CITY AIR AMBULANCE

AE 429: Design Lab Report
by

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Certificate

Certified that this Project Report titled **City Air Ambulance** by **Group 6** is approved by me for submission. Certified further that, to the best of my knowledge, the report represents work carried out by the students.

Date: Signature and Name of Guide

Acknowledgment

We are very grateful to our professors, Prof G R Shevare, Prof R K Pant, Prof Avijit Chatterjee, and Prof Hemendra Arya and teaching assistant, Krishna Dutt who have helped us in making our ideas clear and pointing out our mistakes which led us into the right direction.

Chapter 1

Mission Statement

Faster means of transporting a human being or medical facilities for "immediate life threatening condition" with unfavourable constraints in a a metropolitan area.

1.1 Definitions

- **Immediate life threatening condition**
  The focus with life threatening condition refers to the fact that without immediate treatment for the condition, death is imminent and thus the person will be referred as critical patient.

- **Medical facilities**
  It refers to the facilites or setup which is capable of handling the respective life threatening condition and provide basic aid to the patient.

- **Unfavourable constraints**
  By unfavourable conditions, those conditions are being referred where the mode of transportation is inconvenient due to factors like road congestion, bad weather or ongoing construction, etc.

- **Metropolitan area**
  This means an area which is densely populated and has heavy traffic congestion. Thus providing medical facilities within time is a problem here.

1.2 Why this mission statement?

This mission statement is very critical in the fast growing cities where the road automobile density is increasing exponentially. In such a case, providing medical aid to a critical patient within a specified time becomes of eminent importance. Following are some of the facts which emphasise on the importance of providing medical aid within time:

- The chance of surviving a heart attack reduces by 10 per cent with every minute that passes. (Source: *Hans India*)
- In London the median waiting times for Category A (immediate life threatening condition) calls were four minutes 38 seconds but three years on they have reached seven minutes 26 seconds, the figures show. (Source: *Telegraph UK*)
- London is well-equipped with the required ambulance numbers but still many people are dying just due to the delays in the ambulance.
(Source: *Daily Mail UK*)
- Each year in the U.S., there are approximately 359,400 Emergency Medical Services (EMS)- assessed cardiac arrests outside of a hospital setting and on average, less than 10 percent of victims survice. (Source: *Heart USA*)

Chapter 2

Requirements and Specifications

2.1 End User Requirements

Below are the requirements which have been identified as the ones imposed by the critical patient and a basic aid to the critical patient.

- The critical patient shall get quickly transported to the nearest medical facility from their current position in the metropolitan area
- The critical patient shall quickly get required preliminary medical treatment within the metropolitan area.

2.2 Design constraints from Stakeholders

Here are the requirements which are due to various guidelines and regulations set by respective agencies for the operation of Advanced Life Support Ambulance.

- The dimensions of the vehicle has to be at least $2.7m \times 1.5m \times 1.5m$ which are the requirement of Advanced Life Support Ambulance. [1]
- The weight of Pilot and Certified Emergency Vehicle Operations Technician along with their luggage has to be within 70 Kgs.
- The critical patient along with at least one relative (luggage included) has be within (150+100) Kgs.

[1] `http://www.delhi.gov.in/DoIT/Health/cat.pdf`

- All the medical equipment has to be within 110 Kgs which is much more than what is carried in the Advanced Life Support Ambulance. [2]
- The radius of working area has to be within 10 Km.[3]
- The time to reach the critical patient has to be within 8 minutes.[4]

Design Constraint	Value
Length	2.7 m
Width	1.5 m
Height	1.5 m
Payload	500 Kg
Range	20 Km
Flight Duration	less than 8 minutes

Table 2.1: Design Constraints

[2] http://www.delhi.gov.in/DoIT/Health/cat.pdf, we found weight of each necessary equipment and added them with 20 percent reserve weight

[3] Double of the Maximum distance between all major hospitals in Mumbai by Google Maps

[4] http://www.qualitywatch.org.uk/indicator/ambulance-response-times

Chapter 3

Proposed Conceptual Designs

Keeping in the mind the feasibility of the mission, five different conceptual designs had been proposed in order to achieve the mission requirements. Following are the proposed five concepts:

- Helicopter
- Tilt-jet
- Tilt-rotor
- Tilt-wing
- Lift-fan

3.1 Helicopter

For helicopters, there have been some medical helicopters in the history which are extensively used in the countries like Australia and Canada for transporting patients from one city to another [1]. However, their minimum range of operation is 900 km [2] which is far greater than our design range of operation.

3.1.1 General Characteristics

- **Maximum Speed:** 155 knots/287 kmph
- **Max. Cruise Speed:** 137 knots/ 254 kmph

[1] http://www.ambulance.nsw.gov.au/about-us/Emergency-Operations.html
[2] http://airmed.com.au/equipment/

- **Length:** 35 ft
- **Width:** 8.3 ft
- **Height:** 10.8 ft
- **Max. Rate of Climb:** 1670 fpm/ 8.5 mps
  (Source:`https://www.hospitalwing.com/hospwing/aircraft/`)

Figure 3.1: Medical Helicopters

3.2 Tilt-jet

Jet powered aircrafts are faster than any other category of aircrafts. In our mission problem, we need to minimize the response time as the patient is in critical situation and this is the fastest option available. As the name suggests, it can take off vertically and hence does not need any runway distance for takeoff and landing. So, it removes the problem of less space available in metropolitan area. The jets then will propel the aircraft forward. On the downside, it can create a lot a noise and special takeoff pavement will have to be made for it. For reducing noise, Chevron nozzles can be used. A special pavement will have to be made.

3.2.1 General Characteristics (Bell D-188A)

- **Crew:** 1
- **Length:** 62 ft (18.9 m)
- **Height:** 12 ft 9 in (3.89 m)
- **Wingspan:** 23 ft 9 in (7.24 m)

- **Wing Area:** $194ft^2(18.02m^2)$
- **Empty Weight:** 13,800 lbs (6,260 kg)
- **Loaded Weight:** 23,917 lbs (10,849 kg)
- **Powerplant:** 8 × General Electric J85-GE-5 turbojets, 2600 lbf (11.6 kN) each
(Source: `https://en.wikipedia.org/wiki/Bell_D-188A`)

Figure 3.2: Tilt-jet Aircraft

3.3 Tilt-rotor

An ambulance has to take off and land at any place, so VTOL tilt rotor aircraft can be used. It is similar to an helicopter when taking off and landing but after take off the rotor tilts and the rotors now act as propellers and during its cruise it generates lift due to wings and they are the control surfaces as well. Problem might occur due to wing span not making it feasible to land on 2 lane roads.

3.3.1 General Characteristics (Bell XV-3)

- **Length:** 30 ft 4 in (9.2 m)
- **Height:** 13 ft 3 in (4 m)
- **Wingspan:** 31 ft 4 in (9.5 m)

- **Powerplant:** 1 × Pratt & Whitney R-985-AN-1 radial engine, 450 hp (336 kW)
- **Maximum Speed:** 184 mph (296 kmph
- **Cruise Speed:** 167 mph (269 kmph)
- **Range:** 255 miles (411 km)
(Source: `https://en.wikipedia.org/wiki/Bell_XV-3`)

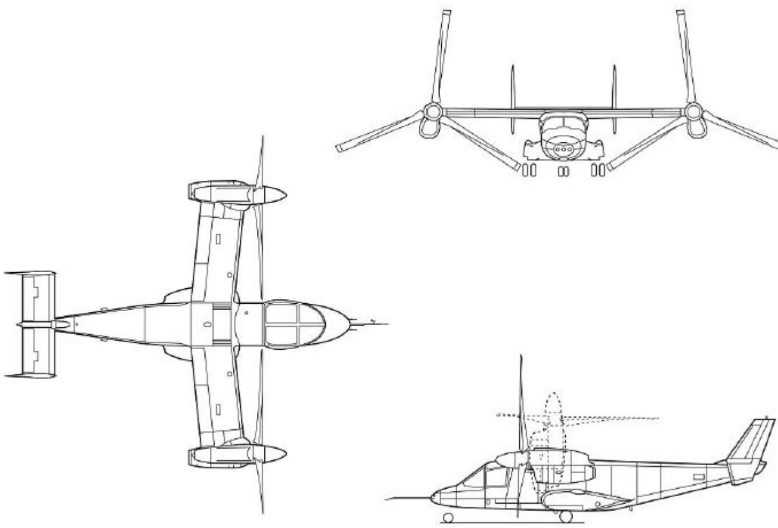

Figure 3.3: Tilt-rotor Aircraft: Bell XV3

3.4 Tilt-wing

A tilt-wing aircraft features a wing that is horizontal for conventional forward flight and rotates up for vertical takeoff and landing. The tilt-wing design offers certain advantages in vertical flight relative to a tiltrotor. Because the slipstream from the rotor strikes the wing on its smallest dimension, the tilt-wing is able to apply more of its engine power to lifting the aircraft. The main drawback of the tilt-wing is control during hover. The wing tilted vertically represents a large surface area for crosswinds to push. Tilt-rotors generally have better hover efficiency than tilt-wings, but less than helicopters.

3.4.1 General Characteristics (Bell XV-3)

- **Length:** 63 ft (19.2 m)
- **Height:** 24 ft 7 in (7.5 m)
- **Wingspan:** 47 ft 11 in (7.5 m)
- **Powerplant:** 2 × Allison T40-A-14 coupled turboprop engines, 5500 hp (4,100 kW) each
- **Maximum Speed:** 253 mph (407 kmph; 220 knots
(Source: `https://en.wikipedia.org/wiki/Tiltwing`)

Figure 3.4: Tilt-wing Aircraft: Hiller X18

3.5 Lift-fan

The aircraft has combination of rotors and propellers or jet engines where VTOL lift is generated by rotor/fans.Here the fans responsible for vertical take-off or landing are located in the later part of the aircraft.

