the BET that is required overcome profile drag of blades, which has been already introduced in momentum theory.

So,the total power can be written as;

$$P = \rho A_e V_{tip}^3 C_p \quad \textbf{(3.96)}$$

BET analysis will not be used at here to calculate total power. BEMT will be used by using 40 stations at the blade. Firstly, inflow ratios at the stations will be calculated and then, thrust and power coefficients will be found by using these inflow ratio values. The estimated value for twist angle from BET will be used with a linear twist distribution assumption.

Blade element momentum theory was used for power calculations for twisted blades.

The inflow ratio on each station can be found by; [4]

$$\lambda(r_n) = \frac{\sigma C_{l\alpha}}{16} \left(\sqrt{1 + \frac{32}{\sigma C_{l\alpha}} \theta(r_n) r_n} - 1 \right) \quad \textbf{(3.97)}$$

Where n=(1,N) element station number, r_n and $\theta(r_n)$ are the radius and pitch angle at the mid-span of each of the N stations. Here, N equals to 40. Thrust coefficients can be calculated and summed by rectangular approach.

$$\Delta C_{T_n} = \frac{\sigma C_{l\alpha}}{2} \left(\theta(r_n) {r_n}^2 - \lambda(r_n) r_n \right) \Delta r \quad \textbf{(3.98)}$$

$$C_T = \sum_{n=1}^{N} \Delta C_{T_n} \quad \textbf{(3.99)}$$

The 8.8 twist angle is used in calculations.

The tables of this process can be seen from the table for sea level computations below.

Table 3.21: Input of BEMT

Seçilen twist	**8.8**
Effective AoA	4.4
Delta r	0.0225

Table 3.22: Output of BEMT

Section	Inflow ratio	Ct	Cpi
0,10000000	0,00000	0,00000000000	0,00000000000
0,11125000	0,01171	0,00000137329	0,00000003217
0,13375000	0,01340	0,00000216138	0,00000005792
0,15625000	0,01496	0,00000314606	0,00000009411
0,17875000	0,01640	0,00000432832	0,00000014199
0,20125000	0,01775	0,00000570686	0,00000020260
0,22375000	0,01901	0,00000727862	0,00000027676
0,24625000	0,02020	0,00000903921	0,00000036510
0,26875000	0,02131	0,00001098319	0,00000046809
0,29125000	0,02236	0,00001310427	0,00000058600
0,31375000	0,02335	0,00001539543	0,00000071896
0,33625000	0,02429	0,00001784912	0,00000086697
0,35875000	0,02517	0,00002045726	0,00000102987
0,38125000	0,02601	0,00002321140	0,00000120741
0,40375000	0,02680	0,00002610269	0,00000139920
0,42625000	0,02755	0,00002912203	0,00000160476
0,44875000	0,02826	0,00003226004	0,00000182349
0,47125000	0,02893	0,00003550711	0,00000205474
0,49375000	0,02957	0,00003885344	0,00000229773
0,51625000	0,03017	0,00004228908	0,00000255165
0,53875000	0,03074	0,00004580391	0,00000281559
0,56125000	0,03127	0,00004938769	0,00000308858
0,58375000	0,03177	0,00005303008	0,00000336960
0,60625000	0,03224	0,00005672064	0,00000365759
0,62875000	0,03268	0,00006044884	0,00000395141
0,65125000	0,03310	0,00006420410	0,00000424990
0,67375000	0,03348	0,00006797577	0,00000455188
0,69625000	0,03384	0,00007175316	0,00000485610
0,71875000	0,03417	0,00007552554	0,00000516132
0,74125000	0,03447	0,00007928217	0,00000546625
0,76375000	0,03475	0,00008301227	0,00000576961
0,78625000	0,03500	0,00008670505	0,00000607009
0,80875000	0,03523	0,00009034975	0,00000636637
0,83125000	0,03543	0,00009393559	0,00000665715
0,85375000	0,03561	0,00009745180	0,00000694110
0,87625000	0,03577	0,00010088766	0,00000721692
0,89875000	0,03590	0,00010423245	0,00000748331
0,92125000	0,03600	0,00010747551	0,00000773899
0,94375000	0,03609	0,00011060623	0,00000798270
0,96625000	0,03615	0,00011361403	0,00000821319
0,98875000	0,03618	0,00011648843	0,00000842926

The thrust coefficients at each station is summed and the obtained result is 0.00206705945. The induced power coefficients are also summed up and profile power coefficient is obtained with the values of profile drag coefficient and solidity. These two power coefficient is summed and total power coefficient is obtained.This coefficient is used to obtain total power value at hover by BEMT. The total out of ground effect power at hover obtained by BEMT at sea level and 2000 meters altitude can be stated as;

Table 3.23: Total power with BEMT

	Sea level	**2000 meters**
Total Power	1.432984 kW	1.565378 kW

When the values for 2000 meters altitude were obtained, the twist angle founded from BET was changed. The value was 10.1 degree approximately. It means at that altitude, the value of required thrust can only be obtained by equal or higher values of twist.This means for a fixed twist more power is needed.

BET can be used to determine at what rate should the blades have and can be used to estimate how much power needed during the hover mission. The modified momentum theory is nearly have the same error rates with BEMT but the momentum theory does not contain the inflow on the blades,twist rates,etc.

Main rotor calculations is completed and the tail rotor design with respect to these power requirements will be done at Chapter 4. These values which were used here should not be written as with too mant unnecessary decimals, but these values used here are calculated using Microsoft Excel and nothing was changed. The Excel format and Matlab code of design uses same decimal values.

At the end of this chapter, the values calculated here, namely the new concept, should be shown in one table.

At cruise speed 80 km/h, helicopter’s some specifications can be summarized as

Table 3.24: Helicopter's specifications

Gross weight	26.22 kg
Required cruise speed	80 km/h
Maximum Forward speed	130 km/h
Blade tip speed	117 m/s
Thrust coeeficient	0.003374764
Solidity	0.032090354
Maximum advance ratio	0.308641
Mean lift coefficient	0.630986557
Airfoil	Boeing-Vertol VR7
Main rotor radius	1.202819122 m
Chord	0.060690997
Blade twist	8.8 degrees
Airfoil lift-curve slope	8.033 /rad
Airfoil profile drag coefficient	0.014683642 at sea level
Number of blades	2
Aspect ratio of main rotor	19.93
Induced power at hover	1.490105537 kW
Profile power at hover	0.311020886 kW
Total power OGE at hover	1.804126422kW
Total power IGE at hover	1.7367 kW for 2 meters
Power at hover with twisted blades	1.432984 kW
Induced power at forward flight (80km/h)	0.337790192 kW
Profile power at forward flight(80km/h)	0.3672635 kW
Parasitic power at forward flight(80km/h)	0.223604028 kW
Total power at forward flight(80km/h)	0.928657 kW

4. TAIL ROTOR DESIGN

Tail rotor provides an anti-torque force to encounter the torque of the main rotor on the airframe. It also gives yaw stability and makes the control of aircraft on the yaw axis desirable. Another advantage of tail rotor its weathercock stability [4]. For example if the aircraft is yawed nose-left ,then the tail rotor will experince an additional climb performance.

4.1. Preliminary tail rotor geometry

Radius of the tail rotor is roughly one sixth of the radius of the main rotor and the tip speed of tail rotor is approximately is the same as the main rotor tip speed which means the rotational velocities of the tail rotors are roughly six times of the main rotor rotational velocity which is an important issue needs to be solved for rotor noise levels problems [4]. For small helicopters, the ratio of the radius of the main rotor and tail rotor can be approximated as six, but with increasing gross weight this ratio will be much more smaller than that [3]. At this work, the ratio will be accepted as six and the tip speed of the tail rotor approximated as 117 m/s. NACA 0012 airfoil will be used for tail rotor blades.

$$R_{tr} = \frac{1{,}202819}{6} = 0{,}20047m \tag{4.1}$$

$$V_{tip,tr} = 117\ m/s \tag{4.2}$$

$$\Omega_{tr} = \frac{V_{tip,tr}}{R_{tr}} = 638.297\ rad/s = 5573.245\ rpm \tag{4.3}$$

This RPM value is too high, but decreasing the tip speed gives increase in power needs of the tail rotor. The airfoil is used for main rotor is different so the drag coefficient of the tail rotor can be stated as 0.011 for NACA 0012 for hovering performance [12]. As it was mentioned at Chapter 2 in trade studies, the number of blades of the tail rotor is 2.

The distance from the main rotor shaft and the tail rotor shaft can be approximated as;

$$L = R_{mr} + R_{tr} + 0.2 = 1.603288\ m \quad \textbf{(4.4)}$$

These 0.2 value is an approximation and it is defined as 0.5 ft in English unit system [3].

The aspect ratios for tail rotors can be changed between 4.5 and 8 [5]. By the definition an approximated chord value can be obtained for an aspect ratio as 6.

$$c_{tr} = \frac{R_{tr}}{AR} = 0.03341\ m \quad \textbf{(4.5)}$$

4.2. Power required

Power needed for tail rotor at hover will be much more than power needed at forward flight. But at high forward speeds,the power sometimes can exceed the hovering value.

4.2.1. Power required at hover

The thrust needed for tail rotor can be stated as;

$$T_{tr} = \frac{P_{total,mr}}{\Omega_{mr} L} \quad \textbf{(4.6)}$$

And the thrust coefficient can be found by;

$$C_{T,tr} = \frac{T_{tr}}{\rho A V_{tip,tr}{}^{2}} \quad \textbf{(4.7)}$$

Tip-loss factor B will be calculated with;

$$B = 1 - \frac{c}{2R} \quad \textbf{(4.8)}$$

or tip-loss factor can be stated with thrust coefficient as;

$$B = 1 - \frac{\sqrt{2C_{T,tr}}}{N_{b,tr}} \quad \textbf{(4.9)}$$

The induced and profile powers for tail rotor can be stated as;

$$(P_i)_{tr} = \frac{1}{B} \frac{T_{tr}^{3/2}}{\sqrt{2\rho A_{tr}}} \quad \textbf{(4.10)}$$

$$(P_0)_{tr} = \frac{\sigma C_{do}}{8} \rho A_{tr} {V_{tip,tr}}^3 \quad \textbf{(4.11)}$$

$$(P_{total,hover})_{tr} = (P_i)_{tr} + (P_0)_{tr} \quad \textbf{(4.12)}$$

The power values for the hovering can be seen from the table above. The profile drag coefficient is assumed to be constant as NACA 0012 airfoil is used for tail rotor zero twisted blades.

Table 4.1: Tail rotor power for untwisted main rotor blades at sea level

Main rotor total power at sea level	**1.800496029 kW**
Main rotor rotational velocity	97.27148318 rad/s
Distance between main and tail rotor shafts	1.603288976 m
Thrust required at tail rotor at hover	11.54502356 N
Thrust coefficient of tail rotor at hover	0.005453044
Tip-loss factor	0.947783892
Density	1.225 kg/m^3
Radius of main rotor	1.202819122 m
Radius of tail rotor	0.200469854 m
Disk area of tail rotor	0.126254835 m^2
Aspect ratio of tail rotor	6
Chord of tail rotor	0.033411642 m
Solidity of tail rotor	0.106060606
Profile drag coefficient of main rotor	0.0096445
Profile drag coefficient of tail rotor	0.011
Tip speed of tail rotor	117 m/s
Number of blades of tail rotor	2
Induced power at hover for tail rotor	74.41763431 W
Profile power at hover for tail rotor	36.12422146 W
Total power at hover OGE for tail rotor	**110.5418558 W**

Total power required of tail rotor at hover OGE at sea level is 0.110 kW.Another table needs to be constructed for 2000 meters altitude. It is already founded that total hovering power for 2000 meters is 1.89 kW for untwisted blades.

For 2000 meters altitude the table will be reconstructed and it can be seen at below. But these values are calculated for untwisted main rotor blades.

Table 4.2: Tail rotor power for untwisted main rotor blades at 2000 meters

Main rotor total power at 2000 meters	**1.890655471 kW**
Main rotor rotational velocity	97.27148318 rad/s
Distance between main and tail rotor shafts	1.603288976 m
Thrust required at tail rotor at hover	12.12313807 N
Thrust coefficient of tail rotor at hover	0.00695118
Tip-loss factor	0.941045866
Density	1.0087 kg/m^3
Radius of main rotor	1.202819122 m
Radius of tail rotor	0.200469854 m
Disk area of tail rotor	0.126305653 m^2
Aspect ratio of tail rotor	6
Chord of tail rotor	0.033411642 m
Solidity of tail rotor	0.106060606
Profile drag coefficient of main rotor	0.0092702
Profile drag coefficient of tail rotor	0.011
Tip speed of tail rotor	117 m/s
Number of blades of tail rotor	2
Induced power at hover for tail rotor	88.85960612 W
Profile power at hover for tail rotor	29.7576887 W
Total power at hover OGE for tail rotor	**118.6172948 W**

For twisted main rotor blades, the power required for main rotor is obtained as P=1.80 kW for sea level and P=1.89 kW for 2000 meters altitude.

For linearly twisted main rotor blades with a twist angle 8.8 degrees, the tables need to be reconstructed and the power requirements for the tail rotor need to be recalculated. Results can be seen from the tables below.

As twist of main rotor has been taken into account, the power needs of main rotor decreased. As it is expected, the tail rotor power needed for encountering the torque produced by main rotor was also decreased.

These power values considering twisted main rotor blades will be used during the design.

Table 4.3: Tail rotor power for twisted main rotor blades at sea level

Main rotor total power at sea level with twisted blades	**1.432984 kW**
Main rotor rotational velocity	97.27148318 rad/s
Distance between main and tail rotor shafts	1.603288976 m
Thrust required at tail rotor at hover	9.188486825 N
Thrust coefficient of tail rotor at hover	0.004338238
Tip-loss factor	0.953426197
Density	1.225 kg/m^3
Radius of main rotor	1.202819122 m
Radius of tail rotor	0.200469854 m
Disk area of tail rotor	0.126305653 m^2
Aspect ratio of tail rotor	6
Chord of tail rotor	0.033411642 m
Solidity of tail rotor	0.106060606
Profile drag coefficient of main rotor	0.0096445
Profile drag coefficient of tail rotor	0.011
Tip speed of tail rotor	117 m/s
Number of blades of tail rotor	2
Induced power at hover for tail rotor	52.51513409 W
Profile power at hover for tail rotor	36.13876144 W
Total power at hover OGE for tail rotor	88.65389552 W

Table 4.4: Tail rotor power for twisted main rotor blades at 2000 meters

Main rotor total power at 2000 meters with twisted blades	**1.565378 kW**
Main rotor rotational velocity	97.27148318 rad/s
Distance between main and tail rotor shafts	1.603288976 m
Thrust required at tail rotor at hover	10.03741502 N
Thrust coefficient of tail rotor at hover	0.005755265
Tip-loss factor	0.946356429
Density	1.0087 kg/m^3
Radius of main rotor	1.202819122 m
Radius of tail rotor	0.200469854 m
Disk area of tail rotor	0.126305653 m^2
Aspect ratio of tail rotor	6
Chord of tail rotor	0.033411642 m
Solidity of tail rotor	0.106060606
Profile drag coefficient of main rotor	0.0092702
Profile drag coefficient of tail rotor	0.011
Tip speed of tail rotor	117 m/s
Number of blades of tail rotor	2
Induced power at hover for tail rotor	66.56879311 W
Profile power at hover for tail rotor	29.7576887 W
Total power at hover OGE for tail rotor	96.32648181 W

4.2.2. Power required at forward flight

Power required at forward flight for both main and tail rotors will be smaller than the hovering rotors. This makes the hovering performance the key point at design process. Tail rotor power will be calculated for different forward flight speeds and of course because of the changing profile drag coefficients, the tail rotor's performance is affected.

$$T_{tr} = \frac{P_{total,mr}}{\Omega_{mr}L} = \frac{(P_{i,f}+P_{0,f}+P_p)_{mr}}{\Omega_{mr}L} \qquad \textbf{(4.13)}$$

where at forward flight, total power which is the sum of profile,parasitic and induced power was calculated before for different airspeeds. These data will be used for obtaining the required thrust for tail rotor. Then; thrust coefficient,tip-loss factor and induced power are obtained for different advance ratios.The data at sea level can be seen the table below.

Table 4.5: Tail rotor power at forward flight at sea level

Forward speed	Total power of main rotor at forward flight	Thrust required at tail rotor to encounter torque	Thrust coefficient	Tip-loss factor	Profile power	Induced power	Tail rotor power at sea level
10	1.670202532	10.70956407	0.005058433	0.949708686	36.21992315	66.35294879	102.5728719
20	1.408851031	9.033742971	0.004266894	0.953810746	36.50702824	51.18377716	87.6908054
30	1.166973806	7.482793556	0.003534337	0.957962298	36.98553671	38.41847983	75.40401654
40	1.008196164	6.464689882	0.003053457	0.960926624	37.65544857	30.75560439	68.41105296
50	0.929873932	5.962477161	0.002816248	0.962475022	38.51676383	27.19841358	65.71517741
60	0.888689749	5.698398621	0.002691516	0.963315424	39.56948247	25.38947223	64.9589547
70	0.890627954	5.710826652	0.002697386	0.963275442	40.8136045	25.47363526	66.28723976
80	0.928657755	5.954678872	0.002812565	0.962499569	42.24912993	27.14437977	69.39350969
90	1.000037767	6.412377139	0.003028749	0.961085038	43.87605874	30.37803777	74.25409651
100	1.104171296	7.080095383	0.003344131	0.959109103	45.69439094	35.31698314	81.01137408
110	1.241631489	7.961508699	0.003760448	0.956638452	47.70412653	42.22193915	89.92606568
120	1.413673901	9.064667863	0.004281501	0.953731754	49.90526551	51.45108638	101.3563519
130	1.621974019	10.40031634	0.004912366	0.950440107	52.29780788	63.45093368	115.7487416

As it can be seen from the tables, at very low or very high airpeeds the power required to transmitted to tail rotor is exceeds the hovering value. Profile drag coefficient is assumed to be constant for tail rotor calculations and the expression which gives the induced part of the total power at forward flight is constructed to obtain a thrust to encounter the main rotor torque.

