Title of the project

Design and Performance Analysis of Fire Fighting Aircraft

Submitted by

Sachit Malhotra (R290209052)

Vaishnav Dutt (R290209068)

Mohit Miyan (R290209034)

Of

BACHELOR OF TECHNOLOGY

In

AEROSPACE ENGINEERING

Under the Supervision of

Dr. Om Prakash

HOD (Aerospace department)

DEPARTMENT OF AEROSPACE ENGINEERING

COLLEGE OF ENGINEERING STUDIES (COES)

UNIVERSITY OF PETROLEUM & ENERGY STUDIES

DEHRADUN- 248007, INDIA

APRIL 2013

CERTIFICATE

I hereby certify that the work which is being presented in the project report entitled "**Design and performance analysis of fire fighting aircraft**" in partial fulfilment of the requirements for the satisfactory performance for B.Tech Aerospace Engineering, Major Project submitted in the Department of Aerospace Engineering, University of Petroleum and Energy Studies, Dehradun is an authentic record of my own work carried out during a period from July 2012 to April 2013.

Submitted by –

Sachit Malhotra (R290209052)

Mohit Miyan (R290209034)

Vaishnav Dutt (R290209068)

Mentor –

Dr. Om Prakash

HOD (Aerospace department)

ACKNOWLEDGEMENTS

It is with a sense of great satisfaction and pride that we are sitting down to pen out our major project thesis report. First and foremost we sincerely salute our esteemed Institution **University of Petroleum & Energy Studies, Dehradun** for giving this golden opportunity for fulfilling our warm dreams of becoming a bachelor.

The first and foremost person we would like to express our deep sense of gratitude and profound thanks to our guide **Prof. Dr. Om Prakash,** Head, Department of Aerospace Engineering, UPES Dehradun for his valuable advice, suggestions, insurmountable guidance which played a vital role in carrying out our project work successfully.

We would like to thank **Prof. Vijay Kumar Pathidar , and Prof. Linsu Sibastin**, for his help & encouragement during the work. We also thank all faculty, staff, and students of Department of Aerospace Engineering and for rendering help during various stages of the project work. We would like to thanks **our parent, our sisters/brothers and our friends.** for their support and encouragement for carrying out our project work in a well sincere manner

Sachit Malhotra [R290209052]

Vaishnav Dutt [R290209068]

Mohit Miyan [R290209034]

ABSTRACT

Fire fighting Aircraft means support of the fire fighters on the ground from aircraft in the air. Aircraft can access steep, rocky or unsafe areas before ground forces are able to gain entry.

Aerial fire fighting is the use of aircraft and other aerial resources to combat wildfires. The types of aircraft used include fixed-wing aircraft and helicopters. Smokejumpers and rappellers are also classified as aerial fire-fighters, delivered to the fire by parachute from a variety of fixed-wing aircraft, or rappelling from helicopters. Chemicals used to fight fires may include water, water enhancers such as foams and gels, and specially formulated fire retardants.

Both fixed wing and rotary wing aircraft are capable of aerial fire-fighting, with possible chemicals including water, foams, gels and fire retardants. The key characteristics of a fire-fighting aircraft include a high useable payload weight and a high cruise speed. Several aircraft designs have demonstrated excellent aerial fire fighting effectiveness, including those specially modified for aerial fire-fighting purposes. For large fires, modified commercial airliners or military transport aircraft have been used with great success. Agricultural aircraft often have poor aerodynamic efficiency, but posses improved manoeuvrable over larger aircraft.

In the undertaken project that is "Design and analysis of fire fighting aircraft" we would be doing the design and performance analysis of a fire-fighting aircraft. A design tailored for unique fire conditions would give the aircraft an advantage in performance and mission effectiveness compared with fire-fighting aircraft currently used in the world. The project will focus on the conceptual phase of the design process and performance analysis of fire-fighting aircraft. Design process include statistical analysis of various aircrafts and calculation of takeoff weight of the aircraft, fuselage design, wing design, tail design, and propulsion system integration Aerodynamic performance include calculation of lift and drag coefficient, weight and balance analysis and calculation of L/D, W/S, T/W. After doing all the above calculations a solid works model of required dimension is made.