3.5.1 General Characteristics (Bell XV-3)

- **Length:** 44 ft 6 in (13.56 m)
- **Height:** 14 ft 9 in (4.5 m)
- **Wingspan:** 29 ft 10 in (9.09 m)

- **Max. Take-off Weight:** 13,600 lbS (6,169 kg) (conventional take-off), 12,300 lbs (5,579.2 kg) (VTOL)
- **Powerplant:** 2 × General Electric J85-GE-5 turbojet with selectable exhaust driven lift fans, 2,658 lbf (11.82 kN) thrust each horizontal thrust
- **Powerplant:** 2 × General Electric X353-5 62.5 in (1.59 m) diameter tip-drive lift fan, 7,500 lbf (33 kN) thrust each
- **Powerplant:** 1 × General Electric X353-5 36 in (0.91 m) diamter tip-drive lift fan
- **Maximum Speed:** 547 mph (880 kmph; 475 knots
- **Range:** 869 nm (1,609 km; 1000 mi)
(Source: `https://en.wikipedia.org/wiki/Ryan_XV-5_Vertifan`)

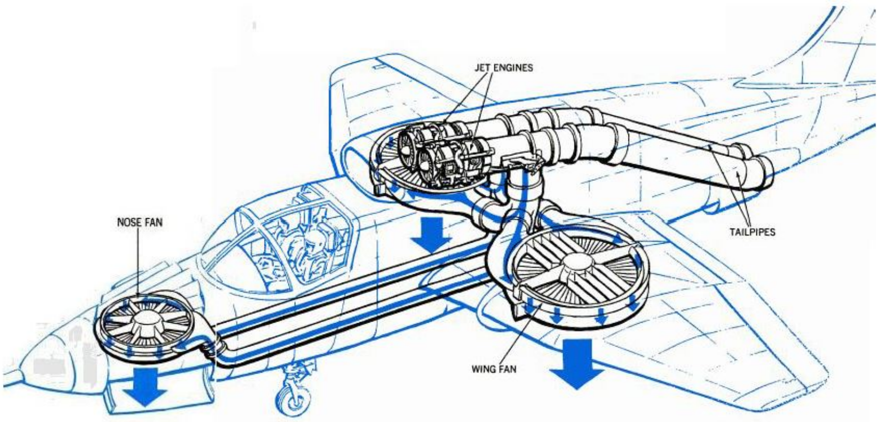

Figure 3.5: Lift-fan Aircraft: Ryan Xv5 Vertifan

3.6 Qualitative Comparison

For qualitative comparison, four vital parameters have been selected for the selection of the aircraft concepts. These are lifted as below:

- Noise: This refers to the noise created by the aircraft in operation. Since the mission requires medical attention, concept with the least noise will be preferred. Thus an aircraft with the most noise will be rated 1 and one with the least will be rated 5.

- Hovering: This refers to the ability of the aircraft to hover at a certain position in the air in the times of need. The aircraft which will be able to perform hovering with ease will be rated 5 and decreasing value of hover from 5 to 1, refers to the decreasing ability for hovering.
- Speed: Since it is a medical requirement mission, we need to serve the patient with the best possible time in transportation. Thus rating for speed will increase from 1 to 5, looking at the aspect if the aircraft can achieve higher speed than others.
- Accessibility: It is the last criterion which was given emphasis. Since we target to make the ambulance service more accessible to the patients, we looked at how easily the corresponding aircraft can be made accessed to a patient in a crowded area. Increasing value of accessibility refers to the better accessibility of the aircraft.

For analysis, one-on-one comparison is done which will lead to giving points to respective concept.

1. Lift Fan vs Helicopter
    - Lift fan is more compact than helicopter, so takes less space
    - Lift fan has less efficiency
    - Lift acts at the center of the fan, so you need additional jets for pitch and roll control. A helicopter can shift the center of lift away from the rotor axis (within limits) by using cyclic changes to the angle of attack of the single rotor blades.
    - Lift fan has higher cruising speed than helicopter
2. Tilt-jet vs Helicopter
    - Tilt-jet produces higher noise than helicopter
    - Tilt-jet has higher cruising speed than helicopter
3. Tilt jet vs Tilt rotor
    - Tilt jet has higher speed than tilt rotor
    - Tilt jet makes higher noise than tilt rotor
4. Tilt jet vs Tilt wing

- Tilt jet has higher cruising speed than tilt wing
- Tilt jet makes more noise than tilt wing

5. Tilt jet vs Lift Fan
    - Lift fan produces less noise as compared to tilt jet
    - Lift fan has better accessibility than tilt jet
6. Tilt wing vs Tilt rotor
    - Cyclic pitch (tilt rotor) vs simple pitch (tilt wing) will give a much greater degree of control during hover, and a greater latitude in dealing with less than perfect CG issues
    - Tilt rotor has higher cruising speed than tilt wing
7. Tilt rotor vs Helicopter
    - Tilt rotor has higher cruising speed than helicopter
    - Tilt rotor has lesser noise as compared to helicopter
8. Tilt rotor vs Lift Fan
    - Tilt rotor makes lesser noise than lift fan
    - Tilt rotor has higher cruising speed than lift fan
9. Tilt wing vs Helicopter
    - The hovering capability of tilt wing is lesser than the helicopter
    - Helicopter has better accessibility than tilt wing
10. Lift fan vs Helicopter
    - Hovering capability of lift fan is lesser than the helicopter
    - Accessibility of helicopter is better than lift fan

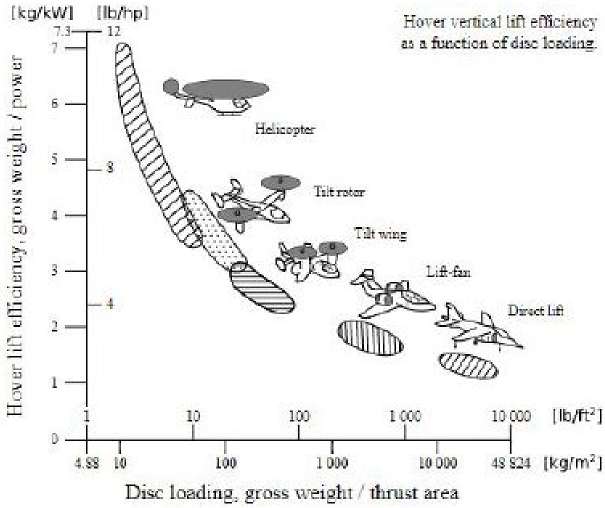

Figure 3.6: Comparison between different concepts for hovering , gross weight/power.

1

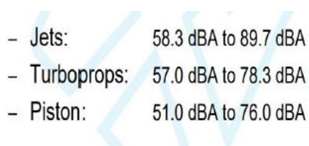

Figure 3.7: Comparison between different concepts for noise

Name	**Noise**	**Hovering**	**Speed**	**Accessibilty**	**Total**
Helicopter	2	5	3	4	14
Tilt jet	1	1	5	1	8
Tilt rotor	3	4	4	4	15
Tilt wing	3	3	3	2	11
Lift Fan	2	2	5	2	11

Table 3.1: Qualitative Comparison

Thus we are finally selecting **Tilt-rotor aircraft** since it seems to have the most suitable criterion satisfaction.

Chapter 4

Subsystems Identification

After discussions and reviews, many subsystems were identified which will be the part of the aircraft system once it has developed completely. Following is the list of the subsystem which we will need for the aircraft:

- **Powerplant & Auxiliary Power Unit (APU)**
  The aircraft should be fitted with turboshaft engines which will be used to drive the shaft for porpeller/rotor rotation.Also an auxiliary power unit is thought to be placed in the aircraft for starting the engines while taking off.

- **Flight Control Systems**
  For the flight control systems, basic control surfaces have been ideated to be installed. Also the connecting linkages have thought to be of hydraulic linkages.

- **Landing Gear**
  The aircraft is planned to be fitted with a non-retractable tricycle-wheel landing gear due to short range requirement. It will be fitted with outriggers on the centre fuselage. Also independent oil struts should be installed on four points to absorb the landing loads.

- **Battery/Fuel**
  Fuel feed systems and fuel storage tanks will be required to drive the fuel from tanks to the engnine and then run the engine respectively. However the type of feed systems can either be pump fed or compressed gas fed. However further fuel/gas storage tanks will be required for the feed systems.

- **Electrical Systems**
  To power various components like lights, flight instruments, navigation aids

or radios, electrical system is a must part of the aircraft. All the electrical components are desired to be connected to bus-bar using wires which will be further incorporated with circuit breakers or fuses. For an extra power source in the aircraft, battery is thought to be implemented to accomplish the task for assistance in engine or power the electrical system whilst the engine is not running.

- **Avionics, Navigation & communication**
  This subsystem is thought to be composed of flight control system and computer, radio altimeter, air data computer, electrical system and payload interfaces. The flight control system integrates redundant sensors, processing elements, drive electronics, wiring and actuator motors etc.

- **Flight Instrumentation & Recording**
  For instrumentation, traditional flight instruments will be installed which includes altimeter, vertical speed indicator, airspeed indicator, attitude indicator and heading indicator. Also the aircraft will be equipped with flight data recorder (FDR) and cockpit voice recorder (CVR).

- **Environment Control Systems**
  Since the aircraft is for medical needs, the altitudes which will require air pressurization will be avoided. And hence air pressurization system will not be required. However, thermal control will be needed as the temperature will be very low at high altitudes and also due to seasons variation in a country like India.

- **Hydraulics/Electrical Actuation System**
  For actuation systems, hydraulic might be needed for obtaining the tilting mechanisms whereas electrical system will be required for basic control surfaces.

- **Emergency & Warning Systems**
  There are two aspects in this subsystem. First will be referred as 'Prevention' which includes alarm & warning system, fire detection & suppression system, fuel system explosion & suppression, and emergency power generation. The second aspect is referred as 'Protection'. It includes anti-crash seats and interiors, and occupant ejection.

Chapter 5

Mission Description

Our mission profile is supposed to be a 7-segment operation which are take-off/climb, transition, accelerate, cruise, decelerate, transition, and landing/descent which are further described here.

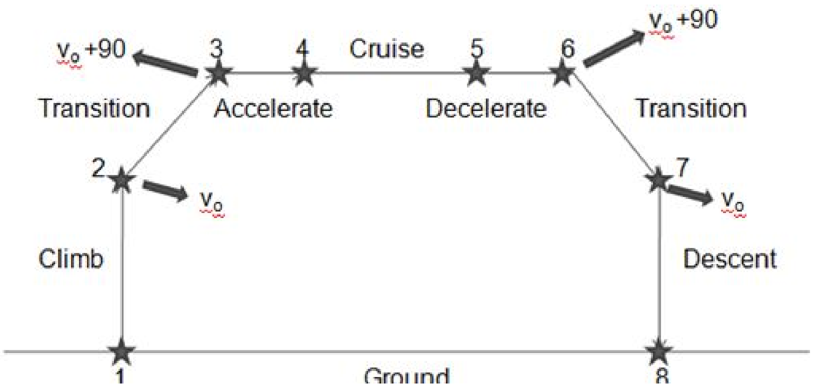

Figure 5.1: Proposed Mission Profile

5.1 Climb/Descent Phase

For climb/descent phase, we will need to provide great thrust to lift the aircraft from the surface and thus acceleration will be applied. Keeping in mind the constraint of medical aircraft, constant acceleration has been thought due to the criticality of the patient. So the condition here is that the critical patient should not feel any jerk. Thus a maximum acceleration of $0.93m/s^2$ has been formulated after inspecting ground ambulance parameters.

5.2 Transition Phase

Transition phase [1] is assumed to have quasi-static transition which means that for a 2.5^o change in the tilting angle of the rotor in transition, the velocity of the aircraft will also change by 2.5 m/s. This can be put in equation as below:

$$\frac{d\theta}{dt} = 2.5^o \eta$$

$$\frac{dv}{dt} = 2.5\eta; \eta = \frac{ds}{dt}$$

where, s = step size
Also, for $\theta = 0$ to $90^o, s = 36.$[2]

Thus here the velocity of the aircraft will increase by 90 m/s after the transition in leg 2-3. Also the speed of the aircraft must be the same before descent transition.