For 2000 meters altitude,the forward flight performance of tail rotor is;

Table 4.6: Tail rotor power at forward flight at 2000 meters

Forward speed	Total power of main rotor at forward flight	Thrust required at tail rotor to encounter torque	Thrust coefficient	Tip-loss factor	Profile power	Induced power	Tail rotor power at 2000m
10	**1.764875549**	11.31662023	0.006488738	0.943040639	29.83652393	79.97187938	109.8084033
20	**1.510775654**	9.687297411	0.005554514	0.947300311	30.07302962	63.05345201	93.12648163
30	**1.253006598**	8.03444743	0.004606801	0.952006244	30.46720576	47.39001232	77.85721807
40	**1.066359945**	6.837643899	0.003920577	0.955724857	31.01905235	37.06119644	68.08024879
50	**0.966309663**	6.196107989	0.003552732	0.957853043	31.72856941	31.89864109	63.6272105
60	**0.896884206**	5.750942586	0.003297483	0.959395305	32.59575692	28.47761124	61.07336816
70	**0.872976863**	5.597645476	0.003209585	0.95994014	33.62061489	27.33106341	60.9516783
80	**0.885024627**	5.674897367	0.00325388	0.959664658	34.80314331	27.90680476	62.70994807
90	**0.928673687**	5.954781029	0.00341436	0.958681967	36.14334219	30.02733116	66.17067335
100	**1.002174823**	6.426080233	0.003684594	0.957078014	37.64121152	33.71820709	71.35941861
110	**1.105199510**	7.086688433	0.004063374	0.954925761	39.29675132	39.13704195	78.43379326
120	**1.238247955**	7.939813018	0.00455254	0.95228973	41.10996157	46.54134284	87.6513044
130	**1.402330357**	8.991931532	0.005155805	0.949226952	43.08084227	56.27325334	99.35409561

5. TOTAL POWER CALCULATIONS AND ENGINE SELECTION

5.1. Total power calculations

At this section, total power required for hover and forward flight will be determined, required rotor shaft power will be determined at sea level and operational altitude and at the end engine shaft power will be determined.

5.1.1. Total power required for hover and forward flight

Total power required for hover is the sum of total main rotor power and total tail rotor power at hover for specified design. The design and all power values will change at the selection of motor at Chapter 5.2. So, all of the calculations at Chapter 3, Chapter 4 and Chapter 5.1 will be recalculated.

Total power for hover can be stated as;

$$P_{total,hover} = P_{total,main\ rotor} + P_{total,tail\ rotor} \quad \textbf{(5.1)}$$

At sea level and the operational altitude 2000 meters the total power required for hover for the design with zero-twisted main rotor blades and with twisted main rotor blades can be seen at the table below.

Table 5.1: Total power at hover

	Main rotor power for hover OGE	**Tail rotor power for hover OGE**	**Total power for hover OGE**
Zero-twisted main rotor blades at sea level	1.804126 kW	0.110541 kW	1.914667 kW
Zero-twisted main rotor blades at 2000 m	1.890655 kW	0.118617 kW	2.009272 kW
Twisted main rotor blades at sea level	1.432984 kW	0.087919 kW	1.520903 kW
Twisted main rotor blades at 2000 m	1.565378 kW	0.096326 kW	1.661704 kW

Total power for forward flight is the sum of main rotor and tail rotor powers at specified speed values.It can be stated as;

$$P_{total,forward@V_{forw}} = P_{total,main\ rotor@V_{forw}} + P_{total,tail\ rotor@V_{forw}} \quad \textbf{(5.2)}$$

All these values can be seen the tables which are constructed for both sea level and 2000 meters are given below.

Table 5.2: Total power at forward flight at sea level

Forward speed	Main rotor power	Tail rotor power	Total power at sea level (kW)
10	1.670202532	0.102572872	1.772775404
20	1.408851031	0.087690805	1.496541836
30	1.166973806	0.075404017	1.242377822
40	1.008196164	0.068411053	1.076607216
50	0.929873932	0.065715177	0.995589109
60	0.888689749	0.064958955	0.953648704
70	0.890627954	0.06628724	0.956915193
80	0.928657755	0.06939351	0.998051265
90	1.000037767	0.074254097	1.074291863
100	1.104171296	0.081011374	1.185182670
110	1.241631489	0.089926066	1.331557555
120	1.413673901	0.101356352	1.515030253
130	1.621974019	0.115748742	1.737722761

Table 5.3: Total power at forward flight at 2000 meters

Forward speed	Main rotor power	Tail rotor power	Total power at 2000 m (kW)
10	1.764875549	0.109808403	1.874683952
20	1.510775654	0.093126482	1.603902136
30	1.253006598	0.077857218	1.330863816
40	1.066359945	0.068080249	1.134440194
50	0.966309663	0.06362721	1.029936874
60	0.896884206	0.061073368	0.957957574
70	0.872976863	0.060951678	0.933928541
80	0.885024627	0.062709948	0.947734575
90	0.928673687	0.066170673	0.994844360
100	1.002174823	0.071359419	1.073534241
110	1.10519951	0.078433793	1.183633303
120	1.238247955	0.087651304	1.325899259
130	1.402330357	0.099354096	1.501684453

5.1.2. Required shaft power

The power required will be selected as the maximum power at forward flight at 2000 meters altitude namely 1.874683952 kW. Power values for hover are higher than that value but the zero twisted condition has not taken into acccount because the blades have twist.

Table 5.4: Comparison of power values

	Maximum Power at forward flight	Power at hover with zero twist	Power at hover with twist
At sea level	1.772775404 kW	1.914667 kW	1.520903 kW
At 2000 meters	1.874683952 kW	2.009272 kW	1.661704 kW

Power needs of helicopter can be seen from the table above. The biggest value of power at both sea level and operational altitude forward flight power tables is the value given above.From now on, zero twisted main rotor configuration is not considered. Therefore, required rotor shaft power can be stated as the maximum power at the table namely 1.874683952 kW

5.1.3. Required total engine shaft power

Required total engine shaft power can be calculated by considering fifteen percent of transmission and accessories losses [3]. So,the required total engine shaft power can be stated as;

$$P_{engine} = (1.15)P_{rotors} = 2.156\ kW \quad \textbf{(5.3)}$$

5.2. Engine selection

Total required engine power was approximated. At here, an engine, which is capable of encountering the power requirements of the helicopter, will be selected here. At first engine's type will be selected. In general, two stroke engines have an excellent power-to-weight ratio and are relatively inexpensive. They are unfortunately also not very fuel efficient and exhibit high vibration and noise levels. Four cylinder engines are the most fuel efficient internal combustion engines. They however have lower vibration and noise levels than the two stroke engines, but have a higher vibration level than a Wankel engine. They also have a lower power-to-weight ratio than a two stroke engine of comparable power output.

A figure shows the displacement required at the 4 stroke engine for the required power.

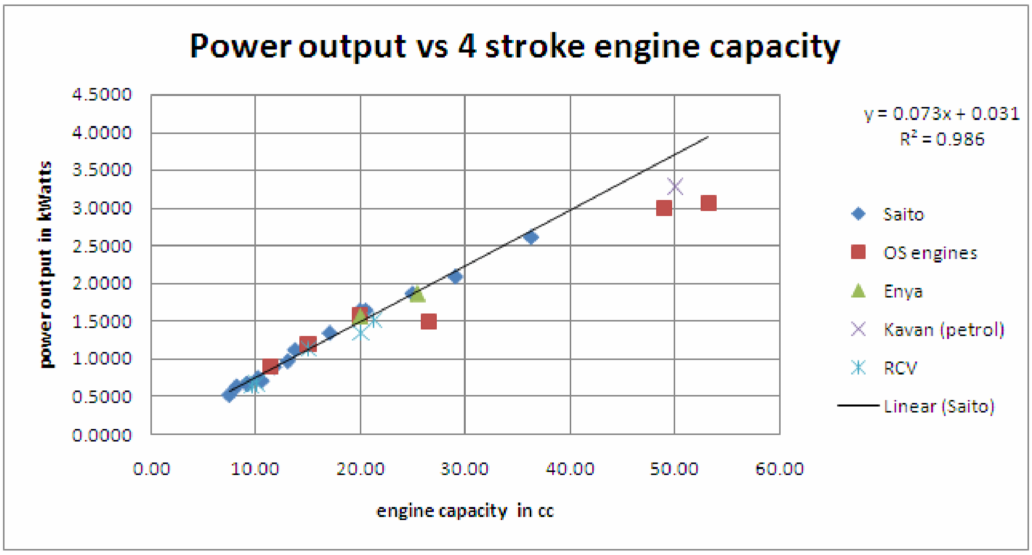

Figure 5.1 : Power output with respect to displacement for 4 stroke engines [13]

By using the formula given at the figure above, the displacement required at a 4-stroke engine can be stated as; [13]

$$P = (0.073)Capacity + 0.031 \cong 2.2\ kW \quad \textbf{(5.4)}$$

A 30cc engine will works for the design. Capacities for the given engines at the table are calculated by using the formulas which are given below; [13]

$$\frac{J}{ccfam} = \frac{Power(Watt)\ x\ 60}{Capacity(cc)\ x\ RPM}\ for\ 2 - stroke \quad \textbf{(5.5)}$$

$$\frac{J}{ccfam} = \frac{Power\,(Watt)\ x\ 60\ x\ 2}{Capacity(cc)\ x\ RPM}\ for\ 4 - stroke \quad \textbf{(5.6)}$$

J/ccfam is an efficiency unit.It means energy in joules per cc of fuel air mixture.

At the table given below specifications of some engines are shown. These values are obtained from different sources. [13]

A wankel engine is selected. Because it produces less noise compared with 2 stroke and 4 stroke engines. The selected engine is NR 20 EH 36 cc, four stroke, petrol (gasoline) engine. Some detailed specifications for the selected engine are given at the Table 5.6. The required engine shaft power 2.195 kW.

Table 5.5: Engine database

Engine	Manufacturer	Type	Mass (kg)	Power output (kW)	Efficiency (J/ccfam)	Capacity (cc)
BGX-1 3500 RN	O.S. Engines	2-stroke methanol powered	1.34	3.06 kW at 10,000 rpm	0.525	35
116 Gx 2	Evolution Engines	2-stroke gasoline powered	1.493	3.015 kW at 8,500 rpm	0.615	34
G450PU	Zenoah(Husqvarna)	2-stroke pre-mixed fuel powered	2.148	2.9 kW at 8000 rpm	0.483	45
FT-300	O.S. Engines	2-cylinder 4-stroke glow fuel powered	1.828	3.0 kW at 7,000 rpm	1.05	50
FF-300 radial	O.S. Engines	5 cylinder 4-stroke glow fuel powered	2.673	3.0 kW at 8,000 rpm	0.905	50
FF-320	O.S. Engines	4 cylinder 4-stroke glow fuel powered	2.19	3.06 kWat 8,000 rpm	0.846	55
FA-300T	Saito Engines	2 cylinder 4-stroke glow fuel powered	1.75	3.504 kW at 8000 rpm	1.168	45
FA-300TD	Saito Engines	2 cylinder 4-stroke glow fuel powered	1.85	3.579 kW at 8000 rpm	1.136	45
Kavan	Kavan	4 stroke petrol powered	2.6	3.28 kW at 9200 rpm	0.856	50
FG-36	Saito Engines	36 cc, four stroke, petrol (gasoline) engine	1.76	2.6 kW at 8300 rpm	1.164	36
Sonic 30	Cubewano	Wankel	1.5	3.72 kW at 8000 rpm		
Wren 44 Heli engine	Wren Turbines	Gas turbine engine	5.2	5.1 kW at 15000 rpm		
NRG 20EH	Nitto	Wankel 20cc Gasoline powered	1.8	1.34 kW		
NR 20EH	Nitto	Wankel 20cc Alcohol powered	1.493	2.24 kW		
Wren heli engine	Wren Turbines	Gas turbine engine	2.077	6 kW at 13500 rpm		
Raket 120	Raket		6.8	10 kW at 9000 rpm		
37SZ-H Ringed heli engine	O.S. Engines		0.293	1.03 kW at 18000 rpm		6.07
50SX-H Ringed Hyper Heli Engine	O.S. Engines		0.406	1.42 kW at 17000 rpm		8.17
55HZ-H Hyper Ringed Heli engine	O.S. Engines		0.408	1.57 kW at 17000 rpm		8.93
70SZ-H Ring 3D Heli Engine	O.S. Engines		0.553	1.86 kW at 16000 rpm		11.5
91HZ Ringed Heli engine	O.S. Engines		0.618	2.74 kW at 15000 rpm		14.95
Speed Tuned 91HZ-R 3C Heli engine	O.S. Engines		0.603	2.98 kW at 15500 rpm		14.9

Figure 5.2 : Nitto NR 20 EH Wankel engine

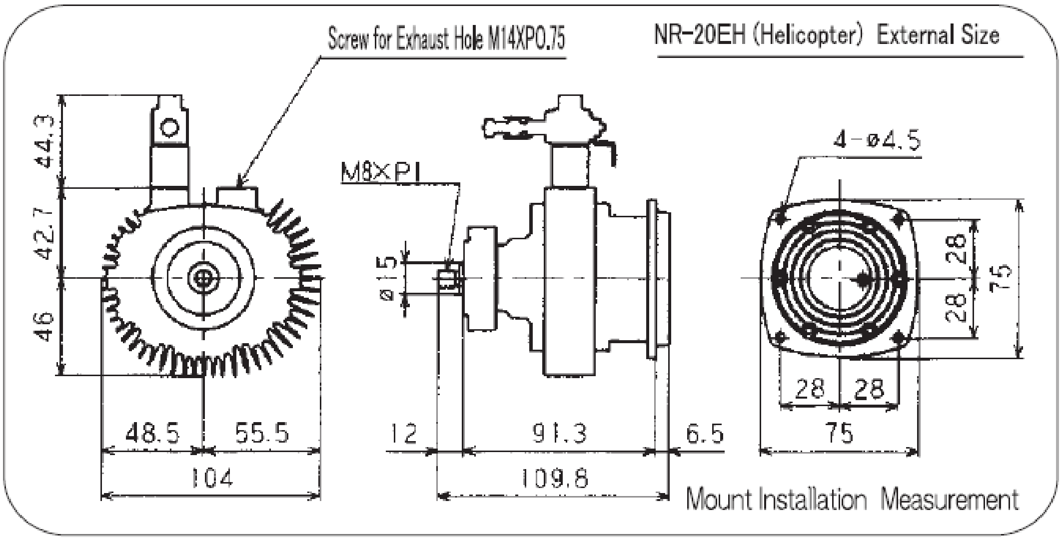

Figure 5.3 : Draftings of selected engine [14]

Table 5.6: Specifications of selected engine [14]

Weight	1.493 kg
Displacement	20cc
Fuel type	Glow(Alcohol)
Fuel consumption	38cc/min
Power	2.24 kW at 9500 RPM
RPM Range	2000-13000
Maximum operational altitude	4000 m

6. DESIGN FOR SELECTED ENGINE

6.1. Main rotor design

For the selected engine a new gross weight iteration has to be done because estimated fuel weight and engine weight is decreased. The engine's fuel consumption is 38cc/min. Density of alcohol is approximately 790 kg/m^3[15]

The new fuel weight for the defined mission time equals to one hour is;

$$W_{fuel} = 0.79 \; x \; 2.28 = 1.8 \; kg \quad \textbf{(6.1)}$$

Engine's weight is 1.493 kg.

So; gross weight iteration can be defined as;

$$W_e = GW - W_{payload} - W_{fuel} \quad \textbf{(6.2)}$$

$$W_{blades} = 0.0373 W_e R^{0.4} \sigma^{0.33} \quad \textbf{(6.3)}$$

$$W_{hub} = 8.169 x 10^{-3} W_e R^{0.42} \quad \textbf{(6.4)}$$

$$W_{main\ rotor} = W_{blades} + W_{hub} \quad \textbf{(6.5)}$$

$$W_{engine} = 1.493 \; kg \quad \textbf{(6.6)}$$

$$Engine\ Power\ Output = 2.24 \; kW \quad \textbf{(6.7)}$$

$$Shaft\ Power = 0.85 \; x \; Engine\ Power\ Output \quad \textbf{(6.8)}$$

At this step of design,it is assumed that transmission losses are 15%, which was assumed that 10% at the premature steps of the design. This gives a more reliable design.

$$W_{propulsion} = 0.8947 \; x \; Shaft\ Power \quad \textbf{(6.9)}$$

$$W_{fuselage} = 0.21W_e \qquad \textbf{(6.10)}$$

$$W_{controls} = 0.06W_e \qquad \textbf{(6.11)}$$

$$W_{electrical} = 0.06W_e \qquad \textbf{(6.12)}$$

$$W_{fixed\ equipment} = 0.28W_e \qquad \textbf{(6.13)}$$

$$W_e' = W_{main\,rotor} + W_{propulsion} + W_{fuselage} + W_{controls} + W_{electrical} + W_{fixed\,equipment} \qquad \textbf{(6.14)}$$

$$GW' = W_e' + W_{fuel} + W_{payload} \qquad \textbf{(6.15)}$$

Using the formulas given above a new gross weight value is obtained and by using the formulas given below a new design is completed.By using the values of this new design, another gross weight value is obtained. Aimed error rate between gross weight is 10%.

$$DL \cong 16{,}932GW^{0.3694} \qquad \textbf{(6.16)}$$

$$R = \sqrt{\frac{T}{\pi.DL}} \qquad \textbf{(6.17)}$$

$$A = \pi R^2 \qquad \textbf{(6.18)}$$

$$C_T = \frac{T}{\rho A V_{tip}{}^2} \qquad \textbf{(6.19)}$$

Advance ratio and blade loading coefficient are;

$$\mu = \frac{V_{f,max}}{V_{tip}} \qquad \textbf{(6.20)}$$

$$BL = -0.1515\mu^2 - 0.0363\mu + 0.1308 \qquad \textbf{(6.21)}$$

$$\sigma = \frac{C_T}{BL} \qquad \textbf{(6.22)}$$

During the design iteration, it has been taken into account that the aspect ratio should be between the interval 15 and 20. When it is more than 20, tip speed is decreased.

$$AR = \frac{R}{c} = \frac{N_b}{\pi\sigma} \qquad \textbf{(6.23)}$$

For new aspect ratio chord can be stated as;

$$c = \frac{\pi R \sigma}{N_b} = \frac{R}{AR} \quad \textbf{(6.24)}$$

By using blade loading, mean lift coefficient can be expressed as;

$$\overline{C_L} = 6\frac{C_T}{\sigma} = 6.BL \quad \textbf{(6.25)}$$

The input of for first iteration of gross weight can be seen the table below.

Table 6.1: Geometric specifications of design before last iteration

Gross weight	**26.22**
Disk loading	56.59149598
Radius	1.202819122
Tip speed	117
Density	1.2250
Area	4.545174068
Maximum forward speed	130
Number of blades	2
Thrust coeff.	0.003374764
Max. advance ratio	0.308641975
Blade loading	0.105164426
Solidity	0.032090354
Aspect ratio	19.83835315
Chord	0.060630997

After the first iteration new gross weight value can be seen the table below. This procedure will go on until error rate will be under 10% percentage.

Table 6.2: Weight of design

Payload weight	**9.54**
Fuel weight	1.8
Empty weight	14.88
Engine weight	1.493
Engine output	2.24
Shaft power	1.904
Propulsion weight	1.703509
Fuselage weight	3.1248
Controls	0.8928
Electrical	0.8928
Fixed eq	4.1664
Blades	0.20599
Hub	0.131358
New gross weight	22.45766

During the gross weight iteration these error percentages and gross weight values are obtained;

Table 6.3: New gross weight iteration

Iteration number	Previous gross weight	New gross weight	Error %	UL/GW
1	26.62	23.70828309	10.93808	0.478314
2	23.70828309	21.85938739	7.798522	0.51877
3	21.85938739	20.68668018	5.364776	0.548179

The iteration process is terminated because the useful load per gross weight ratio can not exceed 0.55.A new gross weight value should be selected.