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ABBREVATIONS

2D..Two -Dimensional
3D ....Three-Dimensional
CAD... Computer Aided Design
CASA ....Civil Aviation Safety Authority
CFS ....Country Fire Service
FAR ....Federal Aviation Regulations
MAC.. Mean Aerodynamic Chord of wing
NACA...................................... National Advisory Committee for Aeronautics

SYMBOL

a	Speed of sound
V_{climb}	Climb Velocity
$(L/D)_{aircraft}$	Aircraft (L/D)
$(L/D)_{aerofoil}$	Aerofoil (L/D)
$(L/D)_{max}$	Maximum L/D
$(L/D)_{wing}$	Wing (L/D)
$(t/c)_{wing}$	Wing thickness ratio
(W/S)	Wing loading
$A_{exhaust}$	Area of Exhaust
A_{intake}	Area of Intake
AR_{wing}	Wing aspect ratio
B	The distance between the nose and the main landing gears
b_{wing}	Wing span
C_d	Aerofoil drag coefficient
CG_{aft}	The distance from the nose of the aircraft to the most aft CG
CG_{fore}	The distance from the nose of the aircraft to the most forward CG
CG_{wing}	Wing centre of gravity
CL	Wing lift coefficient
Cl	Aerofoil lift coefficient
CL_{max}	Maximum wing lift coefficient
Cl_{max}	Maximum aerofoil lift coefficient
$CL_{max, L}$	Maximum wing lift coefficient at landing

$CL_{max, TO}$	Maximum wing lift coefficient at takeoff
$CL\alpha$	Lift-curve slope
C_m	Aerofoil pitching moment coefficient
C_M	Wing pitching moment coefficient
C_{wing}	Wing chord
D	Drag
$D_{fuselage}$	The diameter of the fuselage
D_P	Propeller Diameter
g	Acceleration due to gravity (32.2 slugs/ft3)
H_{lg}	The height of the landing gear (from the ground to the bottom of the fuselage)
H_{tail}	The height of the tail above the bottom of the fuselage
HW	The half-width of the main landing gear, i.e. the lateral distance between a main landing gear and the centre-line of the aircraft
L	Lift
l	characteristic length
$L_{fuselage}$	The length of the fuselage [ft]\
M	Mach number
Ma	The distance between the main landing gear and the most aft CG
Mf	The distance between the main landing gear and the most forwardCG
Na	The distance between the nose landing gear and the most aft CG
Nf	The distance between the nose landing gear and the most forwardCG
N_p	Number of propeller blades
P_{bl}	Blade Power Loading
P_{max}	Maximum power output per engine
S	Platform area
$S_{flapped}$	Flapped surface area
SHP	Uninstalled Engine Power
S_{ref}	Reference surface area

CHAPTER 1

INTRODUCTION

Aerial fire fighting is the use of aircraft and other aerial resources to combat wildfires. The types of aircraft used include fixed-wing aircraft and helicopters. Smokejumpers and rappellers are also classified as aerial fire-fighters, delivered to the fire by parachute from a variety of fixed-wing aircraft, or rappelling from helicopters. Chemicals used to fight fires may include water, water enhancers such as foams and gels, and specially formulated fire retardants.

Air Attack or Air Tactical Aircraft is an airplane that flies over an incident, providing tactical coordination with the incident commander on the ground, and directing air tankers and helicopters to critical areas of a fire for retardant and water drops.

Figure 1.1 Air tactical aircraft Source: Wikipedia

Air tanker is a fixed-wing aircraft that can carry fire retardant or water and drop it on or in front of a fire to help slow the fire down. The S-2T carries 1,200 gallons of retardant and has a crew of one – the pilot.

Figure1.2 Very large air tanker Source: Wikipedia

Helicopter is a rotary-wing aircraft that can be fitted with a tank or carry a bucket with water or fire retardant. The tanks or buckets can be filled on the ground by siphoning water from lakes, rivers or other water sources.

Figure 1.3 Helicopter Source: Wikipedia

Fire Retardant is a slurry mix consisting of a chemical salt compound, water, clay or a gum-thickening agent, and a colouring agent. The retardant is used to slow or retard the spread of a fire.