5.3 Accelerate, Cruise and Decelerate

Since our mission objective is to minimise the time of flight for a range of 20 km, we need to travel at faster speeds. Hence to accomplish this task, we need to achieve a higher velocity in cruise and thus will be required to accelerate to a higher velocity. Also to reach the velocity for transition while descent, we will need to decelerate.

[1] `https://www.iitk.ac.in/aero/abhishek/files/ARF_2015_coaxial.pdf`

[2] ***Note:** From here onwards, every quantity is in SI units unless specified.

Chapter 6

Design Methodology

For the design of the aircraft, we have started with the kinematic optimisation which includes the mission profile optimisation leading to the dynamic optimisation including the design of various aircrafts components. Below is the flow chart which was used while designing phase.

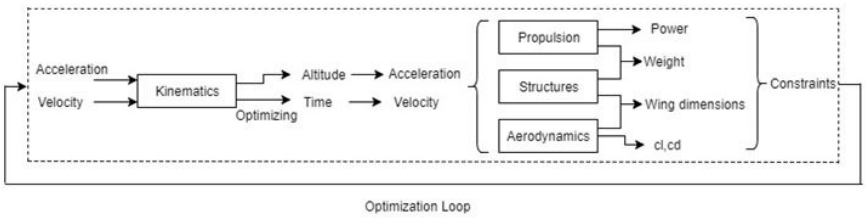

Figure 6.1: Design Flow-chart

6.1 Design Variables

Following is the list of the design variables which will be required to design the aircraft. The focus is kept on obtaining vital parameters which will help in the design further.

- Lift-by-Drag (L/D)
- Aspect Ratio (AR)
- Wing span (b)
- Sweep angle (Λ)
- Taper ratio (λ)

- Wing position

For 1st iteration,parameters like sweep angle has been kept 0 and taper ratio has been kept 1 such that the aircraft will be rectangular.

6.2 Bought Out Items

- Engine
- Propeller
- Nacelle

6.3 Design Constraints

These will be the quantities which might be changed if the need be in further iteration to accomplish a better design.

- Cruise Height, H = 2500 ft (based on regulations)
- Height after leg 1-2, h <= 1300 ft (due to building clearance height)
- Cruise velocity, $v_{cruise} < 231.42$ m/s (=0.7 Mach number, due to Mach Tuck)
- Take off velocity, $v_{to} < 50$ m/s
- Wing span, b < 8.5 m [1]
- Acceleration for leg 1-2, a < 0.93 [2] m/s^2 [Varon J [1997]]

6.4 Design Parameters

- Engine, quantity = 1
- Propeller, quantity = 2
- Nacelle, quantity = 2

[1] (b+3)*1.3 should be less than 15 m which serves as width of the lanes of wider roads in Mumbai. Here 3 is the diameter of the baseline propeller and 1.3 is the landing factor for light aircrafts based on regulations.

[2] https://www.emsworld.com/article/10319229/acceleration-forces

Chapter 7

Preliminary Design: I Iteration

7.1 Fuselage Dimensions

Using the parameters of baseline aircraft, the length of the fuselage was chosen to be approx. 9 m. Also since the minimum design width and height constraints are 1.5 m each, an additional spacing of approx. 0.2 m has been left for various components like fuselage hull, spars, etc on each side for both width and height. Moreover, the structural material for the fuselage has been chosen to be an aluminium alloy. However, the grade of the alloy has not been fixed.

Dimension	**Value**
Length	9 m
Width	1.9 m
Height	1.9 m

Table 7.1: Fuselage Dimensions

7.2 Kinematic Optimisation

In this chapter, we will deal with the various considerations taken to achieve the design specifications. Here attempt is made to arrive at an optimisation function which will optimise the time of flight. With this kinematic optimisation, we will step into our premiliminary design stage for 1st iteration.

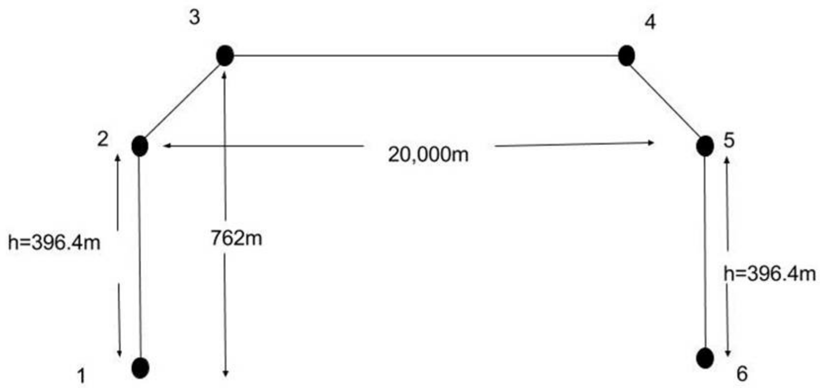

Figure 7.1: 1st Iteration: Kinematic Mission Profile

7.2.1 Assumptions

- Leg 1-2 (Climb): Constant Acceleration (a, in m/s^2)

$$0 <= a <= 0.93$$

- Leg 2-3 (Transition): Quasi Static Transition (h, in m)

$$396.4 <= h <= 762$$

7.2.2 Formulation

Initial formulation has been done for the climb and transition phase, assuming the the transition and descent phase are identical with each other. For climb, we know that the height achieved will be h.
For transition phase, height is as follows:

$$h_{transition} = \int_0^{t_t} (2.5\eta t + \sqrt{2ha}) cos(2.5 \times \frac{\pi}{180}\eta t) dt$$

where, h: Height attained for leg 1-2
a: Acceleration during leg 1-2
t_t = Time in transition leg ; $\eta = \frac{36}{t_{transition}}$
H = 762 m, Cruise height

After integrating,

$$H = t_t[0.900316\sqrt{ah} + 20.8202] + h$$

Also,

$$h = 0.5 \times a \times t_{1-2}^2$$

$$t_{total} = t_1 + t_{1-2}$$

Thus,

$$t_{total} = \sqrt{\frac{2h}{a}} + \frac{H}{0.900316\sqrt{ah} + 20.8202 + h}$$

Above obtained function for total time is the optimization function for leg 1-3.

7.2.3 Results

After substituting the above function in optimizer, following were the results obtained for kinematic analysis.

Cruise Height	762 m (2500 ft)
Climb Height	396.4 m (~1300 ft)
Range covered after leg 1-3	516 m
Time for leg 1-2	29.18 s
Time for leg 2-3	9.6 s
Total time for leg 1-3	38.79 s
Acceleration for leg 1-2	0.93 m/s^2
Velocity after leg 1-2	27.28 m/s
Velocity after leg 2-3	117.28 m/s

Table 7.2: Kinematic Results for leg 1-3

Since the identical profile is assumed in the ascent and descent phase till cruise is achieved, the above obtained results for leg 1-3 will be same for leg 6-8.

Range covered after leg 6-8	516 m
Time for leg 6-7	9.6 s
Time for leg 7-8	29.18 s
Total time for leg 6-8	38.79 s
Acceleration for leg 7-8	0.93 m/s^2
Velocity after leg 7-8	27.28 m/s
Velocity after leg 6-7	117.28 m/s

Table 7.3: Kinematic Results for leg 6-8

Using the above results, we will derive the parameters for acceleration and cruise leg. And the results of the acceleration will then be same as for the deceleration leg.

7.3 Dynamic optimisation

After obtaining the range covered and time taken after transition leg, we need to determine the mission profile parameters for cruise leg. Here, we will again try to design an objective function for time which will be required to optimise. This objective function will relate the time with the dynamic parameters like L/D, or P/W with the help of cruise velocity.

7.3.1 Assumptions

While doing dynamic analysis, the gross take-off weight of the aircraft have to be worked out. For this, we opted to refer to our baseline aircraft which is a commercial aircraft named Elytron [1]. Using our baseline aircraft, we came to following assumptions while estimating the gross weight.

- Structural weight of the aircraft $\approx 0.5W_0$, where, $W_o = W_{structure} + W_{payload} + W_{fuel} + W_{engine}$
- Weight of the payload = 500 kg (design constraint)
- Neglecting the weight of the fuel when compared to structural or payload weight because of the small mission range. Weight of the fuel estimated in

[1] https://www.nextbigfuture.com/2016/10/vertical-takeoff-and-landing-boxwing.html

one mission is around 12.5 kgs. The total fuel mass needed in a single to-and-fro mission is ~ 25 kgs which is only 2% of aircraft gross weight. [1]

- Hence, $W_o = 2W_{engine} + 1000$

7.3.2 Formulation

The objective function for the cruise is obtained by subtracting the range covered in legs 1-3 and legs 6-8 from the total range and subtracting the time taken in the respective legs from 8 min due to the design constraints.

$$t = \frac{range - x_{1-3} - x_{6-8}}{v_{cruise}}$$

where, $t <= 8$ min $-t_{1-3}$ $-t_{6-8}$

$$v_{cruise} = (\frac{L}{D})_{max}(\frac{P}{W})_{max}$$

$$Range = 20km$$

For the above formulation, v_{cruise} is considered directly. If acceleration and deceleration are considered in the horizontal flight, time was 129.5 sec and total time was 207.08 sec. If no acceleration is considered, time for horizontal leg is 170.53 sec and total time is 248.1 sec. Here, reduction in time is just 16% and also there will be less drag, less fuel and no jerk in no acceleration case. Hence we choose zero acceleration in horizontal leg. After the above formulation, max. L/D was found to be a function of aspect ratio (AR) as given below.

$$(\frac{L}{D}) = f(AR) = \frac{1}{2}\sqrt{\frac{\pi\{1.78(1-0.04AR^{0.68}) - 0.64\}AR}{c_{Do}}}$$

Using above assumptions for weight and L/D, the final objective function is:

$$t = \frac{20 \times 10^3 - 2 \times 516}{(\frac{P}{2W_o+1000})(\frac{1}{2}\sqrt{\frac{\pi\{1.78(1-0.04AR^{0.68})-0.64\}AR}{c_{Do}}})}$$

where, C_{D_o} has been estimated to be 0.023 [2] from similar class of aircraft. In order to keep the time minimum, P/W ratio has to be maximised along with the L/D. This formulation of t has been achieved by equating lift with weight and thrust with drag in cruise phase.

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[1] $W_{fuel} = \eta * (SFC) * P * t = (0.8 * 0.8 * 0.33 * 418 * 8 * 60/3600) = 11.77kgs$

[2] `http://s6.aeromech.usyd.edu.au/aerodynamics/index.php/sample-page/aircraft-performance/drag-and-drag-coefficient/`

7.3.3 Powerplant Selection

As we discussed about maximising the P/W ratio, we looked at various turbo-shaft engines and studied their several properties to select the most suitable engine for our aircraft. Here, while calculating P/W_{total}, gross weight of the aircraft was used to find the ratio with the help of assumptions set in previous section.