New gross weight is 20.68668018 kg. For this gross weight geometric specification of the main rotor can be seen at the table below.

Table 6.4: Last geometric specifications of design

Gross weight	**20.68668018**
Disk loading	51.84709662
Radius	1.116201873
Tip speed	113
Density	1.2250
Area	3.914131086
Maximum forward speed	130
Number of blades	2
Thrust coeff.	0.003314603
Max. advance ratio	0.319567355
Blade loading	0.103728026
Solidity	0.031954745
Aspect ratio	19.92254287
Chord	0.056027078

The aspect ratio should be between 15 and 20. Therefore tip speed was decreased.

$$\overline{C_L} = 6\frac{C_T}{\sigma} = 6.BL = 0.622368 \quad \textbf{(6.26)}$$

For this value of lift coefficient, mean angle of attack can be approximated as 4.3 degrees.

$$\alpha = 4.3\,deg = 0.07504\,rad \quad \textbf{(6.27)}$$

Lift curve slope can be calculated as;

$$C_L = C_{l,\alpha}\alpha \tag{6.28}$$

$$C_{l,\alpha} = 8.2827\ /rad \tag{6.29}$$

Profile drag coefficient can be calculated at $\alpha = 0$

To obtain profile drag coefficient values, the assumptions at Chapter 3 should be remade. At this chapter not only the average values will be shown; all tables used in obtaining the average profile drag coefficients will be given step by step.

At hover the profile drag coefficients at sea level and 2000 meters altitude at the blade stations can be seen at the tables given below.

Table 6.5: Profile drag coefficients at blade stations at hover at sea level

r	Speed at station	Mach number	Reynolds number	Profile Drag coefficients at sea level
0.2	22.6	0.066372981	86371.89378	0.015544927
0.3	33.9	0.099559471	129557.8407	0.012837298
0.4	45.2	0.132745962	172743.7876	0.01120734
0.4	45.2	0.132745962	172743.7876	0.01120734
0.5	56.5	0.165932452	215929.7344	0.010086976
0.6	67.8	0.199118943	259115.6813	0.009255235
0.7	79.1	0.232305433	302301.6282	0.008605747
0.8	90.4	0.265491924	345487.5751	0.008080093
0.9	101.7	0.298678414	388673.522	0.00764315
1	113	0.331864905	431859.4689	0.007272351

The average value of the profile drag coefficients at sea level is obtained from the table as 0.010174046.

Table 6.6: Profile drag coefficients at blade stations at hover at 2000 meters

r	Speed at station	Mach number	Reynolds number	Profile Drag coefficients at 2000 meters
0.2	22.6	0.06684	93925.67	0.014942
0.3	33.9	0.100259	140888.5	0.012339
0.4	45.2	0.133679	187851.3	0.010772
0.4	45.2	0.133679	187851.3	0.010772
0.5	56.5	0.167099	234814.2	0.009696
0.6	67.8	0.200519	281777	0.008896
0.7	79.1	0.233939	328739.8	0.008272
0.8	90.4	0.267358	375702.7	0.007767
0.9	101.7	0.300778	422665.5	0.007347
1	113	0.334198	469628.4	0.00699

The average value of the profile drag coefficients at 2000 meters altitude is obtained from the table as 0.009779.The variation of drag coefficient values with respect to location on the blade can be seen at the figure given below.

Table 6.7: Mean profile drag coefficients

	Mean profile drag coefficients at hover	
Sea level	0.010174046	
2000 meters	0.009779289	

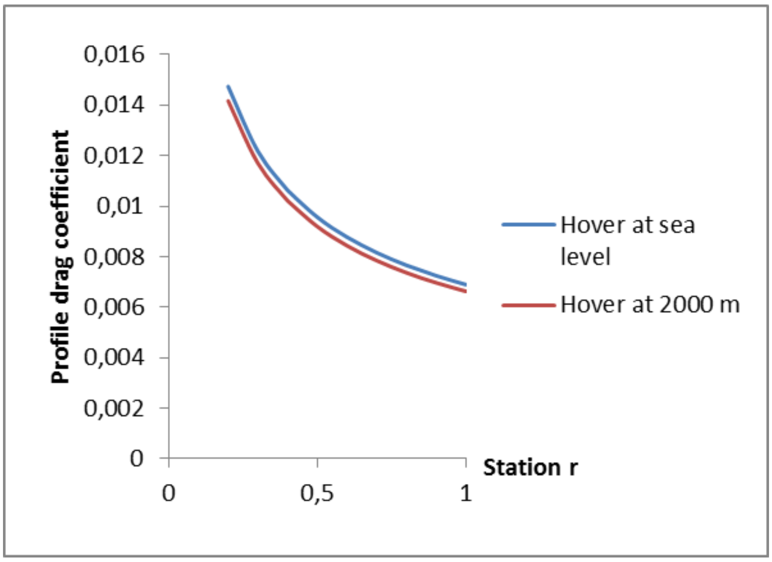

Figure 6.1: Profile drag coefficient change with respect to blade station

Power required to hover OGE can be calculated as;

$$B = 1 - \frac{c}{2R} = 1 - \frac{0.056027}{2x\ 1.116201} = 0.9749028 \quad \textbf{(6.30)}$$

Root cut out ratio is 0.2 and induced power factor is 1.15

$$A_e = \pi R^2 (B^2 - {r_0}^2) = 3.563563\ m^2 \quad \textbf{(6.31)}$$

$$P = P_i + P_0 = \frac{\kappa W^{\frac{3}{2}}}{\sqrt{2\rho A_e}} + \rho A_e V_{tip}^3 (\frac{\sigma C_{d0}}{8}) \quad \textbf{(6.32)}$$

$P_i = 1.1251547\ kW$ at sea level **(6.33)**

$P_0 = 0.255973\ kW$ at sea level **(6.34)**

So the total power required to hover out-of-ground effect at sea level is;

$$P = P_i + P_0 = 1.381128\ kW \tag{6.35}$$

For fixed geometric design parameters,the total power required to hover out-of ground effect at 2000 meters is;

$$P = P_i + P_0 = 1.239936 + 0.202597 = 1.442533\ kW \tag{6.36}$$

Power required to hover IGE can be found by;

Assume that ground effect at 2 meters altitude will be calculated;

$$P = P_0 + k_G P_i \tag{6.37}$$

$$k_G = \frac{1}{A+B(\frac{2R}{z})^2} \tag{6.38}$$

where A= 0.9926 and B= 0.0379

$$k_G = \frac{1}{0.9926+0.0379(\frac{2(1.116201)}{2})^2} = 0.961705107 \tag{6.39}$$

$$P = P_0 + k_G P_i = 0.255973 + (0.961705107)(1.1251547) = 1.33804\ kW \tag{6.40}$$

Twist rate can be calculated by BEMT. Input of BEMT can be seen the table below. Blade element theory is used for calculating the twist by comparing the thrust coefficient value obtained from classical momentum theory. By considering the twist value obtained from blade element momentum theory, inflow ratios, thrust coefficients, power coefficients and power values are obtained at each station with blade element momentum theory. All of these values are summed and total power required value is obtained for blades with defined twist rates. Twist is used at blades to decrease the power requirements of design.

Because of necessary twist, modified momentum theory can not be used. Blade element theory and blade element momentum theory was used for obtaining twist and the new power requirements.

Table 6.8: Input of BEMT

Max. Take-off Weight (W)	**20.68668018**
Gravitational Constant (g)	9.81
Air Density (ρ_SL)	1.225
Air Density (ρ_2000)	1.0087
Effective Disc Area (Ae)	3.563563793
Disc Area (A)	3.914131088
Pi (π)	3.141592654
Rotor Diameter (D)	2.232403746
Rotor Radius (R)	1.1162018730
Number of Blades (Nb)	2
Chord (c)	0.056027078
Solidity (σ)	0.031954744
Cd0	0.010174046
Kapha (κ)	1.15
Blade Tip Velocity (Vtip)	113
Rotor RPM	966.7341905
Rotor Root Cutout	0.2
Radial Velocity (Ω)	101.2361677
Clα	8.2827
Tip loss factor	0.974902802
Power output of engine	2.24 kW

The pre-work with momentum theory has already be done for hover,but it can be seen the table below. At the table, ideal power means the power without tip losses and without root cut-out losses. It is the result of momentum theory, but the actual power is the result of modified momentum theory. The power required is increased by considering the losses with modified theory.

Table 6.9: Modified momentum theory output of BEMT calculator

Ideal Power (P)	**0.9783954**
Thrust (T)	202.9363326
Induced Velocity (vi)	4.8211938
Actual Power (P)	1.3811282
Power Coefficient (Cp)	0.0002193
Induced Power Coefficient (Cpi)	0.0001786
Profile Power Coefficient (Cp0)	0.0000406
Thrust Coefficient (Ct)	0.0036407

The momentum theory has been used to maintain a thrust coefficient value.The value obtained from modified momentum theory has been found as 0.0036407. At the previous steps it has been found differently.It is the result of modified momentum theory; it uses effective rotor disk area in calculations.

Table 6.10: BET output of calculator up to seven degrees

Teta(rad)	0	0.017453	0.034907	0.05236	0.069813	0.087266	0.10472	0.122173
Teta(deg)	0	1	2	3	4	5	6	7
1	-0.00048	0.00029	0.00106	0.00183	0.0026	0.00337	0.004139	0.004909
2		-4.7E-05	-2.3E-05	0.000257	0.000633	0.001064	0.001532	0.002027
3		0.00044	0.001312	0.001541	0.001873	0.002284	0.002741	0.003229
4		-0.00024	-0.0002	0.000426	0.001003	0.001556	0.002107	0.002662
5			0.000864	0.001319	0.00156	0.001957	0.002417	0.002913
6			0.000129	0.000567	0.001184	0.001727	0.00226	0.002799
7			0.000995	0.001167	0.001428	0.001855	0.002338	0.00285
8			2.63E-05	0.00067	0.001266	0.001782	0.002299	0.002827
9			0.001294	0.001067	0.001372	0.001823	0.002318	0.002838
10			-0.00019	0.000742	0.001302	0.0018	0.002309	0.002833
11			0.000884	0.001003	0.001348	0.001813	0.002313	0.002835
12			0.000113	0.00079	0.001318	0.001806	0.002311	0.002834
13			0.001031	0.000961	0.001338	0.00181	0.002312	0.002834
14			-8.6E-07	0.000822	0.001324	0.001808	0.002312	0.002834
15			0.001495	0.000934	0.001333	0.001809	0.002312	0.002834
16			-0.00032	0.000843	0.001327	0.001808	0.002312	0.002834
17			0.000687	0.000916	0.001331	0.001809	0.002312	0.002834
18			0.000282	0.000857	0.001329	0.001808	0.002312	0.002834
19			0.000734	0.000905	0.00133	0.001809	0.002312	0.002834
20			0.00024	0.000866	0.001329	0.001809	0.002312	0.002834

Table 6.11: BET output of calculator between 8 and 15 degrees

Teta(rad)	0.139626	0.15708	0.174533	0.191986	0.20944	0.226893	0.244346	0.261799
Teta(deg)	8	9	10	11	12	13	14	15
1	0.005679	0.006449	0.007219	0.007989	0.008759	0.009529	0.010299	0.011069
2	0.002542	0.003075	0.003621	0.004179	0.004747	0.005324	0.005908	0.006499
3	0.003739	0.004268	0.004811	0.005366	0.005932	0.006507	0.00709	0.007679
4	0.003224	0.003794	0.00437	0.004953	0.005542	0.006137	0.006738	0.007343
5	0.003434	0.003973	0.004526	0.005091	0.005666	0.006249	0.006839	0.007436
6	0.003347	0.003904	0.00447	0.005044	0.005626	0.006215	0.00681	0.00741
7	0.003383	0.00393	0.00449	0.00506	0.005639	0.006225	0.006818	0.007417
8	0.003368	0.00392	0.004483	0.005055	0.005635	0.006222	0.006816	0.007415
9	0.003374	0.003924	0.004486	0.005057	0.005636	0.006223	0.006816	0.007416
10	0.003371	0.003923	0.004485	0.005056	0.005636	0.006223	0.006816	0.007416
11	0.003373	0.003923	0.004485	0.005056	0.005636	0.006223	0.006816	0.007416
12	0.003372	0.003923	0.004485	0.005056	0.005636	0.006223	0.006816	0.007416
13	0.003372	0.003923	0.004485	0.005056	0.005636	0.006223	0.006816	0.007416
14	0.003372	0.003923	0.004485	0.005056	0.005636	0.006223	0.006816	0.007416
15	0.003372	0.003923	0.004485	0.005056	0.005636	0.006223	0.006816	0.007416
16	0.003372	0.003923	0.004485	0.005056	0.005636	0.006223	0.006816	0.007416
17	0.003372	0.003923	0.004485	0.005056	0.005636	0.006223	0.006816	0.007416
18	0.003372	0.003923	0.004485	0.005056	0.005636	0.006223	0.006816	0.007416
19	0.003372	0.003923	0.004485	0.005056	0.005636	0.006223	0.006816	0.007416
20	0.003372	0.003923	0.004485	0.005056	0.005636	0.006223	0.006816	0.007416

The thrust coefficient value obtained from modified momentum theory has been found as 0.0036407. As it can be seen from the figure below, twist angle should be 8.3 degrees(actually by the figure below it is 8.32388 degrees,but it was approximated to 8.3 degrees) .

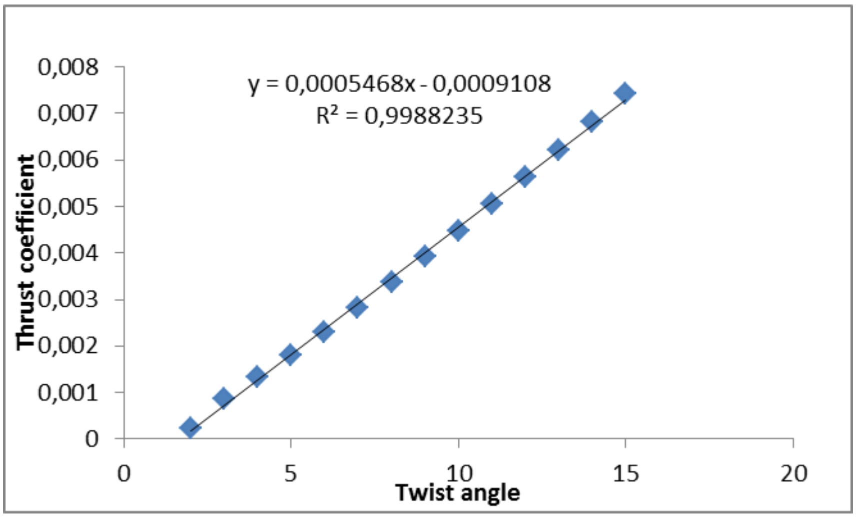

Figure 6.2 : Thrust coefficient twist angle change of last design

Power coefficient can be defined as the formula given below by the uniform inflow and constant drag coefficient assumptions

$$C_p = \lambda C_T + \frac{\sigma C_{d0}}{8} \quad \textbf{(6.41)}$$

In hover with uniform inflow the equation given above, returns into simple momentum theory because for this assumptions $\lambda = \sqrt{C_T/2}$ .So;

$$C_p = \frac{\kappa C_T^{3/2}}{\sqrt{2}} + \frac{\sigma C_{d0}}{8} \quad \textbf{(6.42)}$$

The second term is the extra power predicted by the BET that is required overcome profile drag of blades, which has been already introduced in momentum theory.

So,the total power can be written as;

$$P = \rho A_e V_{tip}^3 C_p \quad \textbf{(6.43)}$$

BET analysis will not be used at here to calculate total power. BEMT will be used by using 40 stations at the blade. Firstly, inflow ratios at the stations will be calculated and then, thrust and power coefficients will be found by using these inflow ratio values. The estimated value for twist angle from BET will be used with a linear twist distribution assumption.

Blade element momentum theory was used for power calculations for twisted blades.

The inflow ratio on each station can be found by;

$$\lambda(r_n) = \frac{\sigma C_{l\alpha}}{16} \left(\sqrt{1 + \frac{32}{\sigma C_{l\alpha}} \theta(r_n) r_n} - 1 \right) \quad \textbf{(6.44)}$$

where n=(1,N) element station number, r_n and $\theta(r_n)$ are the radius and pitch angle at the mid-span of each of the N stations. Here, N equals to 40. Thrust coefficients can be calculated and summed by rectangular approach.

$$\Delta C_{T_n} = \frac{\sigma C_{l\alpha}}{2} (\theta(r_n) r_n{}^2 - \lambda(r_n) r_n) \Delta r \quad \textbf{(6.45)}$$

$$C_T = \sum_{n=1}^{N} \Delta C_{T_n} \quad \textbf{(6.46)}$$

The 8.8 twist angle is used in calculations. The tables of this process can be seen from the table for sea level computations below.

Table 6.12: Input for BEMT for last design

Seçilen twist	**8.3**
Effective AoA	4.3
Delta r	0.0225

The thrust coefficients at each station is summed and the obtained result is 0.00201402625. The induced power coefficients is also summed up and profile power coefficient is obtained with the values of profile drag coefficient and solidity. These two power coefficient is summed and total power coefficient is obtained.This coefficient is used to obtain total power value at hover by BEMT. The total out of ground effect power at hover obtained by BEMT at sea level and 2000 meters altitude can be stated at Table 6.14

Table 6.13: Output of BEMT for last design

Section	Inflow ratio	Ct	Cpi
0.10000000	0.00000	0.00000000000	0.00000000000
0.11125000	0.01135	0.00000129092	0.00000002932
0.13375000	0.01301	0.00000203753	0.00000005302
0.15625000	0.01454	0.00000297301	0.00000008646
0.17875000	0.01596	0.00000409894	0.00000013086
0.20125000	0.01729	0.00000541465	0.00000018724
0.22375000	0.01853	0.00000691772	0.00000025643
0.24625000	0.01970	0.00000860442	0.00000033908
0.26875000	0.02081	0.00001046998	0.00000043567
0.29125000	0.02185	0.00001250876	0.00000054651
0.31375000	0.02283	0.00001471443	0.00000067179
0.33625000	0.02376	0.00001708007	0.00000081154
0.35875000	0.02464	0.00001959829	0.00000096570
0.38125000	0.02547	0.00002226129	0.00000113404
0.40375000	0.02626	0.00002506089	0.00000131628
0.42625000	0.02701	0.00002798863	0.00000151199
0.44875000	0.02772	0.00003103579	0.00000172068
0.47125000	0.02839	0.00003419342	0.00000194177
0.49375000	0.02903	0.00003745237	0.00000217457
0.51625000	0.02963	0.00004080332	0.00000241837
0.53875000	0.03020	0.00004423679	0.00000267233
0.56125000	0.03074	0.00004774319	0.00000293561
0.58375000	0.03125	0.00005131280	0.00000320726
0.60625000	0.03173	0.00005493581	0.00000348631
0.62875000	0.03218	0.00005860233	0.00000377174
0.65125000	0.03260	0.00006230239	0.00000406249
0.67375000	0.03300	0.00006602595	0.00000435744
0.69625000	0.03337	0.00006976295	0.00000465547
0.71875000	0.03371	0.00007350325	0.00000495541
0.74125000	0.03403	0.00007723670	0.00000525608
0.76375000	0.03432	0.00008095313	0.00000555627
0.78625000	0.03459	0.00008464235	0.00000585477
0.80875000	0.03483	0.00008829416	0.00000615035
0.83125000	0.03505	0.00009189836	0.00000644176
0.85375000	0.03524	0.00009544475	0.00000672777
0.87625000	0.03542	0.00009892316	0.00000700715
0.89875000	0.03557	0.00010232344	0.00000727867
0.92125000	0.03569	0.00010563546	0.00000754110
0.94375000	0.03580	0.00010884913	0.00000779323
0.96625000	0.03588	0.00011195440	0.00000803389
0.98875000	0.03594	0.00011494129	0.00000826189

Table 6.14: Power values of BEMT for last design

	Sea level	2000 meters
Total Power	1.0920618 kW	1.156943 kW

When the values for 2000 meters altitude were obtained the twist angle founded from BET was changed. The value was 9.8 degree approximately.(Actual value is 9.75164 degrees) It means at that altitude, the value of required thrust can only be obtained by equal or higher values of twist.This means for the designed twist, helicopter needs more power; more twist. This power difference is easily be encountered because the helicopter actually needs more power from that value at relatively high speeds.