The purpose of this report is to detail the design and performance analysis of a fire-fighting aircraft

1.1. Background

Bushfires present a significant risk to people, land and resources. One of the most effective methods of containing a bushfire is through aerial fire-fighting, which is the use of an aircraft for releasing fire fighting chemicals onto a fire. Both fixed wing and rotary wing aircraft are capable of aerial fire-fighting, with possible chemicals including water, foams, gels and fire retardants. The key characteristics of a fire-fighting aircraft include a high useable payload weight and a high cruise speed. Several aircraft designs have demonstrated excellent aerial fire fighting effectiveness, including those specially modified for aerial fire-fighting purposes. For large fires, modified commercial airliners or military transport aircraft have been used with great success. Agricultural aircraft often have poor aerodynamic efficiency, but posses improved manoeuvrable over larger aircraft.

1.2. Aim and Objective

The aim of this project is to design a fire-fighting aircraft. A design tailored for unique fire conditions would give the aircraft an advantage in performance and mission effectiveness compared with fire-fighting aircraft currently used in the world. The project will focus on the conceptual phase of the design process and performance analysis of fire-fighting aircraft.

CHAPTER 2

LITERATURE REVIEW

The conceptual design of the fire-fighting aircraft required research of current prototypes and design techniques through a literature review. A comprehensive investigation was carried out, which yielded a number of useful references, including textbooks, published reports, databases and websites. These sources will be discussed in the following sections, and include those used for the design of the aircraft structure, configuration and sizing. During the feasibility study and statistical analysis; numerous aircraft were referenced or statistical data. Aircraft primarily designed for aerial fire-fighting did not provide adequate data, so agricultural aircraft were also considered. Of particular interest were the Air Tractor series of aircraft.

The literature used for the project is based on information and equations contained in a range of texts pertaining to different aspects of aircraft design. For the general embodiment design, several textbooks and reference books were used. These were namely the Airplane Design series (Roskam, 2004) and Aircraft Design: A Conceptual Approach (Raymer, 1992). The Roskam series provides an incremental approach to the design of an aircraft, which can be adapted to suit the requirements specific to the fire-fighting aircraft. In contrast, Raymer offers a classical approach to aircraft design with detailed theory and equations.

Aerofoil selection was aided with the use of the UIUC Aerofoil Coordinate Database (UIUC 2008). This database provides a considerable selection of aerofoil designed and recommended for aircraft. In addition, Java foil aerofoil analysis online software was used to compare and select the most appropriate and suitable aerofoils for the aircraft. Introduction to Aeronautics: Design Perspective (Brandt et al. 2004) was used for stability calculations and determination of landing gear location. Other references have also been used throughout the project, and are cited where applicable.

CHAPTER 3

PROTOTYPES

The selection of these prototypes was based on the following:

- Similar physical size to the expected fire-fighting aircraft size
- Similar weight to the expected fire-fighting aircraft size
- Similarity of mission requirements and applications

The Air Tractor 602 is a single engine turboprop agricultural aircraft. It has a maximum takeoff weight of 12,500 lb and has a payload capacity of 630 gallons (2,380 L). The first flight of the Air Tractor 602 occurred in 1995, with production currently continuing. (Air Tractor 2009)

Figure 3.1 - Air Tractor 602 Source: Australian Fire fighting report

The Air Tractor 802F is a single engine turboprop aircraft primary designed for fire-fighting applications. It has a takeoff weight of 1,600lb and a payload capacity of 820 gallons (3,100L). The Air Tractor 802F is a modified version of the Air Tractor 802 agricultural aircraft. The 802 is the largest existing agricultural aircraft, and as such, defines the boundaries of agricultural aircraft design. Both models are popular as they offer high efficiency and similar performance compared with larger twin-engine aircraft. The first flight of the Air Tractor 802 occurred in 1990, and production of both the 802 and 802F models is currently continuing. The 802F can also be fitted with water Floats to create an amphibious aircraft (Air Tractor 2009).

Figure 3.2 - Air Tractor 802 Source: Australian fire fighting report

The Canadair CL-215 is a twin engine amphibious fire-fighting aircraft. It has a take-off weight from land of 43,500 lb and a payload capacity of 1,400 gallons (5,455 L). The first flight occurred in 1967and production ceased in 1998 with 121 aircraft built. The CL-215 has a flying boat configuration, and hence, offers significant aerodynamic advantages when compared with the Air Tractor 802Ffitted with floats. The CL-215 was designed for Canadian conditions, where large lakes provide still flat surfaces where rapid water collection can occur.