Name	P/W (kW/kg)	P (kW)	Weight (kg)	Length (m)
PW 200	3.9456	418	107.5	0.912
RR 300	2.4390	220	91	0.96
RR 500	3.4720	354	102	1.09
RR 250	3.0020	190	62	1.029
Boeing T50	2.5230	246.08	98	0.945
Boeing T60	3.6160	410.13	113	1.374
GTD -350	2.2	298	135	1.35
Turbomeca Artouste	3.24	372.85	115	1.440

Table 7.4: Powerplant Selection 1

Name	Diameter (m)	Volume (m)	P/Weight (kW/kg)
PW 200	0.5	0.1791	0.1983
RR 300	0.55	0.2281	0.1052
RR 500	0.59	0.2980	0.1684
RR 250	0.572	0.2644	0.0921
Boeing T50	0.572	0.2428	0.1173
Boeing T60	0.639	0.4406	0.1941
GTD -350	0.522	0.2889	0.1396
Turbomeca Artouste	0.545	0.3359	0.1763

Table 7.5: Powerplant Selection 2

From the above table, we have selected Pratt & Whitney PW-200 for our aircraft since we are going with the idea of maximising P/W ratio for the aircraft. For this, the diameter of the propeller was chosen to be 10 ft.

7.3.4 Aifoil Selection

For the selection of airfoil, we studied various airfoils for their $(c_l/c_d)_{max}$ value around $\alpha = 0^o$. Our Reynolds number is of the order of 10^7 but the data is available till 10^6. And for higher Reynolds number, c_l and c_d for this class of airfoil tends to be same at zero angle of attack.

Airfoil	Alpha	Cl	Cd	Re	Max. Chamber	Max. Thickness	Cl/Cd
NACA 1408	0	0.115	0.0039	1000000	0.01	0.08	29.11167513
NACA 1410	0	0.121	0.0049	1000000	0.01	0.1	24.70468432
NACA 1412	0	0.121	0.0054	1000000	0.01	0.12	22.22426471
NACA 23012	0	0.124	0.0059	1000000	0.15	0.12	21.14139693
NACA 23015	0	0.132	0.0073	1000000	0.15	0.15	18.16551724

Table 7.6: Airfoil Selection

Using the above data, NACA 1408 has been selected since it has the maximum (c_l/c_d) at $\alpha = 0$. This criteria is keeping in mind of the fact that for most of the horizontal flight, aircraft will fly around $\alpha = 0$ approximately.

7.3.5 Angle of Setting

To find the angle of setting, the following formulas have been used.

$$a = \frac{2\pi AR}{2 + \sqrt{\frac{AR^2(1-m^2)}{k^2}[1 + \frac{tan^2\Lambda_{1/2}}{1-m^2}] + 4}}$$

[3]

$$C_{l_{required}} = C_{l_o} + a * \alpha$$

$$\alpha = \frac{C_{l_{required}} - C_{l_o}}{a}$$

[Jr. [2016]]

[3] http://www.dept.aoe.vt.edu/~lutze/AOE3104/airfoilwings.pdf

where, k = $a_o/2\pi$
a_o = Actual 2D lift curve slope
$\Lambda_{1/2}$ = Wing sweep angle at c/2

7.3.6 Wing Deflection

We will require that the tip deflection/span (δ/b) not exceeds some reasonable upper limit which is usually less than 0.05 in 1G flight. Also the material is chosen to be aluminium. Hence we get the following expressions. [4]

$$\frac{\delta}{b} = 0.018\frac{(W_{fuse} + W_{pay})}{E\tau(E^2 + \epsilon^2)}(1+\lambda)^3(1+2\lambda)\frac{AR^3}{s}$$
$$\tau = \frac{t}{c}; \epsilon = \frac{max.camber}{c}; \frac{\delta}{b} \leq 0.01$$
$$W_{fuse} + W_{pay} = W_o - W_{wing} = 0.9W_o{}^5$$

For our airfoil, in this iteration, we get the upper limit on aspect ratio to be 12.

7.3.7 Analysis

Using the above formulations, wingspan (b) was chosen to be 8.5 m. Now by varying the aspect ratio (AR) of the wing, different parameters were studied in order to arrive at the dimensions for the first iteration of the preliminary design. Below table gives an insight into the study of these parameters.

AR	c	Power(kW)	W/S	P/W (kW/kg)	Setting angle (^{o})	t
5	1.7	306.8	830	25.5	-0.07	0.1768
6	1.42	258	996.95	21.47	0.20	0.14768
7	1.2	223	1163	18.58	0.44	0.1248
8	1.06	197	1329	16.4	0.67	0.11024
9	0.94	177	1495	14.7	0.88	0.09776
10	0.85	161	1661	13.4	1.09	0.0884
11	0.77	148	1827	12.3	1.30	0.08008
12	0.7	137	1993	11.4	1.50	0.0728

Table 7.7: Dynamic Analysis 1

[4] `http://web.mit.edu/16.unified/www/SPRING/systems/Lab_Notes/desvar2.pdf`
[5] Wing weight is assumed to be 10% of the MTOW.

AR	Cd	Cl	Cl/Cd	P/e_fac*d_fac
5	0.0238	0.1092	4.588235	503.152
6	0.024	0.131	5.458333	423.12
7	0.0243	0.1528	6.288066	365.72
8	0.0245	0.1746	7.126531	323.08
9	0.0247	0.1965	7.955466	290.28
10	0.025	0.2183	8.732	264.04
11	0.0253	0.2401	9.490119	242.72
12	0.0256	0.262	10.23438	224.68

Table 7.8: Dynamic Analysis 2

Using the above data, powerplant, and airfoil selection, the power supplied by PW200 is optimum for aspect ratio 8 by keeping the reserves of 20% of maximum power. With the help of this aspect ratio value, the chord was found to be of 1.06 m in length.

7.3.8 Results

By comprehension of all the above data, we can say that the final results of the first preliminary design are as follows.

1st Iteration	
Engine	PW 200
Airfoil	NACA 1408
Wingspan, b	8.5 m
Aspect Ratio	8
Chord, c	1.06 m

Table 7.9: Iteration 1: Dynamic Analysis Results

7.3.9 Take-off & Landing Phase

Using below formulas, the power variation was calculated for both take-off and landing. We use counter-rotating propellers for stability and we make sure that center of gravity will be in line with the propellers by the optimal placement of the engine. The following formulation is for take-off. Similar formulation can be done for landing with negative acceleration and initial velocity.

$$T = ma + mg + \frac{1}{2}\rho V^2 c_D$$
$$P = T * V$$
$$T_{static} = P^{2/3}(2\rho A)^{1/3}$$
$$\rho = \frac{\rho_o + \rho_h}{2}$$
$$m \sim 1200kg; g = 9.8m/s^2; c_D = 2; a = 0.93m/s^2$$
$$Velocity = u + a * t; \rho_o = 1.225; \rho_h = 1.112; Diameter = 3.048m; A = \frac{\pi d^2}{2}$$

Here, T = Thrust required
P = Power required
V = Velocity of the aircraft
a = Acceleration of the aircraft
m = Mass of the aircraft
ρ = Density of the air
c_D = Drag coefficient

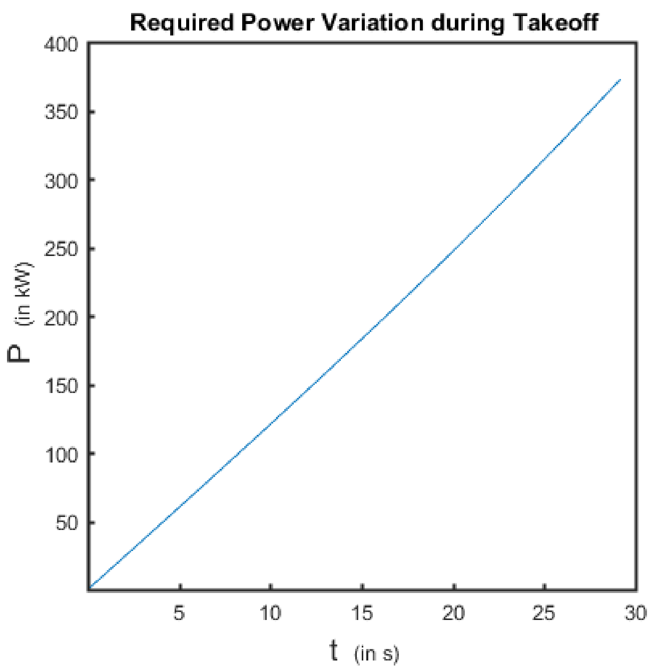

Figure 7.2: 1st iteration: Power Variation during Take-off

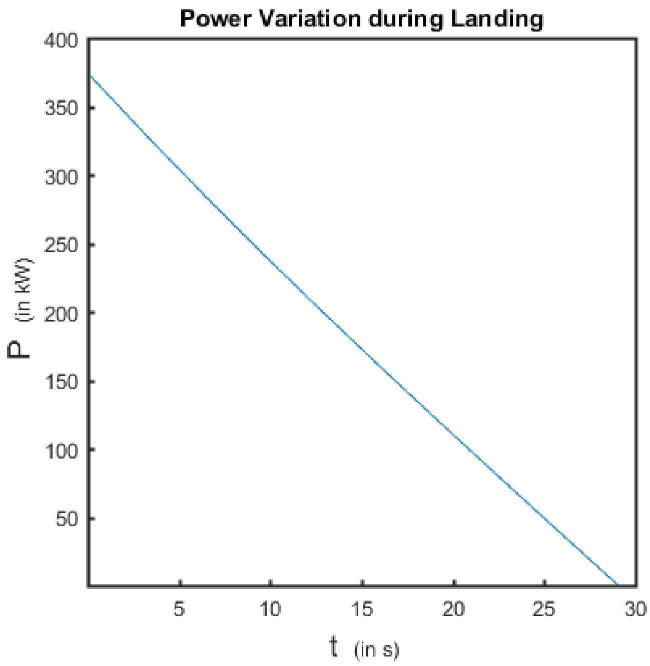

Figure 7.3: 1st iteration: Power Variation during landing

7.3.10 Transition Phase

Transition phase is for a very short time period (≈ 9 sec). So power requirement of transition phase was not studied during this iteration. The free body diagram of the aircraft has been studied, so the thrust and the velocity were made to not align for the thrust to counter the weight as the lift is very minute. However the external power required to make that non-alignment is not considered into account with in the scope of the project.