Induced power for forward flight will be calculated for 0 to 40 km/h forward speeds by classical formula with numerical iteration for induced velocity and 40 to 130 km/h by Glauert's high-speed formula. Profile power is found with the help of Stepniewski constant and parasitic power will be calculated with equivalent flat plate approach.

$$\mu = \frac{V_{forw}}{V_{tip}} = \frac{V_{forw}}{113} > 0.1 \qquad \textbf{(6.47)}$$

This is the Glauert's high-speed formula's necessity.Forward speeds less than 11.5 m/s (40.68 km/h)must be considered with classical induced power formula with numerical approach to induced velocity.

The input table for the speeds higher than 40 km/h can be at Table 6.15

Total main rotor power for forward flight will be calculated for two different altitudes (sea level and 2000 meters) and for a speed range 0-130 km/h. Chord and radius values will be fixed.

The calculator is constructed in Excel. The Matlab codes are not uısed at this stage. Two different density of air values can be seen. The inputs and outputs can be seen at the table given below.

Table 6.15: Forward flight power calculator for last design

Gross weight	20.68668018
Forward velocity in km/h	**INPUT**
Forward velocity in m/s	----------------------
Tip speed	113
Advance ratio	----------------------
Density	1.225 or 1.0087
Main rotor radius	1.116201873
Number of blades	2
Main rotor chord	0.056027078
Profile drag coefficient	**INPUT**
Stepniewski constant	4.7
Equivalent flat plate area	0.02728868
Tip-loss effect	0.9749028
Root cut-out ratio	0.2
Effective rotor disk area	3.5635637
Induced power factor	1.15
Solidity	0.03195474
Profile power for hover	**OUTPUT**
Profile power for forward flight	**OUTPUT**
Induced power for hover	**OUTPUT**
Induced power in forward flight (50-130 km/h)	**OUTPUT**
Parasitic power for forward flight	**OUTPUT**
Total power for hover	**OUTPUT**
Total power for forward flight	**OUTPUT**

And the input of numerical approach to induced velocity to obtain induced power between forward speeds 0 and 40 km/h can be stated as;

Table 6.16: Induced power calculator for advance ratios smaller than 0.1

Gross weight	20.68668018
Density	1.225 or 1.0087
Radius	1.116201873
Chord	0.056027078
Root-cut out	0.2
Forward speed	Conversion of speed to m/s
Alfa(radian)	0.075049158
Tip loss	0.9749028
Effective disk area	3.5635637
Thrust	202.9363325
Vh	Depends on density
Forward speed km/h	**INPUT(0 to 40 km/h)**
Alfa(degrees)	4,3
Induced power factor	1,15
Obtained induced velocity	**Comes from numerical analysis**
Induced power at forward flight	**OUTPUT**

The induced velocities obtained during numerical approach can be seen at the table below for the changing forward speeds.

Table 6.17: Obtained induced velocities for last design

Forward Speed(km/h)	Induced velocity at sea level(m/s)	Induced velocity at 2000 meters(m/s)
10	4.354574572	4.875685391
20	3.442709119	3.988335955
30	2.60693535	3.09981241
40	2.031091019	2.443140059
50	1.647522832	1.991004187
60	1.38136929	1.672808113
70	1.187769475	1.439843648
80	1.041186444	1.26287055
90	0.926550835	1.124211775
100	0.834527345	1.012779361
110	0.759063495	0.921333734
120	0.696076769	0.844970744
130	0.642718715	0.780259238

Induced velocity in m/s change with respect to forward velocity in km/h can be seen at the figure below.

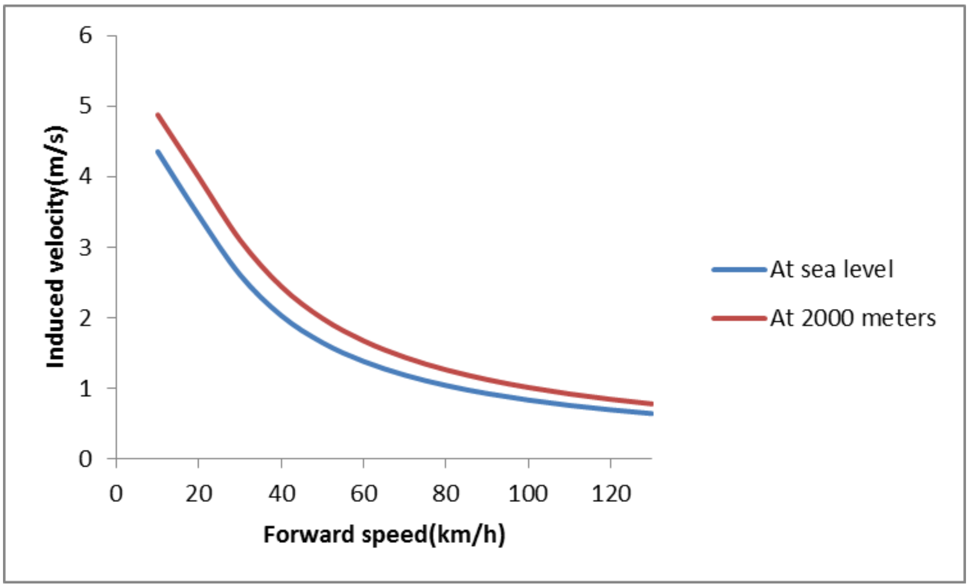

Figure 6.3: Induced velocity variation with respect to forward speed

By using these inputs, the power tables of different velocities at sea level and 2000 meters altitude can be stated as;

Table 6.18: Power values at forward flight at sea level

Forward Speed	Profile Power	Induced Power	Parasitic Power	Total Power
10	0.2567005	1.016256603	0.000358246	**1.273315348**
20	0.258881483	0.803448377	0.002865967	**1.065195827**
30	0.262516454	0.608398184	0.009672637	**0.880587275**
40	0.267605414	0.474009487	0.022927732	**0.764542633**
50	0.274148362	0.390570403	0.044780727	**0.709499492**
60	0.282145299	0.325475336	0.077381096	**0.685001731**
70	0.291596224	0.278978859	0.122878315	**0.693453398**
80	0.302501138	0.244106502	0.183421858	**0.730029498**
90	0.314860041	0.216983557	0.2611612	**0.793004798**
100	0.328672932	0.195285202	0.358245816	**0.882203949**
110	0.343939812	0.177532001	0.47682518	**0.998296993**
120	0.36066068	0.162737668	0.619048769	**1.142447117**
130	0.378835536	0.150219386	0.787066057	**1.316120979**

Table 6.19: Power values at forward flight at 2000 meters

	Profile Power	Induced Power	Parasitic Power	Total Power
10	0.203173111	1.137871769	0.00029499	**1.341339869**
20	0.204899313	0.930785012	0.002359919	**1.138044244**
30	0.207776318	0.723424246	0.007964726	**0.93916529**
40	0.211804124	0.570172166	0.01887935	**0.800855639**
50	0.216982731	0.474322141	0.03687373	**0.728178602**
60	0.223312141	0.395268451	0.063717805	**0.682298397**
70	0.230792352	0.338801529	0.101181515	**0.670775396**
80	0.239423364	0.296451338	0.151034798	**0.686909501**
90	0.249205179	0.263512301	0.215047593	**0.727765073**
100	0.260137795	0.237161071	0.29498984	**0.792288706**
110	0.272221213	0.215600973	0.392631477	**0.880453663**
120	0.285455433	0.197634225	0.509742444	**0.992832102**
130	0.299840454	0.182431593	0.648092679	**1.130364725**

Glauert's high speed formula gives 0.278978859332098 kW induced power for 70 km/h at sea level. The induced velocity at 70 km/h at sea level is 1.18776947545784 m/s. With the induced velocity which is obtained from numerical approach, the induced power is 0.277197818475472 kW. By these data the error percentage of Glauert high-speed formula can be stated as approximately 6%.

The variation of powers for changing forward velocity at sea level and 2000 meters and the total power curves of these two altitudes can be seen the figures given below.

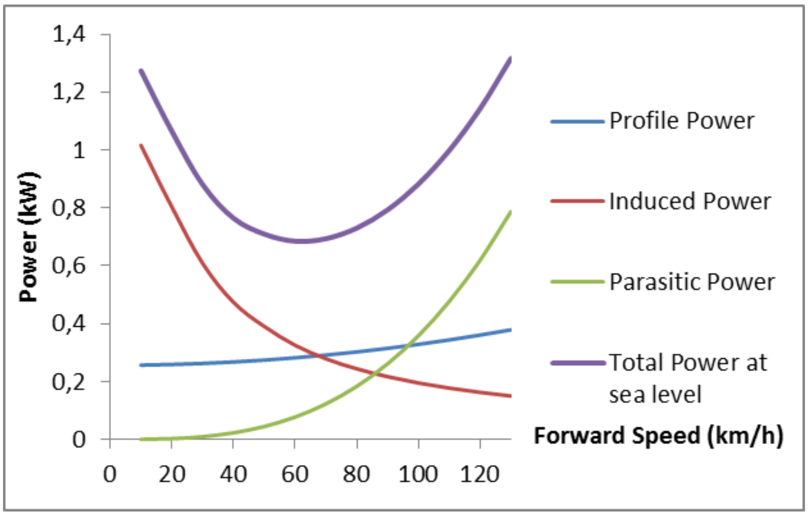

Figure 6.4 : Power versus forward speed at sea level

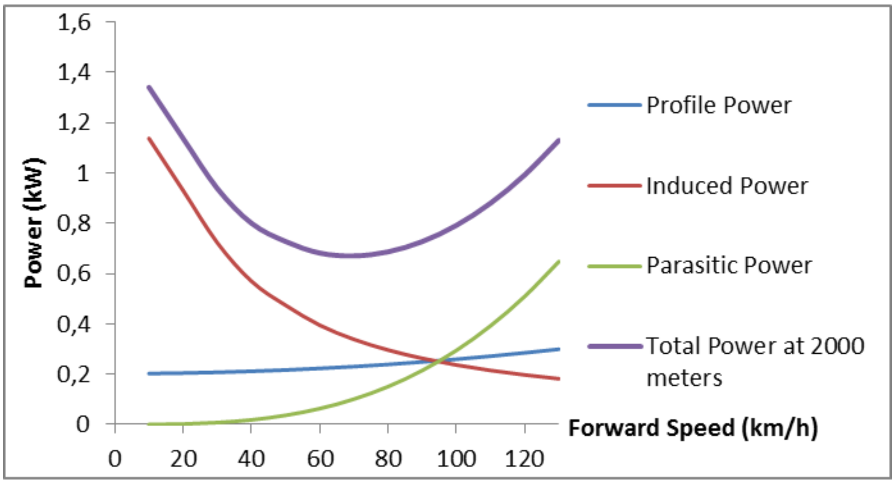

Figure 6.5 : : Power versus forwrad speed at 2000 meters

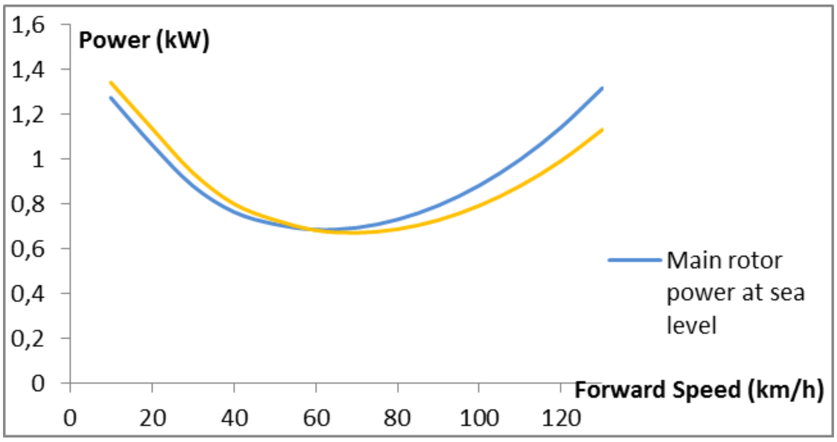

Figure 6.6 : Comparison of total power values obtained at different altitudes

The tip Mach number of the advancing side of the blade is checked.

Table 6.20: Tip mach number for last design

Forward speed km/h	Forward speed m/s	Tip Mach number
10	2.777777778	0.340022842
20	5.555555556	0.348180780
30	8.333333333	0.356338718
40	11.11111111	0.364496655
50	13.88888889	0.372654593
60	16.66666667	0.380812531
70	19.44444444	0.388970468
80	22.22222222	0.397128406
90	25.00000000	0.405286344
100	27.77777778	0.413444281
110	30.55555556	0.421602219
120	33.33333333	0.429760157
130	36.11111111	0.437918094
Tip speed	113	
Speed of sound	340.5	

At specified cruise speed 80 km/h some specifications of main rotor can summarized as;

Table 6.21: Specifications of last design

Gross weight	20.686680182973 m
Cruise speed	80 km/h
Maximum Forward speed	130 km/h
Blade tip speed	113 m/s
Thrust coefficient	0.00331460259294509
Solidity	0.0319547447622814
Maximum advance ratio	0.319567354965585
Mean lift coefficient	0.622368155515528
Airfoil	Boeing-Vertol VR7
Main rotor radius	1.11620187258989 m
Chord	0.0560270784696832 m
Blade twist	8.3 degrees
Airfoil lift-curve slope	8.2827 /rad
Airfoil profile drag coefficient(sea level)	0.010174045641733
Number of blades	2
Aspect ratio of main rotor	1,9922542868157500
Induced power at hover	1.12515470625714 kW
Profile power at hover	0.255973507368591 kW
Total power OGE at hover	1.38112821362573 kW
Total power IGE at hover	1.33804 kW for 2 meters
Power at hover with twisted blades	1.0920618 kW
Induced power at forward flight(80 km/h)	0.244106502 kW
Profile power at forward flight(80km/h)	0.302501 kW
Parasitic power at forward flight(80 km/h)	0.183421858 kW
Total power at 80km/h	0.730029 kW

6.2. Tail rotor design

Radius of the tail rotor is roughly one sixth of the radius of the main rotor and the tip speed of tail rotor is approximately is the same as the main rotor tip speed. For small helicopters, the ratio of the radius of the main rotor and tail rotor can be approximated as six, but with increasing gross weight, this ratio will be much more smaller than that. At this work, the ratio will be accepted as six and the tip speed of the tail rotor approximated as 113 m/s. NACA 0012 airfoil will be used for tail rotor blades.

$$R_{tr} = 0.186033\, m \tag{6.48}$$

$$V_{tip,tr} = 113 m/s \tag{6.49}$$

The airfoil is used for main rotor is different so the drag coefficient of the tail rotor can be stated as 0.011 for NACA 0012. As it was mentioned at Chapter 2 in trade studies, the number of blades of the tail rotor is 2. The distance from the main rotor shaft and the tail rotor shaft can be approximated as;

$$L = R_{mr} + R_{tr} + 0.2 = 1.5022\, m \tag{6.50}$$

The aspect ratios for tail rotors can be changed between 4.5 and 8. [5] By the definition an approximated chord value can be obtained for an aspect ratio as 6.

$$c_{tr} = \frac{R_{tr}}{AR} = 0.031\, m \tag{6.51}$$

Power needed for tail rotor at hover will be much more than power needed at forward flight. But at high forward speeds, the power sometimes can exceed the hovering value.

The thrust needed for tail rotor can be stated as;

$$T_{tr} = \frac{P_{total,mr}}{\Omega_{mr} L} \tag{6.52}$$

And the thrust coefficient can be found by;

$$C_{T,tr} = \frac{T_{tr}}{\rho A V_{tip,tr}{}^2} \tag{6.53}$$

Tip-loss factor B will be calculated with;

$$B = 1 - \frac{c}{2R} \tag{6.54}$$

or tip-loss factor can be stated with thrust coefficient as;

$$B = 1 - \frac{\sqrt{2C_{T,tr}}}{N_{b,tr}} \tag{6.55}$$

The induced and profile powers for tail rotor can be stated as;

$$(P_i)_{tr} = \frac{1}{B}\frac{T_{tr}^{3/2}}{\sqrt{2\rho A_{tr}}} \tag{6.56}$$

$$(P_0)_{tr} = \frac{\sigma C_{d0}}{8}\rho A_{tr} V_{tip,tr}{}^{3} \tag{6.57}$$

$$(P_{total,hover})_{tr} = (P_i)_{tr} + (P_0)_{tr} \tag{6.58}$$

The power values for the hovering can be seen from the tables above. The profile drag coefficient is assumed to be constant as NACA 0012 airfoil is used for tail rotor zero twisted blades. There are four tables located below which two are constructed for two different altitudes,namely sea level and 2000 meters and the other two constructed for illustrating zero-twisted main rotor blades or twisted main rotor blades conditions.

The weight approximations can not be used for unmanned helicopters, these approximations are given for example. Tail rotor power values also depends on these calculations, so the values calculated here are not correct actually.

Hovering performance is the key point at design process because power required at forward flight for both main and tail rotors will be smaller than the hovering rotors except high forward speeds. Tail rotor power will be calculated for different forward flight speeds and of course because of the changing profile drag coefficients, the tail rotor's performance is affected.