Figure3.3Canadair CL-21 Source: Australian Fire fighting report

CHAPTER 4

CONCEPTUAL DESIGN

The conceptual design process aimed to generate, select and develop the most feasible concepts that could meet all the design requirements. This process was conducted using a classical approach involving multiple design iterations. Each iteration led to further development of the concepts until design decisions were made based on sound knowledge and calculations. The following section outlines the conceptual design process, from initial configuration design through to plan form design, aerofoil and control surface selection, fuselage sizing and propulsion system selection. The resultant design is brought together in three view drawings.

4.1 Technical Task

This section outlines design requirements for the aircraft. Requirements due to standards, performance, technological level, economics, main sub-systems and reliability are used to define theoverall constraints on the aircraft.

4.1.1 Performance Requirements

4.1.1.1 Aircraft Base Location and Range

The aircraft is being designed to supplement the existing aerial fire-fighting. The location at which the aircraft would potentially be based is an important consideration when determining the range of the aircraft. Once possible bases are identified, the range can be determined by identifying distances that the aircraft would be required to travel to the site of a fire. So range of about 150 km of radius is assumed so that it can cover most part of Uttrakhand and nearby hilly areas.

4.1.1.2 Payload Weight

Aerial fire-fighting aircraft standards require that fixed wing aircraft drop retardant or water payloads in an effective zone which is no less than 40 m long and 15-20 m wide, and that no more than 15% of the release falls outside of this effective zone (NAFC 2004). The standards require a minimum coverage of 0.2 L/m2. However, coverage up to 4.0 L/m2 is required to suppress the heaviest bushfires .Standards also require a leakage loss rate of no more than15 L/hr. To provide 4 L/m2 coverage to an effective zone of 40m by 20m and assuming a total time between payload delivery and filling of 140 minutes (20 minutes between filling and takeoff, 100 minutes to target and 20 minutes on target).
Long-term fire retardants, such as Phos-Chek D-75-R, are up to three times more effective in containing bushfires than water. The payload of the fire-fighting aircraft can be assumed to have a similar density to Phos-Chek D-75-R of 1.067 kg/L (USDA Forest Service 2006). The payload mass is then 1990 kg, which was rounded up to 2000 kg as a conservative estimate to allow for possible density variations. A payload of 2000 kg of Phos-Chek allows the payload drop types seen in Table 4.1

A three-drop configuration may be possible, depending on the payload delivery system, but is not required by aerial fire-fighting aircraft standards.

Table 4.1: Payload Drop Types.

Drop type	Coverage
One drop	4 L/m2
Two drops	2 L/m2
Four drops	1 L/m2

4.1.1.3 Crew Weight

NAFC outlines a pilot weight of 190 lb (86kg), with 15kg of baggage. The aircraft should only provide accommodation for one crew member. No additional crew members are required to operate the aircraft. Hence, controlling the aircraft and releasing the fire retardant are both performed by the pilot.

4.1.2 Technical Level

The aircraft is designed to replace existing aircraft, and hence, should demonstrate improved technologies. In particular, increased fuel efficiency, improved materials and better manufacturing processes are desirable .The cockpit should also benefit from superior instrumentation. It is intended that this aircraft will be flown by a single pilot with high-level skills and appropriate certification.

4.1.3 Economical Parameters

The aircraft should be affordable by small companies as well as larger organizations and government bodies. It is intended that the aircraft should be more affordable than competing aircraft, in initial purchase cost, running costs and maintenance costs.

4.1.4. Main System Requirements

4.1.4.1. Propulsion System Requirements

Propulsion requirements are outlined in FAR 25 Subpart E. Particular reference should be made to Section 25.961 (Fuel System Hot Weather Operation). No specifications regarding engine number or engine type exist.

4.1.4.2. Landing Gear Subsystem

Rural operation requires that the aircraft must be able to operate from paved and unpaved runways. Amphibious landing capabilities are not required. FAR 25 Section 25.473 requires the following:

- Maximum descent velocity of 10ft/s at the design landing weight
- Maximum descent velocity of 6ft/s at the design takeoff weight

- The coefficient of friction between the tires and the ground should be less than 0.8
- Fuselage Requirements
- The fuselage design is required to accommodate the fire retardant release system.
- Fire Retardant Release System
- NAFC specifies the following requirements:

4.1.4.3. Fuselage Requirements

The fuselage design is required to accommodate the fire retardant release system.