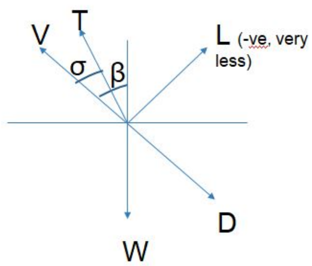

Figure 7.4: Transition phase: Free Body Diagram

7.4 Horizontal Tail Sizing

Using conventional data for turboprop aircrafts, aspect ratio was chosen to be 4 and static margin was selected as 20. Also the tail is assumed to be rectangular. Using below formulas [Raymer [1989]]:

$$\bar{X}_{np} = \frac{C_{L\alpha}\bar{X}_{ac,w} - C_{m_{\alpha,fus}} + \eta_h \frac{S_h}{S_w} C_{L_{\alpha_h}} \frac{\partial \alpha_h}{\partial \alpha} \bar{X}_{ach} + \frac{F_{p\alpha}}{qS_w}\frac{\partial \alpha_p}{\partial \alpha}\bar{X}_p}{C_{L\alpha} + \eta_h \frac{S_h}{S_w} C_{L_{\alpha_h}} \frac{\partial \alpha_h}{\partial \alpha} + \frac{F_{p\alpha}}{qS_w}\frac{\partial \alpha_p}{\partial \alpha}}$$

7.4.1 Results

Aspect Ratio	4
Static Margin	20
Area	1.34201
Chord	0.579
Span	2.317

Table 7.10: Horizontal Tail: 1st iteration

7.5 Vertical Tail Sizing

For vertical tail, using conventional data, aspect ratio was selected as 1.55. All other quantities have been selected using below formulas [Nelson [1998]] [6]. Initially anhedral is chosen to be as 0 deg.

$$C_N = \frac{N}{Q_w S b} = l_v C_{L_{\sigma_v}}(\beta + \sigma) Q_v S_v$$

$$\eta_v(1 + \frac{d\sigma}{d\beta}) = 0.724 + 3.06\frac{S_t/S}{1 + cos\Lambda} + 0.4\frac{z_w}{d} + 0.009AR_w$$

$$N_\beta = N_\beta^{wing} + N_\beta^{tail}$$

$$= \frac{1}{2}\rho V^2 S b C_{L\alpha}(\frac{S_v(l_t - x_{AC})}{Sb} - \frac{x_{AC}}{2b} sin^2\Gamma)$$

[6] https://www.aero.iitb.ac.in/~paranjape/courses.html

7.5.1 Results

Anhedral	0
Aspect Ratio	1.55
Area	1.279
Chord	0.908
Span	1.408
Cn_beta	0.289
High Wing	Yes

Table 7.11: Vertical Tail: 1st iteration

7.6 Landing Gear

The landing gear dimensions and types are available in standard forms. So for our weight of the aircraft, we looked up source and following dimensions had been found out. For weight distribution, 20% of the total aircraft weight have been assumed on front landing gear and 80% on the main landing gear. [Raymer [1989]]

Outside Diameter	33.02 cm (13 in)
Rim Diameter	10.6 cm (4 in)
Rim Width	12 .7 cm (5 in)
Rated Load (Nose, 5 plies)	544. 3 kg
Rated Load (Main LG, 12 plies)	997.9 kg

Table 7.12: Landing Gear

Chapter 8

Preliminary Design: II Iteration

Since the results obtained from kinematic analysis in first preliminary design iteration were exceeding the acceleration values in the transition leg, it has been revisited in the second iteration.

8.1 Kinematic Analysis

Here, the assumptions and formulations made in the first iteration have been kept same. But there is an added constraint of staying in the acceleration limits while transition phase too. This can be mathematically represented as:

$$0 \leq a_{transition} \leq 0.93 m/s^2$$

8.1.1 Results

After adding the above constraint in the previously made optimizer function, following results were obtained. Here the design constraint of cruise height of 2500 ft had to be relaxed in order to keep the patient safe from any kind of jerk.

Cruise Height	4083.66 m (13400 ft)
Climb Height	396.4 m (∼1300 ft)
Range covered after leg 1-3	890 m
Time for leg 1-2	29.18 s
Time for leg 2-3	96.77 s
Total time for leg 1-3	125.96 s
Acceleration for leg 1-2	0.93 m/s^2
Velocity after leg 1-2	27.15 m/s
Velocity after leg 2-3	117.15 m/s

Table 8.1: Kinematic Results: II Iteration for leg 1-3

Range covered after leg 6-8	890 m
Time for leg 6-7	96.77 s
Time for leg 7-8	29.18 s
Total time for leg 6-8	125.96 s
Acceleration for leg 7-8	0.93 m/s^2
Velocity after leg 7-8	27.15 m/s
Velocity after leg 6-7	117.15 m/s

Table 8.2: II Iteration: Kinematic Results for leg 6-8

Using the above data, we can see that the design constraint set on the cruise height have been violated and hence this will require pressurization. Since pressurization is not preferred at all, the kinematic analysis is done again by redefining the assumptions.

8.1.2 Re-Analysis

For doing the kinematic analysis again, below are the assumptions modified to find a better solution satisfying the design constraints for the transition phase.

$$0 \leq a_{horizontal} \leq 0.93 m/s^2$$

$0 \leq a_{vertical} \leq 1.6$[1,2]$m/s^2$

The horizontal acceleration is set due to medical conditions like black-out or red-out. And the vertical acceleration is set using data for different aircrafts.

[1] https://www.quora.com/What-is-the-typical-vertical-acceleration-of-a-commercial-jet-during-ta

[2] https://arxiv.org/ftp/arxiv/papers/1701/1701.04126.pdf

8.1.3 Results

The above formulations were added in the optimizer function built initially and the below results were obtained.

Cruise Height	2400 m (~7900 ft)
Climb Height	396.4 m (~1300 ft)
Range covered after leg 1-3	2051.75 m
Time for leg 1-2	53.5 s
Time for leg 2-3	56.3 s
Total time for leg 1-3	109.8 s
Acceleration for leg 1-2	0.2765 m/s^2
Acceleration for leg 2-3 (hor.)	0.93 m/s^2
Acceleration for leg 2-3 (ver.)	1.6 m/s^2
Velocity after leg 1-2	14.8 m/s
Velocity after leg 2-3	104.8 m/s

Table 8.3: Reiterated Kinematic Results for leg 1-3

Using the same results for leg 6-8, below is the compiled table.

Range covered after leg 6-8	2051.75 m
Time for leg 6-7	56.3 s
Time for leg 7-8	53.5 s
Total time for leg 6-8	109.8 s
Acceleration for leg 7-8	0.2765 m/s^2
Acceleration for leg 6-7 (hor.)	0.93 m/s^2
Acceleration for leg 6-7 (ver.)	1.6 m/s^2
Velocity after leg 7-8	14.8 m/s
Velocity after leg 6-7	104.8 m/s

Table 8.4: Reiterated Kinematic Results for leg 6-8

Since above satisfies all the criterion, these has chosen to be the final kinematic analysis which will then be used to arrive at different dynamic results using low fidelity (hand calculations) and high fidelity (simulations) calculations.

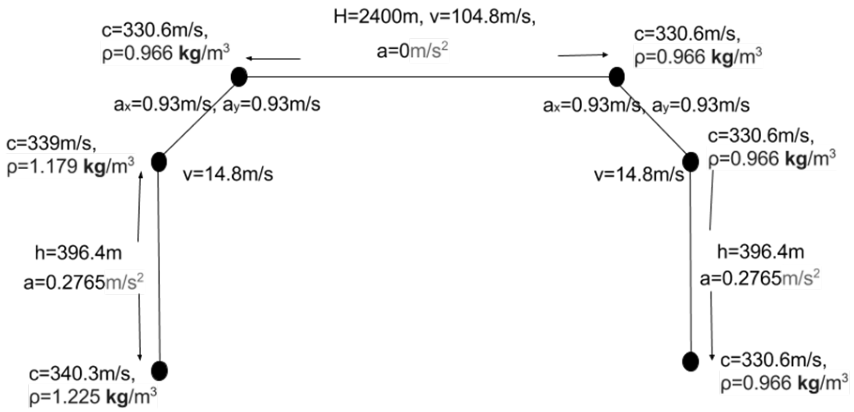

Figure 8.1: 2nd Iteration: Kinematic Analysis Final Results

8.2 Dynamic Optimisation

Using the reiterated kinematic analysis results obtained in previous section, all the parameter like power-to-weight ratio, W/S, etc. are studied again for the most optimum results for the aircraft. These results will then be analysed using high fidelity simulation softwares like XFLR5. Also the angle of setting and tip deflection theory has been used same as in first iteration. But the taper ratio is taken to be 0.4 [3] which is usually taken for our class of aircraft. Also the airfoil and power-plant is continued for further iterations. From the tip deflection, we get a limit of aspect ratio to be 16.

[3] http://nptel.ac.in/courses/101106035/035_Chapter%206_L26_(04-10-2013).pdf

8.2.1 Dynamic Analysis

AR	c	Power(kW)	W/S (kg/m^2)	P/W (kW/kg)	Cl	Cd	Cl/Cd
5	1.7	195	830	16.2	0.1589	0.0248	6.407258
6	1.42	165	996.95	13.7	0.1907	0.0252	7.56746
7	1.2	144	1163	12	0.2225	0.0257	8.657588
8	1.06	128.5	1329	10.7	0.2543	0.026	9.780769
9	0.94	116.5	1495	9.7	0.2861	0.0267	10.71536
10	0.85	107.03	1661	8.9	0.3179	0.0273	11.64469
11	0.77	99.4	1827	8.2	0.3496	0.0278	12.57554
12	0.7	93.12	1993	7.76	0.3814	0.0285	13.38246

Table 8.5: II Iteration: Dynamic Analysis

AR	Setting Angle (in ^{o})	P/efficiency*d_{fac}
5	0.42365	343.0171875
6	0.7039	290.2453125
7	0.9767	253.305
8	1.2463	226.0395313
9	1.5144	204.9307813
10	1.7819	188.2724594
11	2.0482	174.8508125
12	2.3155	163.8039

Table 8.6: II Iteration: Dynamic Analysis Results

As we can see that the power required for different aspect ratios have decresed since last iteration. Again, we will chose the aspect ratio which supplies us enough power keeping in mind the reserves of 20 % of maximum pressure. Since selecting aspect ratio would result in lower reserves percentage, aspect ratio 6 has been chosen for optimum design of the aircraft in accordance with the selected power-plant.

8.2.2 Results

Below are the results for the second iteration of preliminary design.