Table 6.22: Tail rotor power for untwisted main rotor blades at sea level

Main rotor total power at sea level with zero twisted main rotor blades	**1.381128214 kW**
Main rotor rotational velocity	101.2361677 rad/s
Distance between main and tail rotor shafts	1.502235518 m
Thrust required at tail rotor at hover	9.081556191 N
Thrust coefficient of tail rotor at hover	0.005339916
Tip-loss factor	0.948328363
Density	1.225 kg/m^3
Radius of main rotor	1.116201873 m
Radius of tail rotor	0.186033645 m
Disk area of tail rotor	0.108725863 m^2
Aspect ratio of tail rotor	6
Chord of tail rotor	0.031005608 m
Solidity of tail rotor	0.106060606
Profile drag coefficient of main rotor	0.010174046
Profile drag coefficient of tail rotor	0.011
Tip speed of tail rotor	113 m/s
Number of blades of tail rotor	2
Induced power at hover for tail rotor	55.91549795 W
Profile power at hover for tail rotor	28.02599804 W
Total power at hover OGE for tail rotor	**83.94149598 W**

Table 6.23: Tail rotor power for untwisted main rotor blades at 2000 meters

Main rotor total power at 2000 meters with zero twisted main rotor blades	**1.442533 kW**
Main rotor rotational velocity	101.2361677 rad/s
Distance between main and tail rotor shafts	1.502235518 m
Thrust required at tail rotor at hover	9.485321035 N
Thrust coefficient of tail rotor at hover	0.006770574
Tip-loss factor	0.941816781
Density	1.0087 kg/m^3
Radius of main rotor	1.116201873 m
Radius of tail rotor	0.186033645 m
Disk area of tail rotor	0.108769626 m^2
Aspect ratio of tail rotor	6
Chord of tail rotor	0.031005608 m
Solidity of tail rotor	0.106060606
Profile drag coefficient of main rotor	0.009779289
Profile drag coefficient of tail rotor	0.011
Tip speed of tail rotor	113 m/s
Number of blades of tail rotor	2
Induced power at hover for tail rotor	66.21582622 W
Profile power at hover for tail rotor	23.08669617 W
Total power at hover OGE for tail rotor	**89.30252239 W**

Table 6.24: Tail rotor power for twisted main rotor blades at sea level

Main rotor total power at sea level with twisted main rotor blades	**1.0920618 kW**
Main rotor rotational velocity	101.2361677 rad/s
Distance between main and tail rotor shafts	1.502235518 m
Thrust required at tail rotor at hover	7.18081095 N
Thrust coefficient of tail rotor at hover	0.004220587
Tip-loss factor	0.954062068
Density	1.225 kg/m^3
Radius of main rotor	1.116201873 m
Radius of tail rotor	0.186033645 m
Disk area of tail rotor	0.108769626 m^2
Aspect ratio of tail rotor	6
Chord of tail rotor	0.031005608 m
Solidity of tail rotor	0.106060606
Profile drag coefficient of main rotor	0.010174046
Profile drag coefficient of tail rotor	0.011
Tip speed of tail rotor	113 m/s
Number of blades of tail rotor	2
Induced power at hover for tail rotor	39.07030051 W
Profile power at hover for tail rotor	28.03727849 W
Total power at hover OGE for tail rotor	**67.107579 W**

Table 6.25: Tail rotor power for twisted main rotor blades at 2000 meters

Main rotor total power at 2000 meters with twisted main rotor blades	**1.156943 kW**
Main rotor rotational velocity	101.2361677 rad/s
Distance between main and tail rotor shafts	1.502235518 m
Thrust required at tail rotor at hover	7.60743482 N
Thrust coefficient of tail rotor at hover	0.005430148
Tip-loss factor	0.947893626
Density	1.0087 kg/m^3
Radius of main rotor	1.116201873 m
Radius of tail rotor	0.186033645 m
Disk area of tail rotor	0.108769626 m^2
Aspect ratio of tail rotor	6
Chord of tail rotor	0.031005608 m
Solidity of tail rotor	0.106060606
Profile drag coefficient of main rotor	0.009779289
Profile drag coefficient of tail rotor	0.011
Tip speed of tail rotor	113 m/s
Number of blades of tail rotor	2
Induced power at hover for tail rotor	47.25501791 W
Profile power at hover for tail rotor	23.08669617 W
Total power at hover OGE for tail rotor	**70.34171408 W**

$$T_{tr} = \frac{P_{total,mr}}{\Omega_{mr}L} = \frac{(P_{i,f}+P_{0,f}+P_p)_{mr}}{\Omega_{mr}L} \quad \textbf{(6.59)}$$

where at forward flight, total power which is the sum of profile, parasitic and induced power was calculated before for different airspeeds. These data will be used for obtaining the required thrust for tail rotor. Then thrust coefficient,tip-loss factor and induced power are obtained for different advance ratios.The data at sea level can be seen the table below.

Table 6.26: Tail rotor power at forward flight at sea level

Forward speed	Total Power	Thrust required at tail rotor to encounter torque	Thrust coefficient	Tip-loss factor	Profile power	Induced power	Tail rotor power
10	1.273315348	8.372636783	0.004923075	0.950386118	28.1056	49.39054	77.49614004
20	1.065195827	7.004154761	0.004118413	0.95462152	28.34439	37.62296	65.96734147
30	0.880587275	5.790268231	0.003404653	0.958740741	28.74237	28.15771	56.90008386
40	0.764542633	5.027221086	0.002955984	0.961555336	29.29955	22.71267	52.01222263
50	0.709499492	4.665287003	0.002743169	0.962965092	30.01592	20.27484	50.29076909
60	0.685001731	4.504202904	0.002648452	0.963610084	30.89149	19.22101	50.11249813
70	0.693453398	4.559776523	0.002681129	0.96338628	31.92625	19.58238	51.50863116
80	0.730029498	4.800281281	0.002822545	0.962433094	33.12021	21.17288	54.29309366
90	0.793004798	5.214372974	0.003066029	0.960846271	34.47336	24.01042	58.48378019
100	0.882203949	5.80089861	0.003410904	0.958702884	35.9857	28.23641	64.22210964
110	0.998296993	6.564264021	0.003859759	0.956069606	37.65724	34.08318	71.74042028
120	1.142447117	7.512117691	0.004417093	0.953004825	39.48798	41.85998	81.34795394
130	1.316120979	8.654103584	0.005088576	0.949559065	41.4779	51.94714	93.42504598

Table 6.27: Tail rotor power at forward flight at 2000 meters

Forward speed	Total Power	Thrust required at tail rotor to encounter torque	Thrust coefficient	Tip-loss factor	Profile power	Induced power	Tail rotor power
10	**1.341339869**	8.819929442	0.00629562	0.943894649	23.15226505	59.24126416	82.3935292
20	**1.138044244**	7.48316677	0.00534145	0.948320959	23.34897169	46.08111318	69.43008487
30	**0.939165290**	6.175445745	0.004408	0.953053212	23.67681608	34.37442891	58.05124498
40	**0.800855639**	5.265995885	0.00375884	0.956647715	24.13579823	26.96614773	51.10194596
50	**0.728178602**	4.788110784	0.00341773	0.958661584	24.72591814	23.33088085	48.05679899
60	**0.682298397**	4.486427234	0.00320239	0.959985071	25.4471758	21.13180711	46.57898292
70	**0.670775396**	4.410658179	0.00314831	0.960324406	26.29957123	20.59146809	46.89103931
80	**0.686909501**	4.516747372	0.00322403	0.959850085	27.28310441	21.34938987	48.63249428
90	**0.727765073**	4.785391638	0.00341579	0.958673324	28.39777535	23.31072396	51.70849931
100	**0.792288706**	5.209664338	0.00371863	0.956880213	29.64358404	26.52816525	56.1717493
110	**0.880453663**	5.789389674	0.00413244	0.954544324	31.0205305	31.153242	62.1737725
120	**0.992832102**	6.528329833	0.00465989	0.951730503	32.52861471	37.41444599	69.9430607
130	**1.130364725**	7.432670384	0.0053054	0.948495619	34.16783667	45.6070683	79.77490497

As it can be seen from the tables, at very low or very high airpeeds the power required to transmitted to tail rotor is exceeds the hovering value. Profile drag coefficient is assumed to be constant for tail rotor calculations and the expression which gives the induced part of the total power at forward flight is constructed to obtain a thrust to encounter the main rotor torque.

6.3. Total power calculations

Total power for hover can be stated as;

$$P_{total,hover} = P_{total,main\ rotor} + P_{total,tail\ rotor} \quad \textbf{(6.60)}$$

At sea level and the operational altitude 2000 meters the total power required for hover for the design with zero-twisted main rotor blades and with twisted main rotor blades can be seen at the table below.

Table 6.28: Total power at hover

	Main rotor power for hover OGE	Tail rotor power for hover OGE	Total power for hover OGE
Zero-twisted main rotor blades at sea level	1.381128 kW	0.0839414 kW	1.465069 kW
Zero-twisted main rotor blades at 2000 m	1.442533 kW	0.0893025 kW	1.531835 kW
Twisted main rotor blades at sea level	1.0920618 kW	0.067107 kW	1.159169 kW
Twisted main rotor blades at 2000 m	1.156943 kW	0.070341 kW	1.2272847 kW

Total power for forward flight is the sum of main rotor and tail rotor powers at specified speed values.It can be stated as;

$$P_{total,forward@V_{forw}} = P_{total,main\ rotor@V_{forw}} + P_{total,tail\ rotor@V_{forw}} \quad \textbf{(6.61)}$$

All these values can be seen the tables which are constructed for both sea level and 2000 meters are given at Table 6.29 and Table 6.30. As it can be seen from the tables with the increasing altitude, power requirements of helicopter durng forward flight decreases. This happens because of decreasing air drag force with increasing altiude,therefore decreasing parasitic drag.

Table 6.29: Total power at forward flight at sea level

Forward speed	Main rotor power	Tail rotor power	Total power at forward flight at sea level
10	1.273315348	0.07749614	1.350811488
20	1.065195827	0.065967341	1.131163168
30	0.880587275	0.056900084	0.937487359
40	0.764542633	0.052012223	0.816554855
50	0.709499492	0.050290769	0.759790261
60	0.685001731	0.050112498	0.735114229
70	0.693453398	0.051508631	0.74496203
80	0.730029498	0.054293094	0.784322592
90	0.793004798	0.05848378	0.851488578
100	0.882203949	0.06422211	0.946426059
110	0.998296993	0.07174042	1.070037414
120	1.142447117	0.081347954	1.223795071
130	1.316120979	0.093425046	1.409546025

Table 6.30: Total power at forward flight at 2000 meters

Forward speed	Main rotor power	Tail rotor power	Total power at forward flight at 2000m
10	1.341339869	0.082393529	1.4237334
20	1.138044244	0.069430085	1.20747433
30	0.93916529	0.058051245	0.99721653
40	0.800855639	0.051101946	0.85195759
50	0.728178602	0.048056799	0.7762354
60	0.682298397	0.046578983	0.72887738
70	0.670775396	0.046891039	0.71766644
80	0.686909501	0.048632494	0.735542
90	0.727765073	0.051708499	0.77947357
100	0.792288706	0.056171749	0.84846046
110	0.880453663	0.062173772	0.94262744
120	0.992832102	0.069943061	1.06277516
130	1.130364725	0.079774905	1.21013963

The power required will be selected as the power at maximum forward speed. 1.86267959 kW for sea level and 1.61449467 kW for 2000 meters altitude are considered and then compared with hovering values. All of these values will be compared during the total power calculation step and if the engine will be too powerful again, it needs to be replaced again and all of these steps should be redone again.

Table 6.31: Comparison of power values for new design

	Maximum power at forward flight	Power at hover with zero twist	Power at hover with twist
At sea level	1.409546025 kW	1.465069 kW	1.159169 kW
At 2000 meters	1.4237334 kW	1.531835 kW	1.2272847 kW

Power needs of helicopter can be seen from the table above. The biggest value of power at both sea level and operational altitude forward flight power tables is the value given above. Required rotor shaft power can be stated as the maximum power at the table namely 1.4237334 kW. The zero twisted conditions are not taken into account.

Required total engine shaft power can be calculated by considering fifteen percent of transmission and accessories losses. Maximum engine power output is 2.24 kW. So,the required total engine shaft power can be stated as;

$$P_{rotorshaftavailable} = (0.85)P_{engineshaft} = 1.904\ kW \tag{6.62}$$

A safety factor should be taken into account as 1.2 to overcome the instability conditions, other environmental effects and inaccuracy rate of momentum theory, modified momentum theory, BET and BEMT [16]. The rotor shaft power with respect to these statements can be defined as;

$$P_{safetyrotor} = 1.2P_{rotorshaftrequired} = 1.2(1.4237334) = 1.708\ kW \tag{6.63}$$

So the engine is said to be suitable for the new design. The difference between the available and the required shaft powers are considered in determining the climb performance at Chapter 7.

7. PERFORMANCE ANALYSIS

7.1. Effect of density altitude

Two different altitudes was considered at this work;sea level and 2000 meters. With the increasing altitude, the power required is expected to increase in hover and lower airspeeds but at higher airspeeds with increasing altitude the power required is expected to decrease because of the lower density results lower parasitic drag. Higher density altitude actually affects the engine’s performance but it has not been taken into account here because of the decreasing power requirements of helicopter with increasing altitude value.

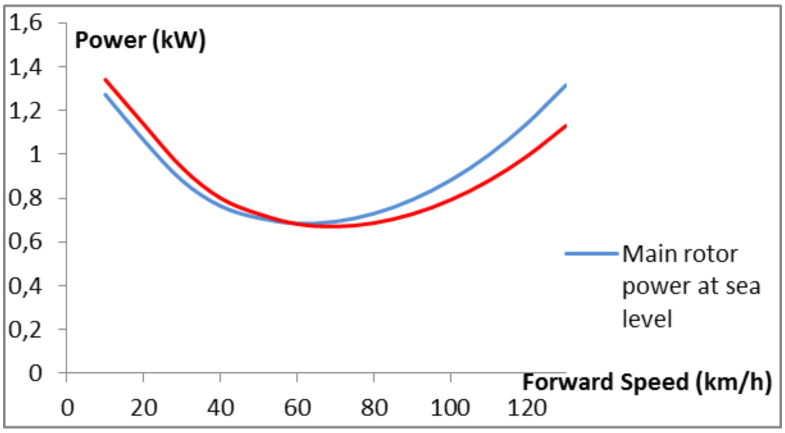

Figure 7.1: Power curves due to forward speed and different altitudes

These statements can be seen from the figure above. At 2000 meters at higher airspeed less power is required by rotors compared with sea level but at low airspeeds at 2000 meters higher power is required by rotors than the sea level.

7.2. Lift-to-Drag Ratios

The lift-to-drag ratio of the rotor alone is; [4]

$$\frac{L}{D} = \frac{T\cos\alpha_{TPP}}{(P_i + P_0)/V_{forw}} \cong \frac{GW\ x(9{,}81)x\ V_{forw}}{P_i + P_0} \tag{7.1}$$

The lift-to-drag ratio of the complete helicopter can be stated as;

$$\frac{L}{D} = \frac{T\cos\alpha_{TPP}}{(P_i+P_0+P_p+P_{tr})/V_{forw}} \cong \frac{GW\ x(9{,}81)x\,V_{forw}}{P_i+P_0+P_p} \qquad \textbf{(7.2)}$$

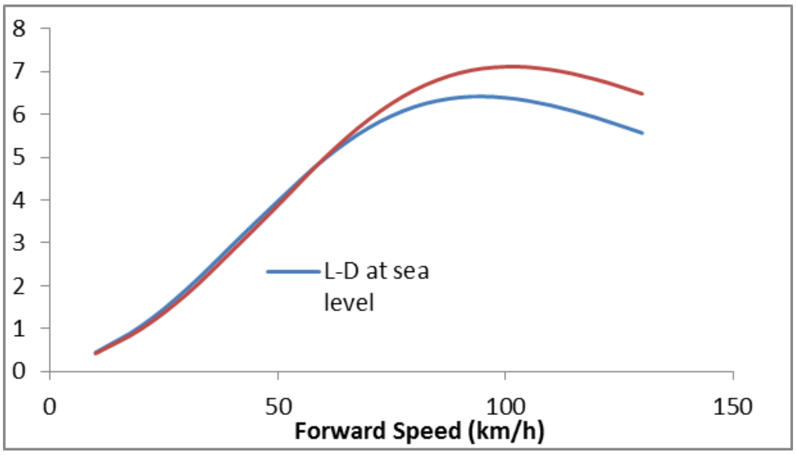

Figure 7.2: Lift to drag ratios of design

L/D increases raipdly as induced power requirements decrease and reaches a maximum,then drops off as the parasitic power requirements rapidly increase.

The values that is plotted at the figure above can be seen at the table given below.

Table 7.1: Lift to drag ratio values of design

Forward Speed	L/D at sea level	L/D at 2000m
10	0.442712	0.420260404
20	1.0584195	0.990668048
30	1.9204639	1.800679948
40	2.9492772	2.815548807
50	3.9726035	3.870698982
60	4.9376112	4.957174493
70	5.690338	5.882720605
80	6.1774165	6.565197125
90	6.3977019	6.97121709
100	6.3898154	7.114982592
110	6.2114105	7.042769702
120	5.9211007	6.813382048
130	5.5680721	6.483090183

7.3. Climb performance

The climb power can be defined as the time rate of increase of potential energy. The rate of increase of potential energy can be stated as;

$$P_{climb} = Increase\ rate\ of\ potential\ energy = W\dot{h} = TV_c = WV_c \qquad \textbf{(7.3)}$$

At here, it is assumed that thrust equals to weight of helicopter. However,there is usually an increment in thrust required to overcome weight because of vertical drag on the fuselage that results from the action of the rotor slipstream velocity. The vertical drag value is approximately 5% of the gross weight for conventional helicopters [4].

For the worst scenario, it is assumed that thrust required needs to increase 10% .

Thrust needs an increment because of vertical drag penalty.

$$T = GW + D_v \tag{7.4}$$

$$\Delta T = D_v = \frac{1}{2}\rho \bar{v}^2 f_v \tag{7.5}$$

At this formula, $\bar{v}$ is the average velocity in the rotor slipstream and f_v is the equivalent drag area.

The momentum theory states that for axial climb; [4]

$$\frac{v_i}{v_h} = -\left(\frac{V_c}{2v_h}\right) + \sqrt{\left(\frac{V_c}{2v_h}\right)^2 + 1} \tag{7.6}$$

For low rates of climb it is assumed that

$$\frac{v_i}{v_h} = -\left(\frac{V_c}{2v_h}\right) + \sqrt{\left(\frac{V_c}{2v_h}\right)^2 + 1} \approx 1 - \frac{V_c}{2v_h} \tag{7.7}$$

$\bar{v}$ is defined as the average velocity in the rotor slipstream; namely the wake velocity.