4.1.4.4. Fire Retardant Release System

NAFC specifies the following requirements:

- The fire retardant release system must be able to produce a "full dump" with a minimum flow rate of 1000 liters per second under typical dumping conditions.
- The system must be capable of dropping fire retardants at rates less than the maximum flow rate.
- It is recommended that the system is capable of at least four flow rates. Flow rates of 500 liters per second, 1000 liters per second and 1500liters per second are recommended
- The systems must be capable of splitting the load into more than one drop. Systems with capacity greater than 3000L must be able to drop the load in four parts.
- The system should be well constructed and include appropriate sealing mechanisms to prevent leakages. During sixty minutes of static ground testing, losses should be less than two liters. During a twenty minute turnaround, mission losses should be less than five liters. The systems should have the capability to inject the water payload with a measured amount of foam concentrate.

4.1.4.5. Reliability and Maintainability

NAFC recommends the following:

- Systems should be simple, robust and reliable
- Systems should have an appropriate level of redundancy.
- In the event of partial equipment failure, it must be possible to continue the firebombing mission.
- The use of specialized parts should be avoided

4.1.4.6. Safety

FAR 91 Section 91.107 states the requirements of one shoulder safety belt as a minimum requirement for all aircraft. FAR Part 137 requires that agricultural aircraft be fitted with a bird proof windshield, wire cutters and wire deflectors due to their low altitude operation. The criteria will also be applied to the aircraft.

4.1.4.7. Unification level

The vehicle should incorporate both new and existing design components. Inherited design elements include the wing and empennage aerofoil, the propulsion system, and the flight deck instrumentation. New designs will occur for the fuselage and fire retardant release system. Iterative design of the aircraft aerodynamics and the fire retardant release system will be required to reach the optimal design solution.

4.1.4.8. Ergonomics

NAFC recommends that the aircraft should be controllable without excessive strength or movement by the pilot. In particular, fire retardant release should not result in large pitch movements or excessive trim changes.

4.1.4.9. Cabin Design

To achieve high accuracy when releasing the fire retardant, the pilot visibility pattern must be considered. The cockpit should be designed such that the over-nose angle is a minimum of ten degrees. The pilot should have over-the-side vision of 35 degrees, with 70 degrees of head movement. The pilot should have completely unobstructed upward vision angles. The cockpit windscreen should have a minimum angle of 30 degrees to prevent mirroring effect of sunshine angles.

CHAPTER 5

STATISTICAL ANALYSIS

Statistical analysis of relevant data is required to produce the technical diagram and suggest base parameters for design. The technical task outlined a payload capability of 8,820 lb and a range of584nm. These definitions were used to determine the relevance of aircraft data. Only aircraft currently in use were considered.

The statistical analysis was limited by relevant fire-fighting aircraft. Consequently, additional data points were obtained by using agricultural aircraft and small regional turboprops. The investigated aircraft included the following:

- Bombardier Canadair 415 (Fire-fighting Aircraft)
- Bombardier Canadair CL-215 (Fire-fighting Aircraft)
- Air Tractor AT602 (Fire-fighting Aircraft)
- Air Tractor AT802 (Fire-fighting Aircraft)
- PZL-Mielec_M-18_Dromader (Agricultural Aircraft)
- Antonov An-2 (Agricultural Aircraft)
- G-164B Super B Turbine (Agricultural Aircraft)
- Pac Cresco (Agricultural Aircraft)
- CASA C-212 (Regional twin turboprop)
- Saab 340B (Regional twin turboprop)
- Sukhoi Su-80 (Regional twin turboprop)
- Convair CV-240 (Regional twin turboprop)
- Embraer EMB 110 Bandeirante (Regional twin turboprop

Properties that were investigated included:

· Weights (takeoff, empty and payload weights)
· Speed (maximum, cruse and stall speed)
· Rate of climb
· Range
· L/D ratio