1st Iteration	
Wingspan, b	8.5 m
Aspect Ratio	6
Chord, c	1.42 m
Taper Ratio	0.4
Cruise Velocity	104.8 m/s
Cruise Height	2400 m

Table 8.7: Iteration 2: Dynamic Analysis Results

8.2.3 Take-off & Landing Phase

Same theory has been used as in previous iteration to obtain required power variation graphs.

$$\rho_h = 0.9996kg/m^3; a = 0.28m/s^2$$

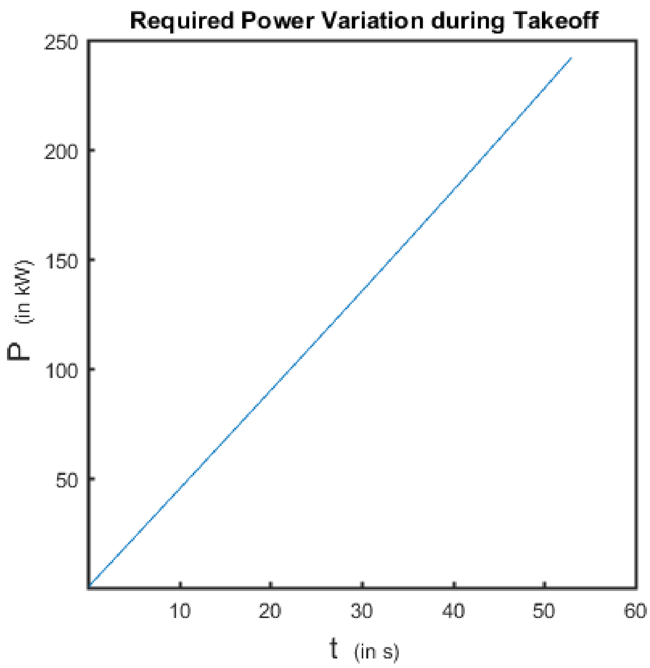

Figure 8.2: 2nd iteration: Power Variation during Take-off

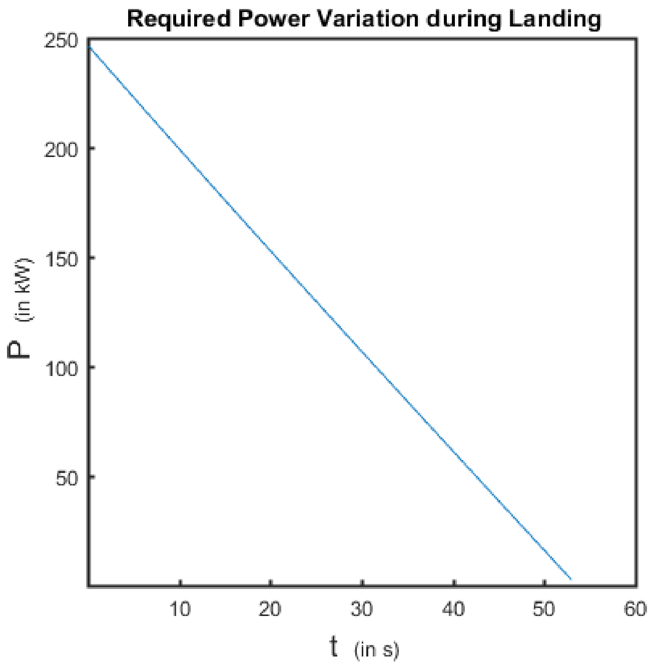

Figure 8.3: 2nd iteration: Power Variation during landing

8.2.4 Transition Phase

Transition is not affecting the wing or tail design as it doesn't contribute aerodynamically since flow over the wing or tail is not streamlined in this phase. Hence after the cruise analysis in third iteration, the power requirement will be checked for this phase.

8.3 Wing Weight Estimation

For initial determination of the wing weight [4], an empirical formula had been used using a reference paper. Below is the formula (in fps system) used to estimate the wing weight.

$$W_w = 0.000912 k_{no} k_\lambda k_e k_{uc} \times [k_b n_w (W_{des} - 0.8 W_w)]^{0.55} \frac{b^{1.675}}{\tau_r^{0.45} (cos \Lambda_{1/2})^{1.325}} + W_{hld}$$

$$k_{no} = 1 + \frac{2.5}{\sqrt{b}}$$

[4] http://www.emeraldinsight.com/doi/pdfplus/10.1108/eb034867

$$k_\lambda = (1+\lambda)^{0.4}$$

$$k_e = 1; k_{uc} = 0.95; k_b = 1$$

where, W_w = Weight of the wing
k_e = Correction factor for effect of non-optimum structure
k_λ = Correction factor for effect of taper ratio
k_e = Correction factor for effect of engine location (no wing mounted engines for propeller-driven aircraft for the effect of under-carriage location)
k_{uc} = Correction factor for effect of wing mounted under-carriage
k_b = Correction factor for effect of strut location on braced wings (1 for cantilever wings)
n_w = Ultimate Load Factor W_{des} = Gross design weight of the aircraft
b = Wingspan
τ_r = Wing section thickness to chord ratio $\Lambda_{1/2}$ = Sweep Angle of mid-chord line
W_{hld} = Weight of high lift devices (considered zero here)

After substituting all the values in the above formula, the estimated weight of the wing is calculated as 82.88 kgs. This weight does not include the nacelle and rotor weight which we were not able to estimate after considering many sources. This goes well with the thumb rule according to which wing weight is around 10% of the gross weight of the aircraft.

8.4 Horizontal Tail Sizing

Here aspect ratio has been kept same as in previous iteration which is equal to 4 and static margin was selected same as 20. Also the tail has been kept as rectangular for further iterations. Using formulas mentioned in previous sections, following results have been obtained.

8.4.1 Results

Aspect Ratio	4
Static Margin	20
Area	2.416
Chord	0.777
Span	3.109

Table 8.8: Horizontal Tail: 2nd iteration

8.5 Vertical Tail Sizing

Here, the previous assumed aspect ratio and anhedral has been kept same and the same formulas are used which were there in first iteration to arrive at results.

8.5.1 Results

Anhedral	0
Aspect Ratio	1.55
Area	1.731
Chord	1.057
Span	1.638
Cn_beta	0.287
High Wing	Yes

Table 8.9: Vertical Tail: 2nd iteration

Chapter 9

Preliminary Design: III Iteration

Here, the kinematic analysis has been utilized from the previous iteration and thus is not performed again. However, dynamic analysis is as followed.

9.1 Dynamic Optimization

Here, high fidelity calculations have been performed using softwares like XFLR5 [1] to obtain the following results.

AR	Cl at zero angle of attack
5	0.074
6	0.078
7	0.082
8	0.086

Table 9.1: XFLR5 Simulation Results

AR	c	Power(kW)	P/W	Setting Angle	Cl	Cd	Cl/Cd
5	1.7	195	16.2	0.813754	0.1589	0.0248	6.407258
6	1.42	165	13.7	1.080792	0.1907	0.0252	7.56746
7	1.2	144	12	1.345499	0.2225	0.0257	8.657588
8	1.06	128.5	10.7	1.609706	0.2543	0.026	9.780769

Table 9.2: 3rd Iteration: Dynamic Results 1

[1] http://www.xflr5.com/xflr5.htm

AR	p/efficiency*density
5	343.0171875
6	290.2453125
7	253.305
8	226.0395313

Table 9.3: 3rd Iteration: Dynamic Results 2

Here, aspect ratio 5 has been finally selected due to the same argument of keeping 20% reserves of maximum pressure and selecting an optimum value for that.

9.1.1 Results

3rd Iteration	
Wing Span (b)	8.5
AR	5
Chord length (c)	1.7
Taper Ratio	0.4
Cruise Velocity	104.8 m/s
Anhedral	5
Cruise Height	2400 m

Table 9.4: 3rd iteration: Results

9.1.2 Transition Phase

- **6 DOF equations for the aircraft**
  Following are the 6-dof equations [2] which are used for dynamic analysis of transition phase.

[2] `https://www.sciencedirect.com/science/article/pii/S1000936109602363`

$$
\begin{aligned}
\mathbf{F}^{B} = {} & m_{body}\left(\frac{d\mathbf{V}_{BE}^{B}}{dt} + \omega_{BE}^{B} \times \mathbf{V}_{BE}^{B}\right) + \\
& \sum_{i=1}^{2} m_{nacelle}\left[\begin{array}{l}
\underbrace{\left(\frac{d\mathbf{V}_{BE}^{B}}{dt} + \omega_{BE}^{B} \times \mathbf{V}_{BE}^{B}\right) + \frac{d\omega_{BE}^{B}}{dt} \times \mathbf{R}_{NP}^{B} + \omega_{BE}^{B} \times \left(\omega_{BE}^{B} \times \mathbf{R}_{NB}^{B}\right) +}_{\text{Convected}} \\
\underbrace{\frac{d\omega_{NB}^{B}}{dt} \times \mathbf{R}_{NP}^{B} + \omega_{NB}^{B} \times \left(\omega_{NB}^{B} \times \mathbf{R}_{NP}^{B}\right)}_{\text{Relative}} + \underbrace{2\omega_{BE}^{B} \times \left(\omega_{NB}^{B} \times \mathbf{R}_{NP}^{B}\right)}_{\text{Coriolis}}
\end{array}\right] \\
& + \sum_{i=1}^{2} m_{rotor}\left[\begin{array}{l}
\underbrace{\left(\frac{d\mathbf{V}_{BE}^{B}}{dt} + \omega_{BE}^{B} \times \mathbf{V}_{BE}^{B}\right) + \frac{d\omega_{BE}^{B}}{dt} \times \mathbf{R}_{RB}^{B} + \omega_{BE}^{B} \times \left(\omega_{BE}^{B} \times \mathbf{R}_{RB}^{B}\right) +}_{\text{Convected}} \\
\underbrace{\frac{d\omega_{NB}^{B}}{dt} \times \mathbf{R}_{RP}^{B} + \omega_{NB}^{B} \times \left(\omega_{NB}^{B} \times \mathbf{R}_{RP}^{B}\right)}_{\text{Relative}} + \underbrace{2\omega_{BE}^{B} \times \left(\omega_{NB}^{B} \times \mathbf{R}_{RP}^{B}\right)}_{\text{Coriolis}}
\end{array}\right]
\end{aligned}
$$

Figure 9.1: Force Equation

$$
\begin{aligned}
\mathbf{H}_{body} + \sum_{i=1}^{2} \mathbf{H}_{nacelle} + \sum_{i=1}^{2} \mathbf{H}_{rotor} = {} & \int_{body} \left(\mathbf{R}_{BE}^{E} + \mathbf{R}^{E}\right) \times \left(\mathbf{V}_{BE}^{E} + \omega_{BE}^{E} \times \mathbf{R}^{E}\right) \delta m \; + \\
& \sum_{i=1}^{2} \int_{nacelle} \left(\mathbf{R}_{NE}^{E} + \mathbf{R}_{N}^{E}\right) \times \left(\mathbf{V}_{CG}^{g} + \omega_{CG}^{g} \times \mathbf{R}_{N}^{E}\right) \delta m \; + \\
& \sum_{i=1}^{2} \int_{rotor} \left(\mathbf{R}_{RE}^{E} + \mathbf{R}_{R}^{E}\right) \times \left(\mathbf{V}_{RE}^{E} + \omega_{RE}^{E} \times \mathbf{R}_{R}^{E}\right) \delta m \\
= m_{body} \mathbf{R}_{BE}^{E} \times \mathbf{V}_{BE}^{E} + \mathbf{I}_{body}^{E} \omega_{BE}^{E} + {} & \sum_{i=1}^{1} \left(m_{nacelle} \mathbf{R}_{NE}^{E} \times \mathbf{V}_{NE}^{E} + \mathbf{I}_{nacelle}^{E} \omega_{NE}^{E}\right)_{i} + \\
& \sum_{j=1}^{2} \left(m_{rotor} \mathbf{R}_{RE}^{E} \times \mathbf{V}_{RE}^{E} + \mathbf{I}_{rotor}^{E} \omega_{RE}^{E}\right)_{j}
\end{aligned}
$$