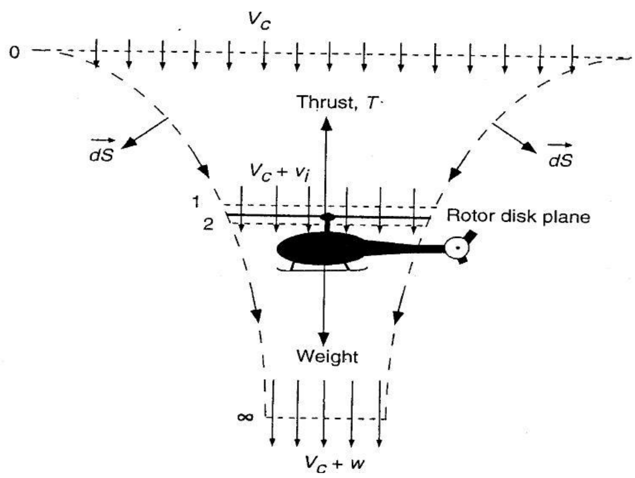

Figure 7.3: Climb [4]

Frım the figure given above it can be seen at the total wake velocity is equals to;

$$\bar{v} = V_c + w = V_c + 2v_i \qquad \textbf{(7.8)}$$

$$\frac{v_i}{v_h} = 1 - \frac{V_c}{2v_h} \qquad \textbf{(7.9)}$$

$$\frac{v_i}{v_h} = \frac{2v_h - V_c}{2v_h} \qquad \textbf{(7.10)}$$

$$2v_i = 2v_h - V_c \qquad \textbf{(7.11)}$$

$$2v_h = V_c + 2v_i = \bar{v} \qquad \textbf{(7.12)}$$

$$\bar{v} = 2v_h \qquad \textbf{(7.13)}$$

It can be seen that wake velocity is independent of climb velocity.

The rotor thrust can be stated as;

$$T = GW + D_v = GW + \frac{1}{2}\rho\bar{v}^2 f_v = GW + 2\rho {v_h}^2 f_v \qquad \textbf{(7.14)}$$

Induced velocity for hover can be written as;

$$v_h = \sqrt{\frac{T}{2\rho A}} \qquad \textbf{(7.15)}$$

So the thrust is;

$$T = GW + T\left(\frac{f_v}{A}\right) \qquad \textbf{(7.16)}$$

Rearranging and solving for thrust gives;

$$T = \frac{GW}{1 - \left({f_v}/{A}\right)} \qquad \textbf{(7.17)}$$

$\left({f_v}/{A}\right)$ is smaller than 0.1 [4]

$$\left({f_v}/{A}\right) < 0.1 \qquad \textbf{(7.18)}$$

For the worst scenario it can be assumed as equals to 0.1. So the thrust with vertical drag penalty can be stated as;

$$T = 1.11GW \quad \textbf{(7.19)}$$

It should be remembered that gross weight is in kilograms,thrust is in newton. The gross weight is also multiplied by gravitational constant g.For that new thrust value, climb velocity will be estimated.

The total power equation can be used to estimate the climb velocity at any forward speed.

$$V_c = \frac{P_{available} - (P_{tr} + P_0 + P_p + P_i)}{T} \quad \textbf{(7.20)}$$

It is assumed that for low rates of climb power, induced power,profile power and the airframe drag D remain constant.

$$V_c = \frac{P_{available} - P_{level}}{T} = \frac{\Delta P}{T} = \frac{\Delta P}{1.11GW} \quad \textbf{(7.21)}$$

Here; P_{level} is the net power required to maintain level flight conditions at the same forward speed, $P_{available}$ is the installed power and ΔP is the excess power available.

The output of selected is engine 2.24 kW and 15% of its output goes to transmission losses.The remaining part is 1.904 kW which is called actual rotor shaft power available. The available power with this engine is 1.904 kW. Required rotor shaft power is 1.46 kW which is the maximum power requirement of the helicopter. But at the end of Chapter 6, a safety factor is constructed as 1.2.This factor should be taken into account in order to model the losses because of off-design conditions such as environmental effects and modelling errors. This constant can not be 1.5, it just had taken into account for small losses. It does not represent all effects such as high speed winds or bad weather conditions. The level power varies with forward flight speed. It has been given at the tables before at Chapter 6.

Table 7.2: Maximum rate of climb with respect to forward velocity

Forward Speed(km/h)	Maximum rate of climb at sea level(m/s)	Maximum rate of climb at 2000m(m/s)
10	1.256446137	0.867977022
20	2.426555208	2.020030897
30	3.45830369	3.140114822
40	4.102534519	3.913937331
50	4.404930487	4.317324046
60	4.536384481	4.569609382
70	4.483923343	4.629332251
80	4.274242017	4.534105689
90	3.916435824	4.300073683
100	3.410685508	3.93256722
110	2.75218392	3.430921506
120	1.933087152	2.790871336
130	0.943555905	2.005832323

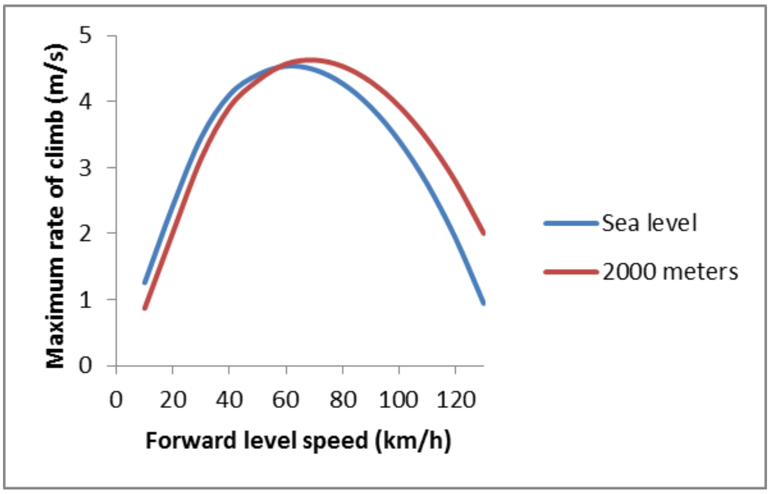

Figure 7.4 : Maximum rate of climb curves

The maximum point on the curve gives best rate of climb. At the excel a curve can be fitted into that scatter data. By getting derivative and making equals to zero of that curve's equation, an extremum point can be found. In order to determine these value at sea level, an excel curve will be fitted into that data.

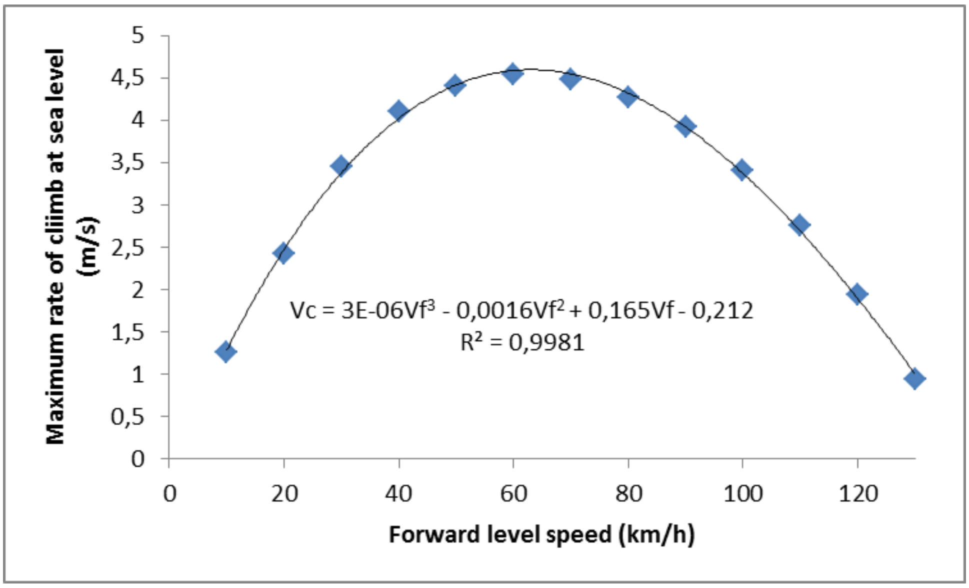

Figure 7.5 : Constructing maximum rate of climb formula

The maximum point on the curve gives best rate of climb. At the excel a curve can be fitted into that scatter data.By getting derivative and making equals to zero of that curve's equation, an extremum point can be found. Third degree polynomial curve fits these data with variation 0.9981.

By doing this calculations, forward speed coordinate of the pick point of the curve is 57.83 km/h. For this value, climb rate is approximately 4.56 m/s. This value is too high for that type of helicopters. The maximum rate of climb has to have a limit for the proper use of formulas and assumptions. While calculating the climb performance by the formulas given above, it was assumed that the helicopter will have low rates of climb. Similar unmanned helicopters have approximately 2.5 or 3 m/s best rate of climb. This result means the selected engine is too strong for that design, but actually in order to be capable of comprimising the power values the selected engine is not too strong.

7.4. Fuel consumption of the engine

The specific fuel consumption of the engine is 0.8041 [14] and at this work, fuel consumption is assumed as constant. Generally, SFC quickly approaches a nearly constant value as the engine reaches its maximum continuous rated power output. Clearly the engine operates more efficiently under these condition.

Fuel flow rates for constant SFC with respect to motor output will be a linear line. Various fuel flow rates can be seen the table given below.

Table 7.3: Fuel consumption of selected engine

Power output (kW)	Fuel consumption (kg/h)
0.00	0.000000
0.25	0.201025
0.50	0.402050
0.75	0.603075
1.00	0.804001
1.25	1.005125
1.50	1.206150
1.75	1.407175
2.00	1.608200
2.24	1.801184

7.5. Speed for minimum power and best endurance

Maximum rate of climb is obtained at the speed for minimum power in level flight. This speed value also determines the speed to give best endurance.To obtain maximum endurance, fuel burn per unit time must be minimum.The weight of fuel can be defined as;

$$W_{fuel} = SFC\ x\ Pt \qquad \textbf{(7.22)}$$

Or the endurance can be defined as

$$t = \frac{W_{fuel}}{SFC\ x\ P} \qquad \textbf{(7.23)}$$

At this design SFC assumed as constant 0.8041 and the maximum fuel weight that helicopter can carry is 1.8 kg. As it can be seen from climb performance; maximum rate of climb is obtained at 57.83 km/h. For that speed value, the required power of helicopter is 0.87 kW approximately with the safety factor 1.2. 57.83 km/h is the speed for minimum power or speed for best endurance. The endurance can be found as

$$t = \frac{W_{fuel}}{SFC\ x\ P} = \frac{1.8}{0.8045\ x\ 0.87} = 2.57\ hours \qquad \textbf{(7.24)}$$

The best endurance of the designed helicopter is 2.5 hours approximately. When the helicopter is on level flight at 57.83 km/h, it can reach at that values. The aimed endurance is one hour but it should be noted that these value is best endurance; not the endurance and the variation of specific fuel consumption of the engine is not considered.

7.6. Speed for maximum range

The range of helicopter is the distance it can fly with its payload and takeoff weight. It can be defined as;

$$R_{max} = \frac{V_{mr} W_{fuel}}{SFC \; x \; P_{V_{mr}}} \tag{7.25}$$

To obtain the maximum range value of the helicopter; it should be noted that V/P ratio has to be maximum for fixed fuel weight,1.8 kg, and constant SFC. Here V_{mr} is the maximum range speed and $P_{V_{mr}}$ is the power required at that forward speed. These data can be obtained from lift-to-drag ratios given before. Bu using the data at sea level;

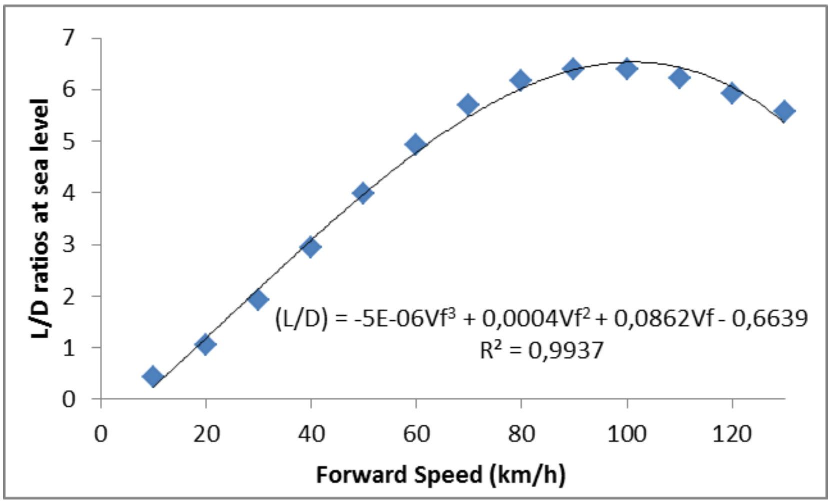

Figure 7.6 : Relationship between lift to drag ratio and forward speed

Best lift-to-drag ratio occurs approximately at 107 km/h.

$$V_{mr} = 107 \; km/h \tag{7.26}$$

For that speed value the power required by helicopter is 1.096 kW

The maximum range of the helicopter can be obtained as;

$$R_{max} = \frac{V_{mr} W_{fuel}}{SFC \; x \; P_{V_{mr}}} = \frac{107 \; x \; 1.8}{0.8045 \; x \; 1.096} \cong 218.4 \; km \tag{7.27}$$

The formula given above is actually does not completely represent the range value. The range will be higher than the result although this increase is very little. As the fuel burns, total weight decreases;so the power requirements of the helicopter decreases. At the formula which has been given above, the fuel weight is decreasing with time. Another approach can be made as;

$$\frac{dW_{fuel}}{dR} = \frac{P\;x\;SFC}{V} \qquad \textbf{(7.28)}$$

$$R = \int_{GW}^{GW-W_{fuel}} \frac{V}{P\;x\;SFC} dW \approx W_{fuel} \left[\frac{V}{P\;x\;SFC}\right]_{W=W'} \qquad \textbf{(7.29)}$$

where $W' = GW - \frac{W_{fuel}}{2}$

The power required for that value of weight can be calculated and by using the formulas, range at any cruise speed can be calculated. For small helicopters the fuel weight can be assumed as constant and the formulas given above affect the range values very little; so it can be neglected [4].

7.7. Ceiling

Two different ceiling definition is investigated here; hover ceiling and service ceiling. Hover ceiling is the maximum altitude that the helicopter is able to hover at and service ceiling is the the altitude that best power values for V_{mp}reaches available shatf power. At this work, it is assumed that engine's characteristics does not depend on altitude. Engine's performance actually decreases with increasing altitude.

Available rotor shaft power after transmission losses can be stated as;

$$P_{shaft} = 0.85 P_{engine} = 1.904\;kW \qquad \textbf{(7.30)}$$

During the calculation of climb performance a safety factor 1.2 has been taken into account. So a more realistic power value will be 1.52 kW.

Power at hover can be defined as;

$$P_{hover} = P_i + P_0 = \frac{\kappa T^{\frac{3}{2}}}{\sqrt{2\rho A_e}} + \rho A_e V_{tip}^3 \left(\frac{\sigma C_{do}}{8}\right) = P_{available} \qquad \textbf{(7.31)}$$

where effective rotor disk area $A_e = 3.563\ m^2$, thrust $T = GW\ x\ 9.81 = 202.936\ N$, induced power factor $\kappa = 1.15$, tip speed $V_{tip} = 113\ m/s$, solidity $\sigma = 0.031954$, profile drag coefficient $C_{d0} = 0.010174$

Solving the equation above gives that;

$$\rho = 0.859\ kg/m^3 \tag{7.32}$$

This is the highest altitudes' density value for hover.

$$\rho = \rho_0 e^{\frac{-0.0296h}{304.8}} \tag{7.33}$$

The altitude value is 3654.8 meters. This the hover ceiling of the designed helicopter. Actually an iteration process is needed therefore the profile drag coefficient also depends on density at the ceiling. But at ceiling calculations profile drag coefficient at sea level is accepted to modelling the maximum profile drag coefficient so minimum air density,maximum celing value. This means the ceiling values obtained here are overvalued.

Service ceiling is the altitude at which the total power required at speed for minimum power equals to available power transmitted to rotor. It actually models that the minimum power requirements of rotor reaches at available power at which density, so on at which altitude. It can be said that; the service ceiling is the altitude that helicopter can operate with its minimum power configuration. This speed value is 57.83 km/h.

$$P_{required} = P_i + P_0 + P_p + P_{tr} + P_c = P_{available} \tag{7.34}$$

Putting the values of powers for 57.83 km/h data as a function of unknown quantity ρ and solving gives the maximum altitude that helicopter can maintain a level flight; in another words service ceiling. To find climb power, the climb rate should be taken as a very small value [4].

Induced power at speed fo minimum power V_{mp} can be stated by Glauert's high speed formula as the advance ratio is larger than 0.1;

$$\mu = \frac{V_{mp}}{V_{tip}} = \frac{16.063}{113} = 0.14 > 0.1 \tag{7.35}$$

$$P_i = \frac{\kappa T^2}{2\rho A V_{mp}} \quad \textbf{(7.36)}$$

Profile power at speed for minimum power depends on profile drag coefficient and density as area, solidity and tip speed are constants.But at the service ceiling calculations the maximum value of profile drag coefficient, namely sea level value is used. Tail rotor power is approximately 6% of total power at this work. As the total power value reaches maximum available power values, the tail rotor power at the ceiling is assumed as constant equals to 6% available shaft power. Climb velocity will be accepted as 1m/s as a small value to be able to model the level flight conditions at the ceiling.

Available rotor shaft power after transmission losses can be stated as;

$$P_{shaft} = 0.85 P_{engine} = 1.904\ kW \quad \textbf{(7.37)}$$

During the calculation of climb performance a safety factor 1.2 has been taken into account. So a more realistic available power value will be 1.52 kW.

$$\frac{\kappa T^2}{2\rho A_e V_{mp}} + \rho A_e V_{tip}^3 \frac{\sigma C_{do}}{8} + \frac{1}{2} f V_{mp}^3 \rho + (0.06) P_{available} + P_{climb} = P_{available} \quad \textbf{(7.38)}$$

Solving this equation for density gives that

$$\rho = 0.67\ kg/m^3 \quad \textbf{(7.39)}$$

By using

$$\rho = \rho_0 e^{\frac{-0.0296h}{304.8}} \quad \textbf{(7.40)}$$

These values are too high when it is compared with similar helicopters. A value of approximately 2500 meters was expected for hover ceiling and 3500 meters for service ceiling.This overvaluation can be the result of selecting a more powerful engine than needed or using a constant profile drag coefficient. But actually it is the result of engine’s neglected performance; the engine’s performance has not been taken into account due to altitude change and this makes the calculations highly overvalued. It has been already stated as engine does not work above 4000 meters altitudes.

8. VISUAL DESIGN WITH CATIA

8.1. Main rotor

With respect to selected airfoil, radius, chord and twist, blades are constructed in Catia. VR7 airfoil is selected.