Table 5.1 Data of some relevant aircrafts

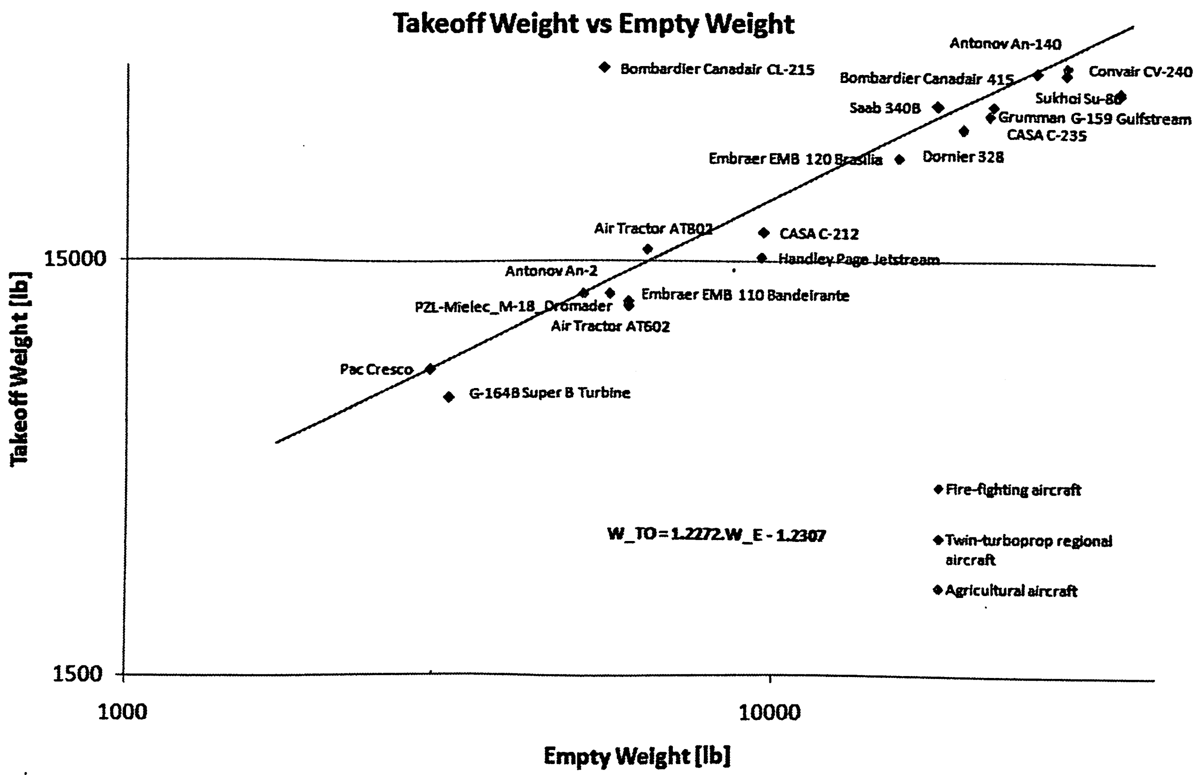
Aircraft	Take-off weight (lb)	Empty Weight	Payload (lb)	Maximum Speed	Range (nm)
Bronco OV-10	14444	6893	3000	281	576
TBM Avenger	17893	10545	2000	240	869
Douglass DC-3	25200	18300	5000	206	890
Grumman F7F-3 Tiger cat	25720	16270	1000	400	1000
Grumen S2-Tracker	26147	23435	-	260	1390
Grumman CDF S-2A Tracker	27000	18315	6664	243	869
Bombardier Canadair 415	37850	28400	13500	203	1310
Bombardier Canadair CL-215	43500	26900	12000	160	1310
Consolidated PB4Y-2 Privateer	65000	27485	8000	206	2450
Boeing B-17 Flying Fortress	65500	36135	6000	249	1738
Alenia C-27J Spartan	70106	37479	19841	315	1160
Douglas DC-4	73000	43300	-	244	1897
Fairchild C-119 Boxcar	74000	40000	10000	257	1980
Beriev Be-200 Altair	83550	60850	655600	388	1000
Shinmaywa US-1A	94800	56505	30000	276	2060
P3-Orion	142000	77200	-	411	2070
McDonnell Douglas DC-7	143000	72763	24990	353	4001
C-130 Hercules	155000	83000	45000	348	2835
JRM Mars	165000	75573	32000	192	4300
McDonnell Douglas DC-10-10	430000	240171	99960	530	3302
Boeing 747	833000	392800	200287	510	6700
Antonov An-2 'Colt'	-	7300	3307	139	485