Figure 9.2: Angular Momentum Equation

$$\mathbf{M}^B = \frac{d\left(\mathbf{I}^B_{body}\omega^B_{BE}\right)}{dt} + \omega^B_{BE} \times \left(\mathbf{I}^B_{body}\omega^B_{BE}\right) +$$

$$+\sum_{i=1}^{N_1}\left\{\begin{array}{l} m_{rotor}\mathbf{R}^B_{RB} \times \left[\begin{array}{l} \left(\dfrac{d\mathbf{V}^B_{BE}}{dt} + \omega^B_{BE} \times \mathbf{V}^B_{BE}\right) + \dfrac{d\omega^B_{BE}}{dt} \times \mathbf{R}^B_{RB} + \omega^B_{BE} \times \left(\omega^B_{BE} \times \mathbf{R}^B_{RB}\right) + \\ \dfrac{d\omega^B_{NB}}{dt} \times \mathbf{R}^B_{RP} + \omega^B_{NB} \times \left(\omega^B_{NB} \times \mathbf{R}^B_{RP}\right) + 2\omega^B_{BE} \times \left(\omega^B_{NB} \times \mathbf{R}^B_{RP}\right) \end{array}\right] + \\ \underbrace{\dfrac{d}{dt}\left[\mathbf{L}_{ER}\mathbf{I}^R_{rotor}\left(\mathbf{L}_{RB}\omega^B_{BE} + \mathbf{L}_{RN}\omega^N_{NB} + \omega^R_{RN}\right)\right]}_{\text{Caused by the rotation of the rotor}} \\ + \underbrace{\omega^B_{BE} \times \left[\mathbf{L}_{ER}\mathbf{I}^R_{rotor}\left(\mathbf{L}_{RB}\omega^B_{BE} + \mathbf{L}_{RN}\omega^N_{NB} + \omega^R_{RN}\right)\right]}_{\text{Gyroscopic moment}} \end{array}\right\}$$

$$\sum_{i=1}^{N_1}\left\{\begin{array}{l} m_{nacelle}\mathbf{R}^B_{NB} \times \left[\begin{array}{l} \left(\dfrac{d\mathbf{V}^B_{BE}}{dt} + \omega^B_{BE} \times \mathbf{V}^B_{BE}\right) + \dfrac{d\omega^B_{BE}}{dt} \times \mathbf{R}^B_{NB} + \omega^B_{BE} \times \left(\omega^B_{BE} \times \mathbf{R}^B_{NB}\right) \\ + \dfrac{d\omega^B_{NB}}{dt} \times \mathbf{R}^B_{NP} + \omega^B_{NB} \times \left(\omega^B_{NB} \times \mathbf{R}^B_{NP}\right) + 2\omega^B_{BE} \times \left(\omega^B_{NB} \times \mathbf{R}^B_{NP}\right) \end{array}\right] + \\ \underbrace{\dfrac{d}{dt}\left[\mathbf{L}_{EN}\mathbf{I}^N_{nacelle}\left(\mathbf{L}_{NB}\omega^B_{BE} + \omega^N_{NB}\right)\right]}_{\text{Caused by therotationof thenacelle}} \\ + \underbrace{\omega^B_{BE} \times \left[\mathbf{L}_{EN}\mathbf{I}^N_{nacelle}\left(\mathbf{L}_{NB}\omega^B_{BE} + \omega^N_{NB}\right)\right]}_{\text{Gyroscopicmoment}} \end{array}\right\}$$

Figure 9.3: Moment Equation

- **Power Requirement**
  Here power required has been calculated to check that if it is in the limit of the engine selected after 1st iteration using an existing code [3]. Also we have

[3] https://github.com/Tiltrotor/ardupilot

kept the rotation rate for the propeller to be $0.25^o/sec$ so that the transition is smooth and there is not much change in centripetal acceleration.

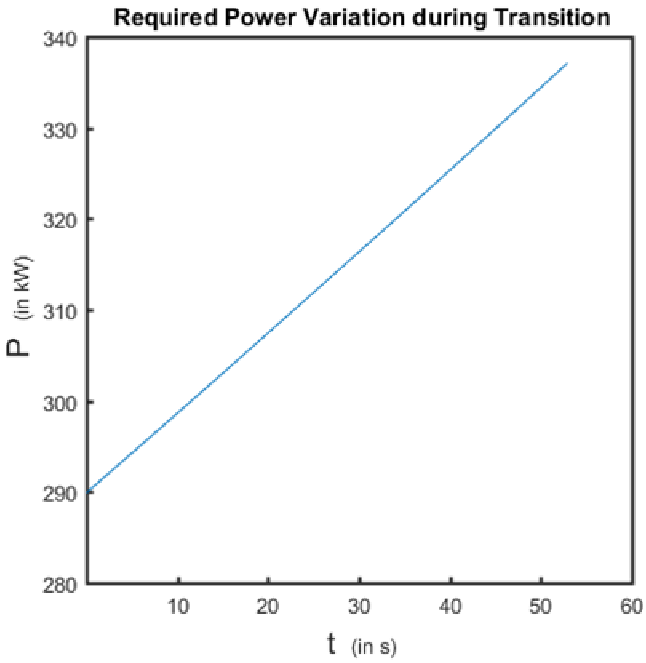

Figure 9.4: Power required for Transition phase

9.2 Horizontal Tail Sizing

Here aspect ratio has been kept same as in previous iteration which is equal to 4 and static margin was selected same as 20. Using formulas mentioned in previous sections, following results have been obtained.

9.2.1 Results

Aspect Ratio	4
Static Margin	20
Area	3.522
Chord	0.938
Span	3.753

Table 9.5: Horizontal Tail: 3rd iteration

9.3 Vertical Tail Sizing

Here, the previous assumed aspect ratio has been kept same however anhedral has been increased to 5 deg because of rolling stability. The same formulas are used which were there in first iteration to arrive at results.

9.3.1 Results

Anhedral	0
Aspect Ratio	1.55
Area	2.091
Chord	1.161
Span	1.800
Cn_beta	0.283
High Wing	Yes

Table 9.6: Vertical Tail: 3rd iteration

9.4 Spar Design

Using above derived results, that is angle of setting as 0.8^o and cruise velocity as 105 m/s. First of all, local lift coefficient was found using software XFLR5 [4]. After that bending moment was plotted against the wingspan. Using the maximum value of bending moment at root, the spar was designed assuming an I-beam.

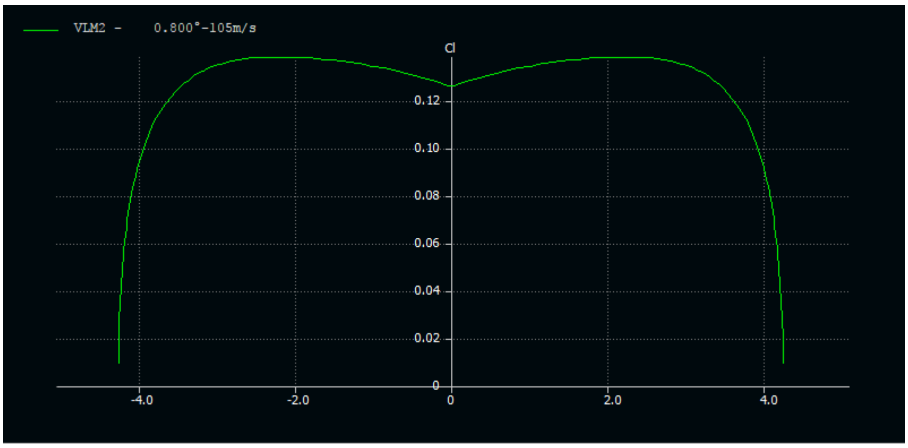

Figure 9.5: Lift Coefficient along the span

[4] http://www.xflr5.com/xflr5.htm

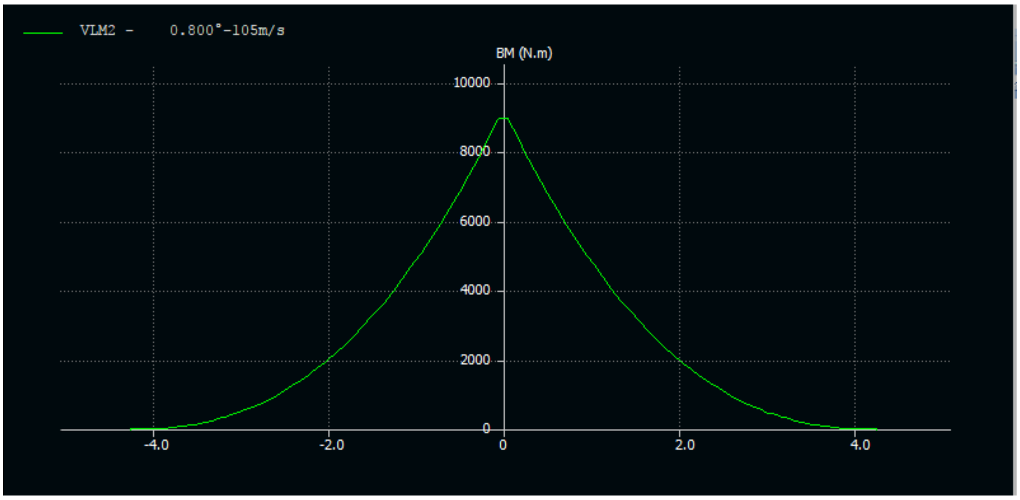

Figure 9.6: Bending Moment along the span

From graph, we can find that max. bending moment is 9062 Nm. The material chosen is Al7075-T6. It has a breaking strength of 503 MPa. A safety factor of 2 has been selected for strength calculations.

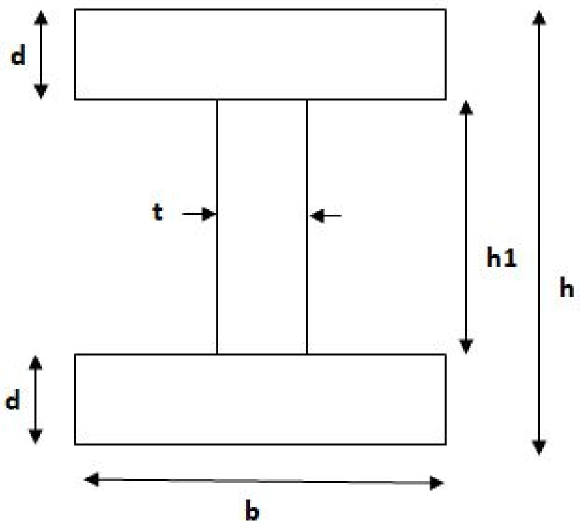

Figure 9.7: Cross-section of the I-beam spar

The spar is supposed to be at a distance of c/3 from the leading edge where thickness of the airfoil at c/3 is 0.0797*c (= 19.35 cm) which will be the height of the spar. Also d is assumed to be 6% of height of spar and t is assumed to be 1.8%

of height of the spar. Using this value in below formulas, we can find the width of the spar.

$$\sigma = \frac{My}{I}$$

$$\sigma = \sigma_{breaking}/1.5$$

$$I = \frac{1}{12}(bh^3 - bh_1^3 + th_1^3); y = \frac{h}{2}$$

The above values are as follows:

$$d = 1.16mm; h1 = 17.05cm; t = 3.5mm; M = 9062Nm$$

For these values, the value obtained of b is 1.06 cm.