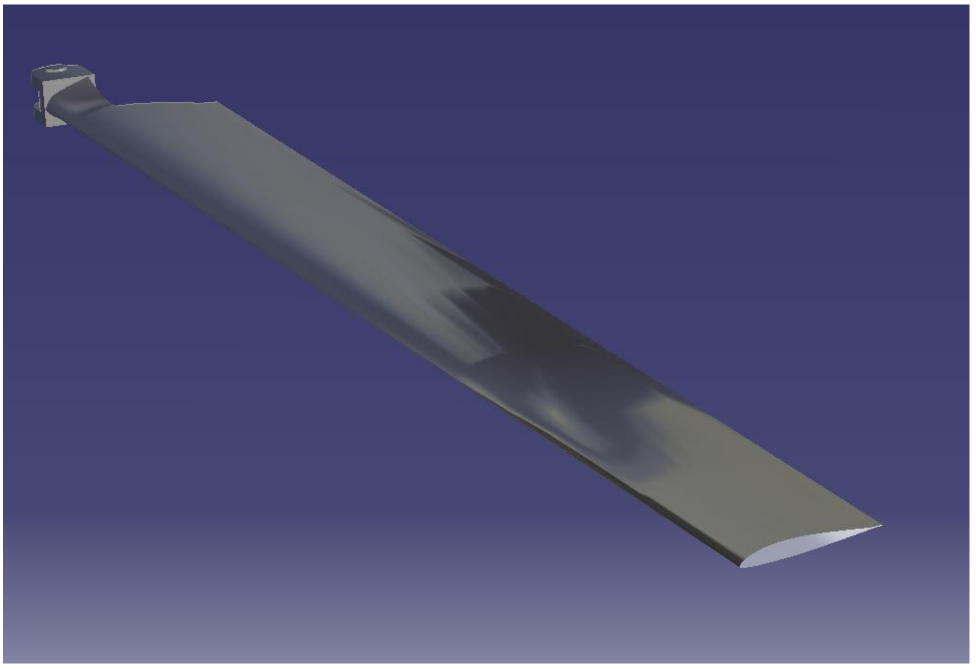

Figure 8.1 : Main rotor blade

A blade can be seen from the figure given above. Blade holder can be seen at the figure given below.

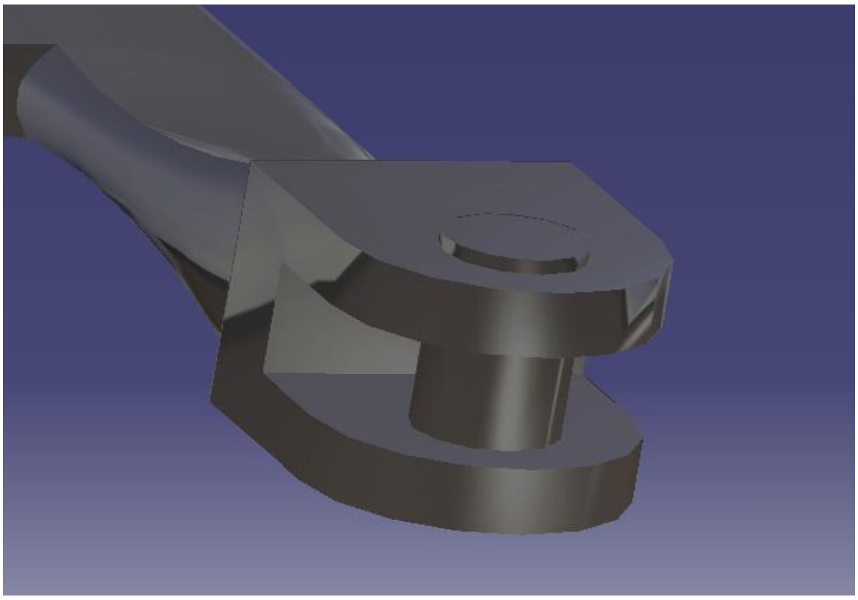

Figure 8.2 : Main rotor blade holder

Hub, flybar, paddles, hub roof, swashplate, pitch links,flexbeam, rotor shaft can be seen at the figure below. NACA63418 profile is used at paddles.

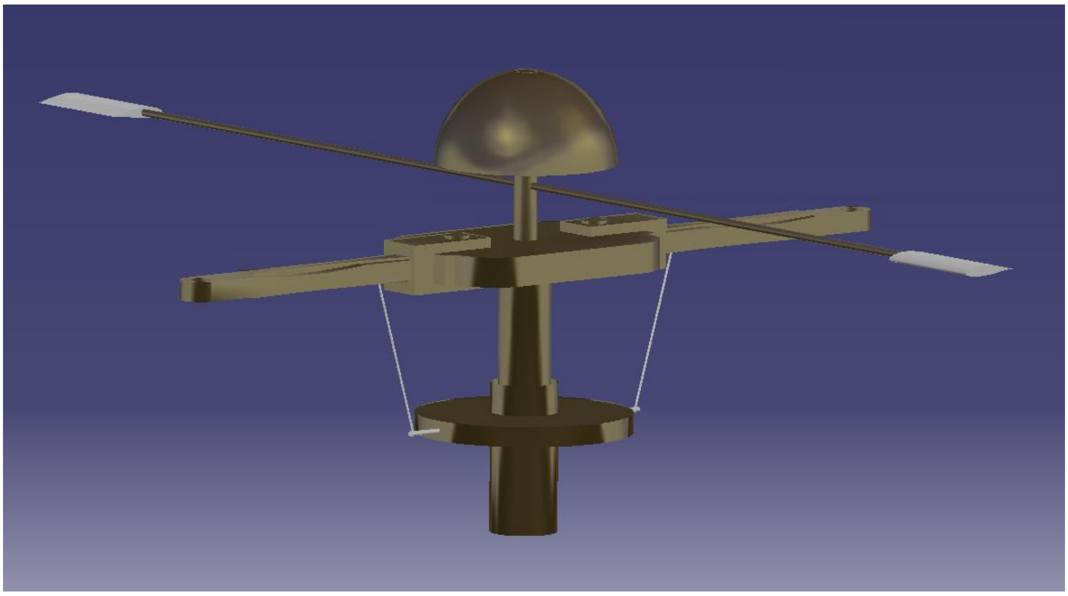

Figure 8.3 : Hub, swashplate, flybar and pitchlinks

Flexbeam can be seen more detailed at the figure below.

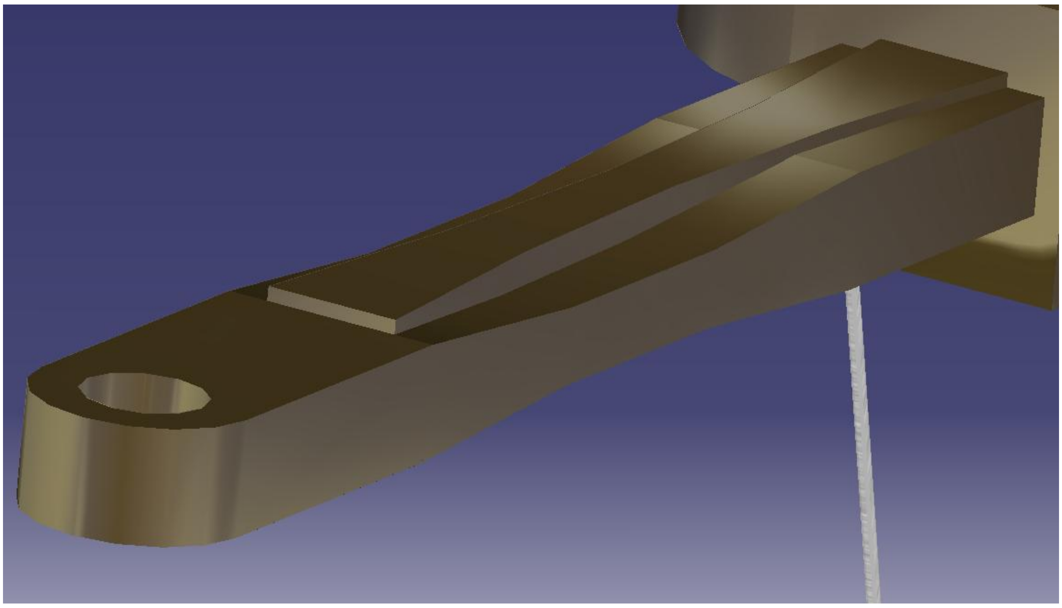

Figure 8.4 : Flexbeam

8.2. Tail rotor

NACA 0012 airfoil is selected for the tail rotor blades. The blades and joint mechanism can be seen at the figure below.

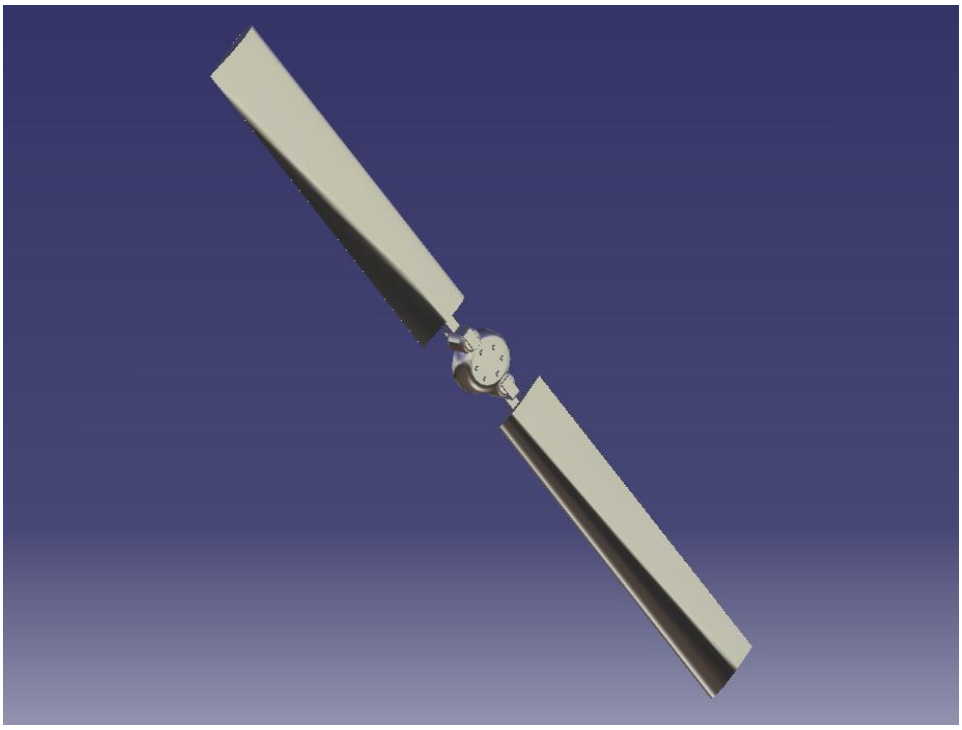

Figure 8.5 : Tail rotor blades and joint

8.3. Fuselage

Fuselage is designed for covering all the payloads and other equipments required for the mission. Aerodynamic behaviour of the fuselage has been taken into account visually while constructing the sketches. The helicopter needs to be relatively faster compared with the similar designs; so the fuselage is designed for the aim to obtain an aerodynamically efficient structure. The design can be seen from the figures given below.

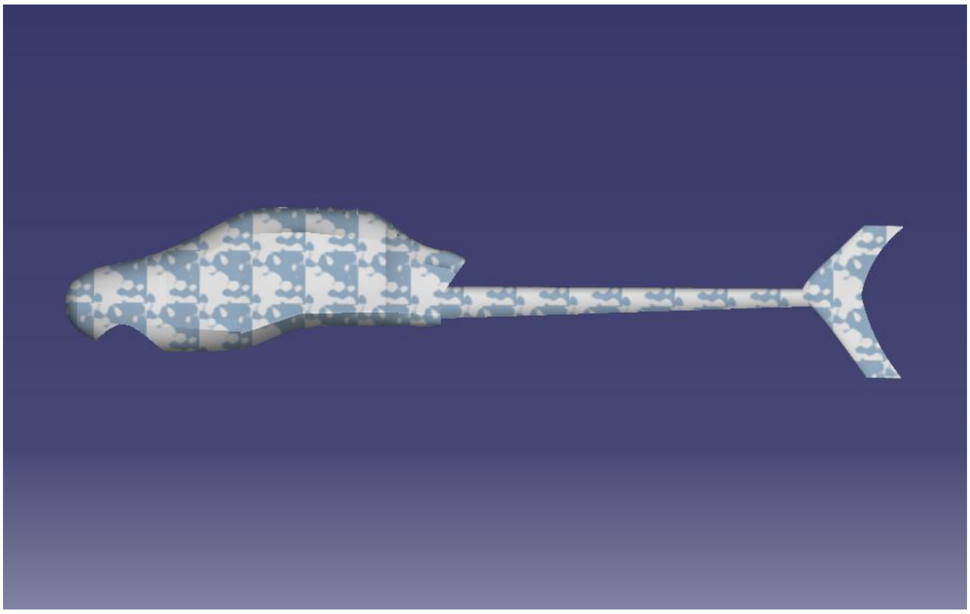

Figure 8.6 : Left view of fuselage

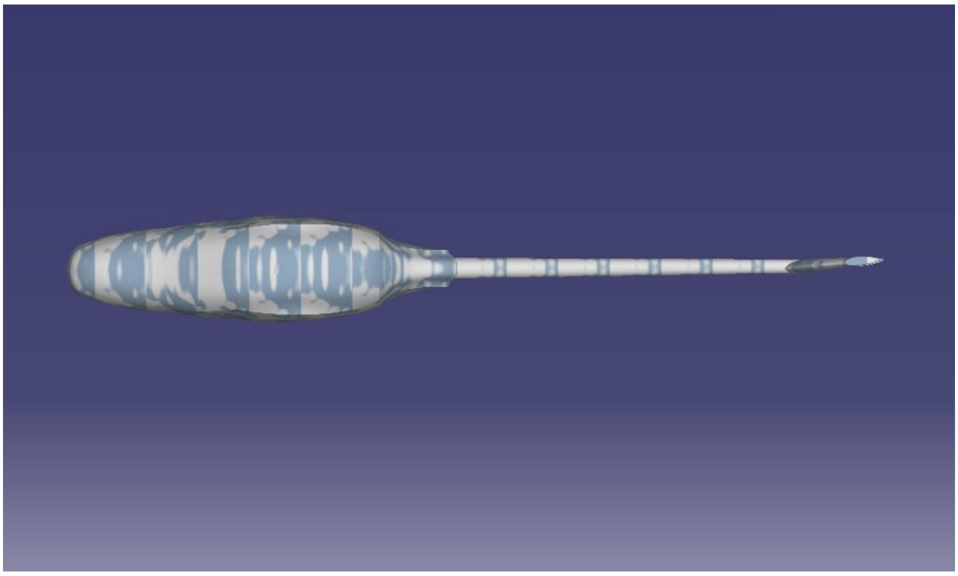

Figure 8.7 : Top view of fuselage

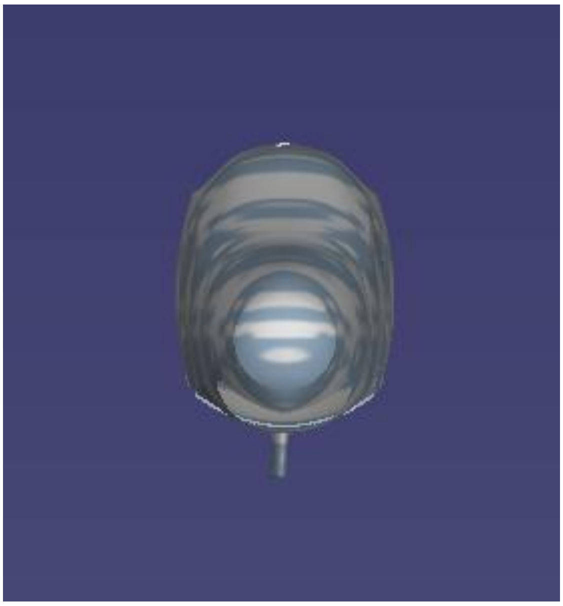

Figure 8.8 : Front view of fuselage

The exhaust of the engine and the tail joint are covered with roofs to prevent flow seperation. It can be seen at the figure.

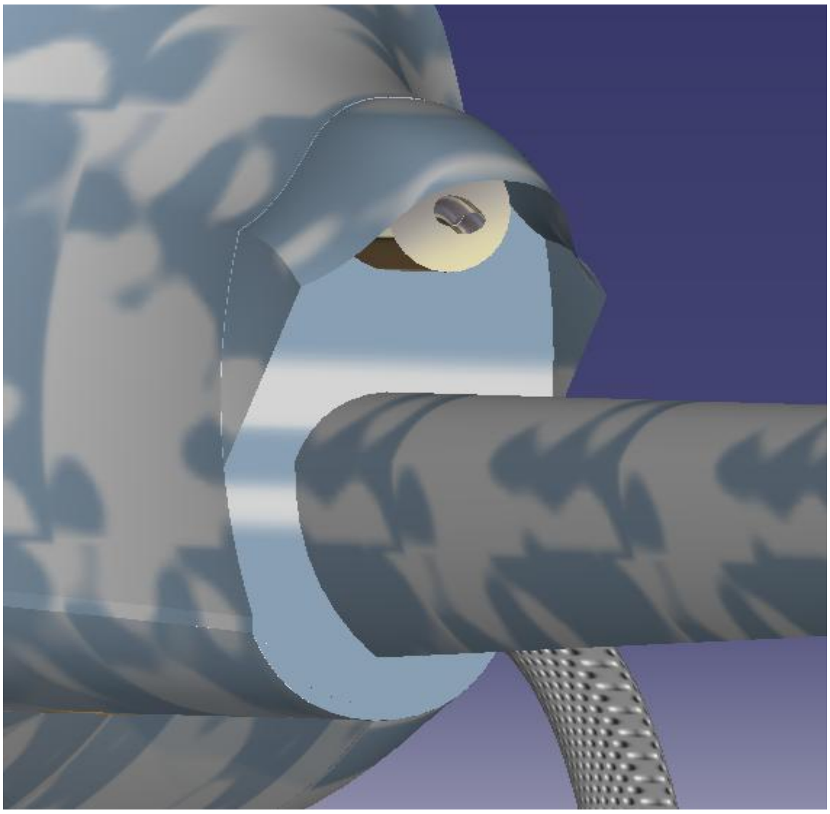

Figure 8.9 : Designed exhaust and its roof

The fuel tank is a seperated design and with the mounting of the tank the fuselage itself has a flat or another name smooth bottom surface for aerodynamic efficiency. The tank can be mounted to the fuselage easily and the fuel inputs to the fuselage, then be sent to the engine with a pump located in the fuselage. The fuel tank and its joint at the fuselage can be seen at the figures given below.