Chapter 10

CAD Designs

All units for each iteration is written in meters.

10.1 Preliminary Design: Iteration I

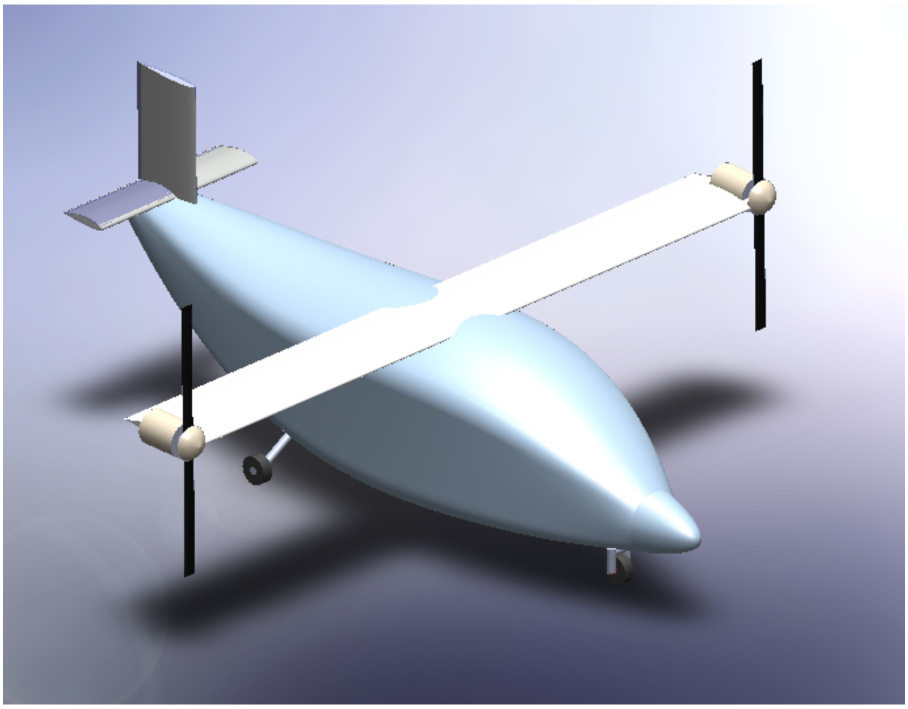

Figure 10.1: 1st Iteration: Isometric View

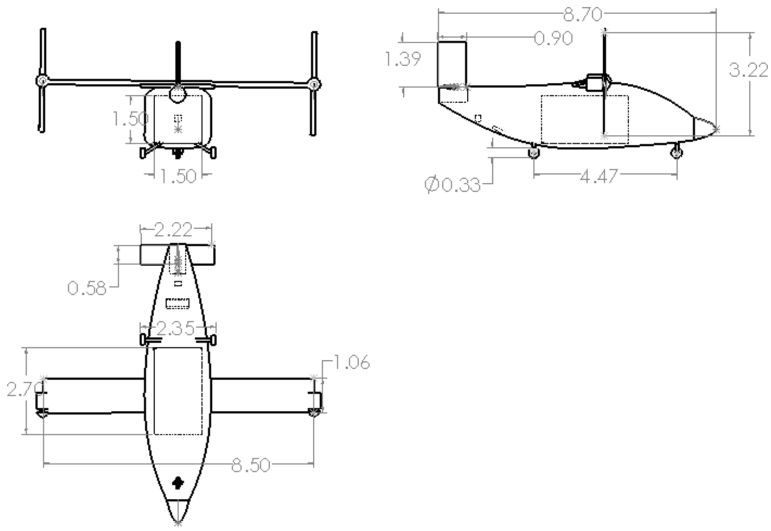

Figure 10.2: 1st Iteration: 3-axis View

10.2 Preliminary Design: Iteration II

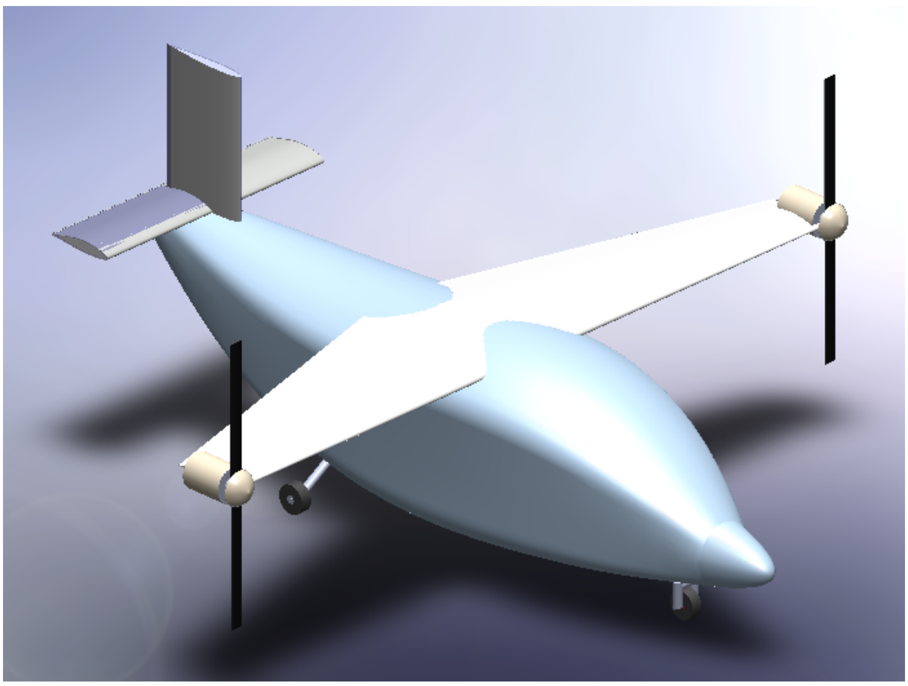

Figure 10.3: 2nd Iteration: Isometric View

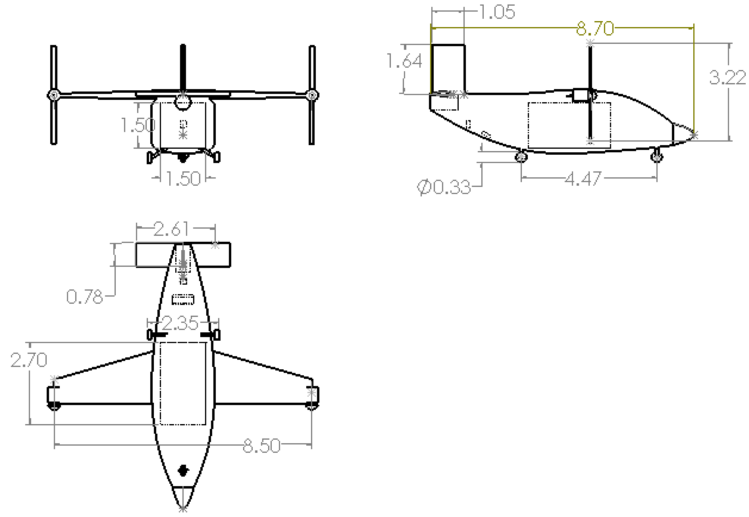

Figure 10.4: 2nd Iteration: 3-axis View

10.3 Preliminary Design: Iteration III

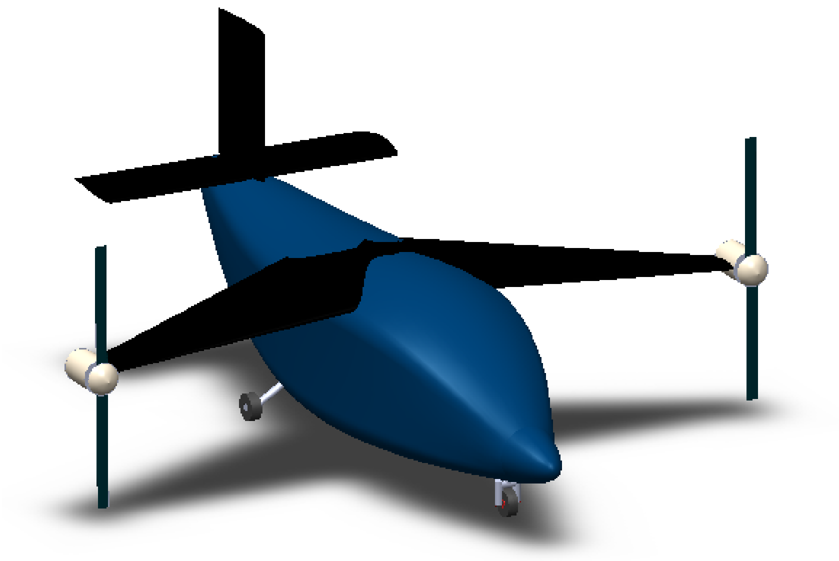

Figure 10.5: 3rd Iteration: Isometric View

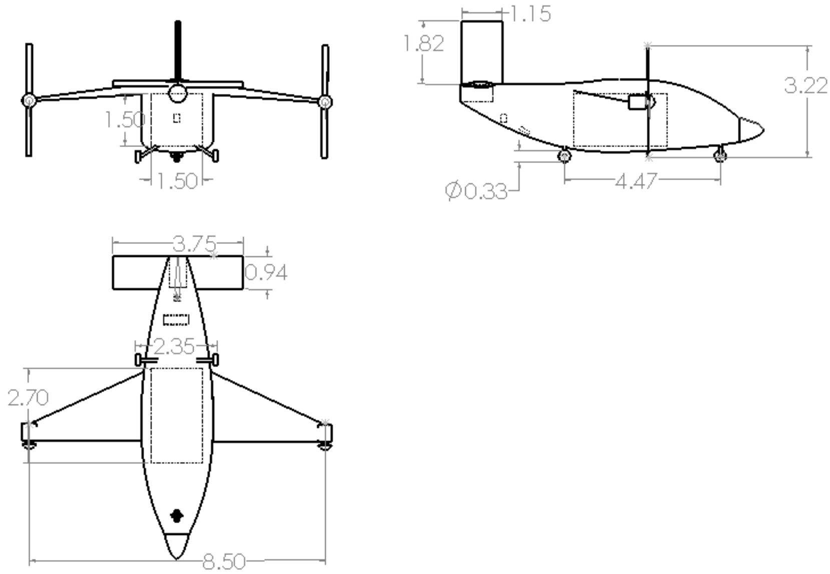

Figure 10.6: 3rd Iteration: 3-axis View

Chapter 11

Conclusions

- Designed a medically compliant stable aerial vehicle with all necessary emergency support and medical attendant
- Landing of the vehicle is facilitated on the nearby wider lane which is a requirement for a metropolitan city
- Mission target of 20 km within 8 mins has been achieved in less than 6 mins in the current stage of preliminary design

Chapter 12

Future Works

- Addition of control surfaces
- Single engine to twin engine configuration
- Addition of an extra jet engine or some flap to deal with misalignment of CG during take-off or landing
- Development of Tilting mechanism

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