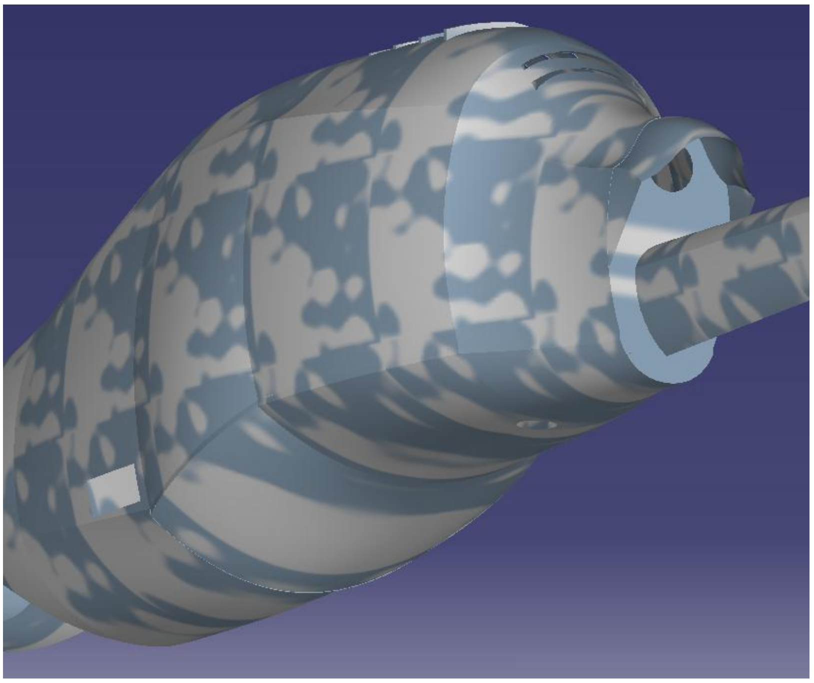

Figure 8.10 : Fuselage without fuel tank

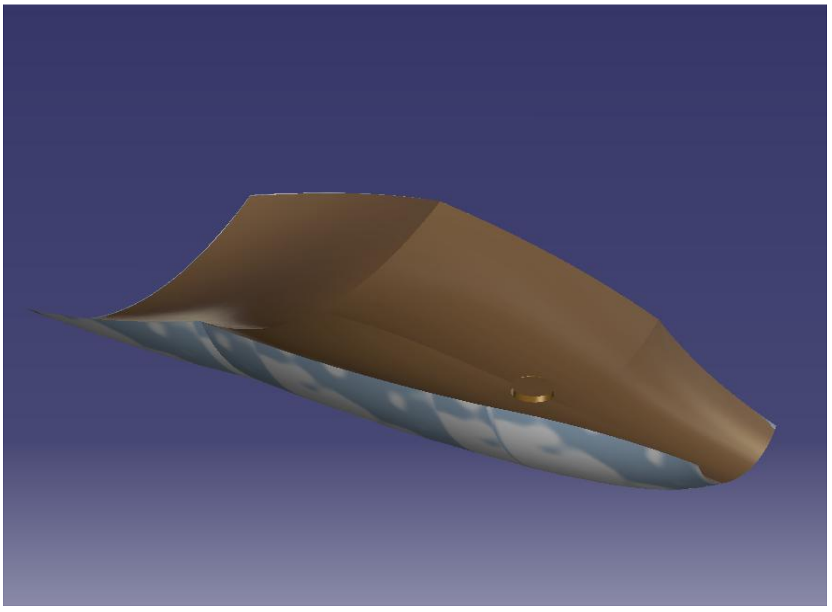

Figure 8.11: Fuel tank

To be able to cool the engine, air intakes and exhausts are designed visually. The engine air intakes can be seen at the figure below.

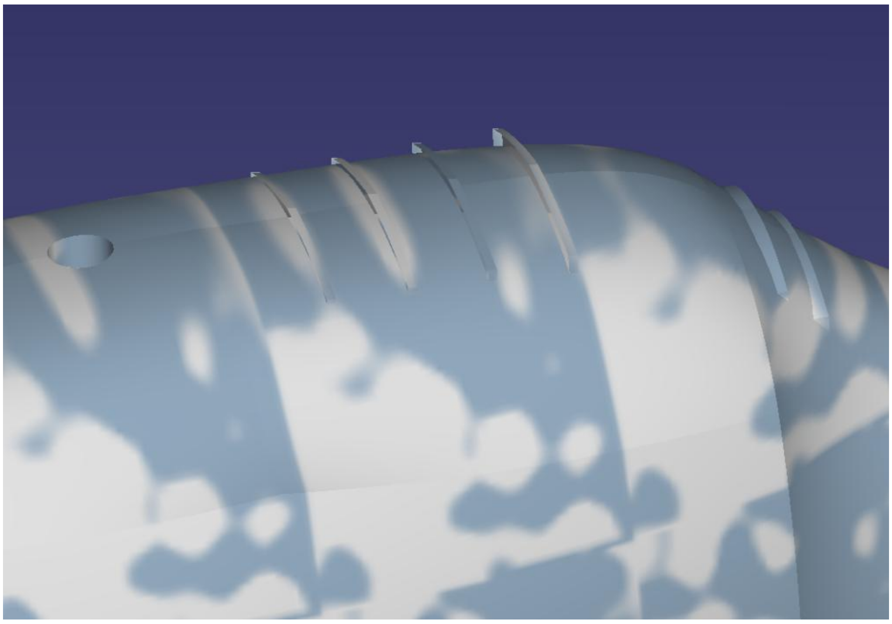

Figure 8.12: Air intakes

NACA63421 airfoil is used at the vertical stabilizers at the tail.It can be seen at the figure below.

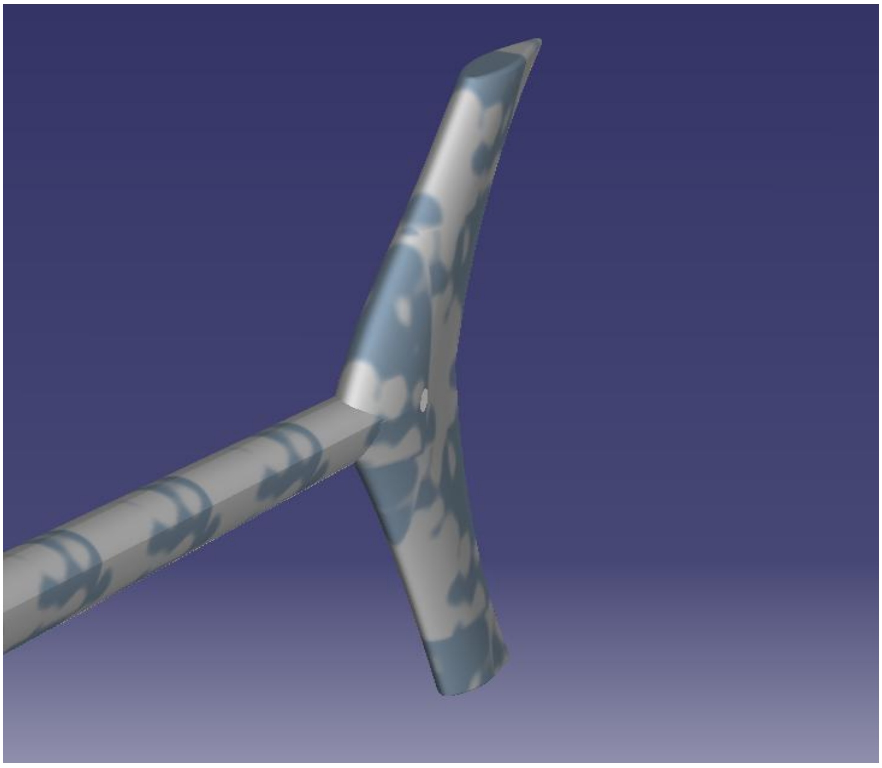

Figure 8.13 : Tail and vertical stabilizer

Twin skid landing gear is designed virtually. The fuel tank is located outside and under the fuselage; so the landing gear is fixed above the tank. The gear only can be seen detailed at the figure below.

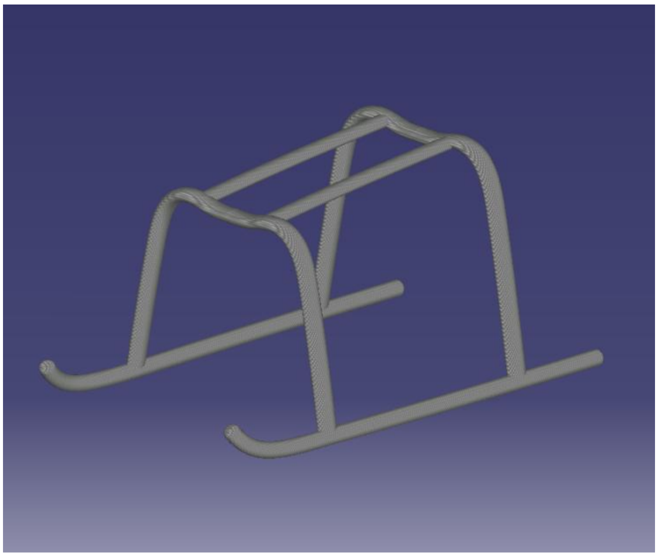

Figure 8.14 : Landing gear

Camera, mission computer, transmission system, engine,fuel pump can be seen at the figure given below. The camera is wanted to keep at the most possible forward direction in the fuselage in order to obtain a reasonable center of weight distribution.

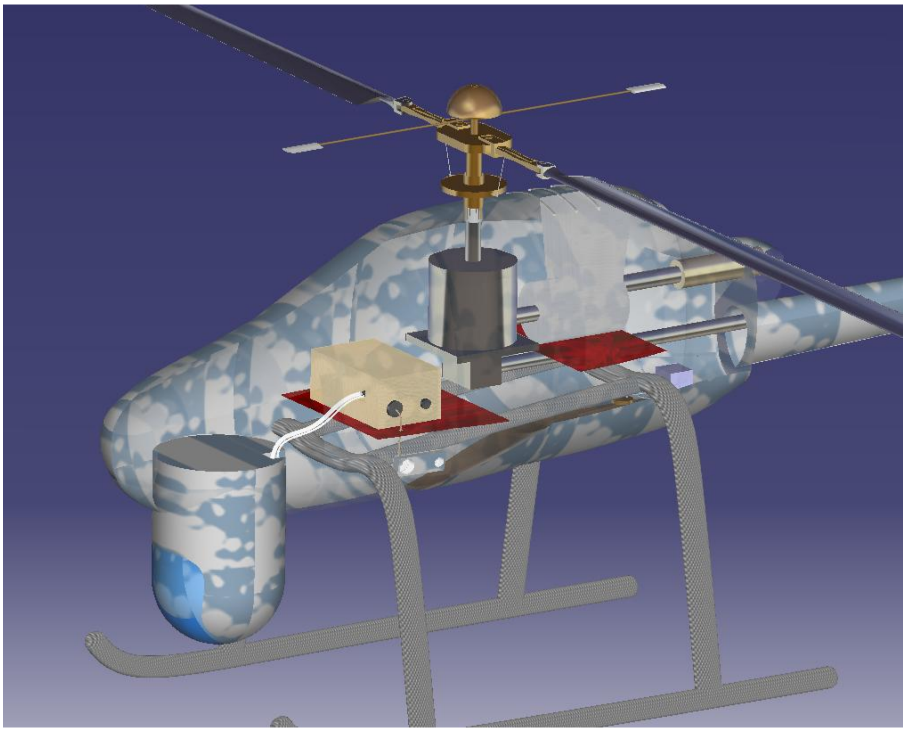

Figure 8.15 : An overview of interior

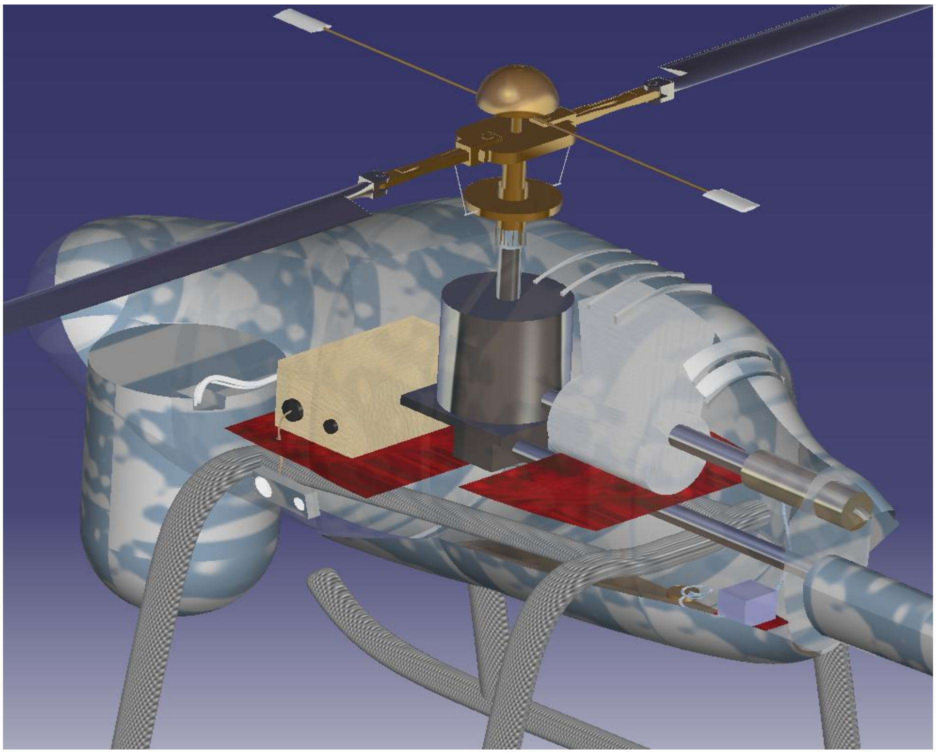

Figure 8.16 : Another overview of interior

The complete virtual design based on the calculations given above can be seen at the figure given below.

Figure 8.17 : Complete design

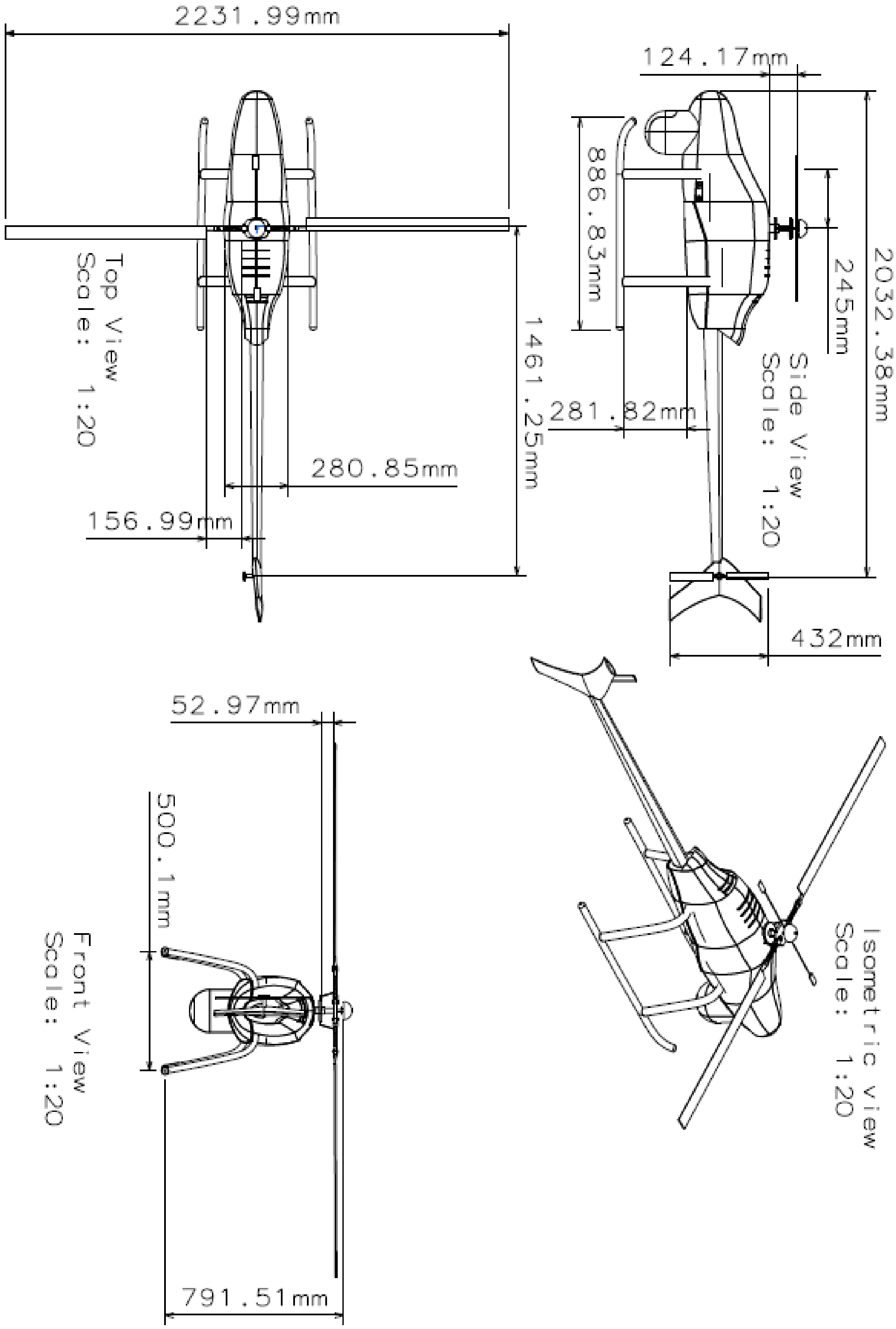

Figure 8.18 : Drafting of design

9. CONCLUSION

Preliminary design just gives an overview to the designer at the first steps of detailed design. The data obtained from here are all approximated values and lots of assumptions such as constant drag coefficients can not be made during the detailed design process. But these assumptions can be accepted as valid for preliminary calculations.

At this work, the systems which will be used in the design was selected by trade-off studies and a rough value of gross weight is approximated at Chapter 2. By the selections have been done here, main rotor and tail rotors are designed with a new gross weight iteration at Chapter 3 and Chapter 4 respectively. Then, total power requirements were obtained and a suitable engine was selected at Chapter 5. At Chapter 6, all rotor design process was repeated and a new concept was constructed. All geometrical specifications of rotors were obtained at there. At Chapter 7, helicopter's performance analysis was made and at Chapter 8, helicopter was drawned with the help of commercial software Catia.

Unmanned helicopters have lots of area of use. They can be used in different mission types. At this work, the mission profile should not be considered so important because at the payload of helicopter improved technology is used. This turret which has seven sensors on it, is capable of doing lots of mission profiles except the missions which needs high maneuverability. The helicopter was designed to achieve high altitudes flights but actually the selected engine does not support above 4000 meters altitudes. At the future works, a more complicated, high performance turboshaft engine should be selected. This gives extra weight to helicopter; but this design is capable of flying with a turboshaft engine and its fuel required with a reduction in mission time.

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CURRICULUM VITAE

Candidate's full name: Aykut Ceyhan

Place and date of birth: İstanbul / 03.09.1981

Permanent Address: Evtaş Blok. O-2 Blok. No:16 Gaziosmanpaşa/İstanbul

Universities attended: İTÜ(Astronautical Engineering)