Table 5.2 Statistical analysis of some relevant aircraft

Aircraft	Empty Weight (lb)	Take-off weight (lb)	Cruise Speed (knots)	Stall Speed (knots)	Range (nm)
G-164B Super B Turbine	3150.00	7020.00	113.00	53.85	172.00
Pac Cresco	2950.00	8250.00	140.00	45.00	364.00
PZL-Mielec_M-18_Dromader	5975.00	11700.00	100.00	59.00	540.00
Antonov An-2	7300.00	12000.00	100.00	26.00	456.00
Air Tractor AT602	5600.00	12500.00	126.00	86.00	538.77
Embraer EMB 110 Bandeirante	7837.00	12500.00	184.00	76.00	1060.00
Handley Page Jetstream	9613.00	15332.00	230.00	86.00	680.00
Air Tractor AT802	6400.00	16000.00	169.00	93.00	695.00
CASA C-212	9680.00	17600.00	170.00	75.00	237.00
Embraer EMB 120 Brasilia	15655.00	26378.00	300.00	55.00	850.00
Dornier 328	19670.00	30840.00	335.00	93.08	1000.00
CASA C-235	21605.00	33290.00	245.00	107.00	1549.25
Grumman G-159 Gulfstream I	21900.00	35100.00	250.00	90.00	2206.00
Saab 340B	17945.00	35245.00	250.00	115.00	935.00
Sukhoi Su-80	34241.00	38045.00	232.00	95.00	702.00
Antonov An-140	28240.00	42220.00	250.00	95.38	745.00
Convair CV-240	25445.00	42500.00	243.00	86.92	1042.00
Bombardier Canadair CL-215	26900.00	43500.00	156.00	92.00	1310.00
Bombardier Canadair 415	28400.00	43850.00	180.00	68.00	1319.11

5.1 Empty Weight versus Takeoff Weight

A technology diagram was created to determine the relationship between takeoff weight and emptyweight.



Plot- 5.1 Takeoff Weight versus Empty Weight [Raymer 2006]

Three data sets were used to determine a relationship between takeoff weight and empty weight. The data sets were chosen to match the desired aircraft demographic as closely as possible .Sufficient data on fire-fighting aircraft were not available, so data on large agricultural aircraft and regional twin turbo-prop aircraft were used to supplement the statistical analysis. All aircraft used a turboprop engine for propulsion, and were all designed within the last thirty years. The relationship between takeoff weight and empty weight is best described using a logarithmic equation. The outlier (Bombardier Canadair CL-215) was The following resulting relationship was used as part of the matching diagram:

5.2 Cruise Speed

The technical task outlines a cruise speed of 375km/h (202 knots). Agricultural aircraft exhibit substantially lower speeds than that required, whilst regional aircraft exhibit speeds higher than the design requirement. The difference in trends between the three data sets shows that the statistical analysis is attempting to define an aircraft that is not simply classified. The aircraft required by the technical task has the roles of a fire-fighting aircraft, and operates similarly to an agricultural aircraft.

The aircraft is heavier than an agricultural aircraft, and lighter than a twin turboprop aircraft

5.3 Rate of Climb

The rate of climb from the statistical analysis was determined to be 850 ft. This was influenced by the Air Tractor AT-802F fire-fighting aircraft. As discussed in the technical task, FAR 25 requirements dictate the minimum rate of climb as 300ft, which is much lower than the rate of climb from the statistical analysis. The difference is due to the agility and manoeuvrability required in order to fight fires effectively.

5.4 Cruise Altitude

The cruise altitude from the statistical analysis was based on the Air Tractor AT-802F, which was deemed to have the same altitude requirements for fire fighting. The altitude from prototyping in the statistical analysis was 14,000ft.

5.5 L/D Estimation

Data on L/D statistics are not readily available. For the statistical analysis, the L/D was calculated from other statistics using the Breguet Range equation. Usage of this equation is likely to be accurate to within 30%, due to the following assumptions

· The aircraft is cruising for the entire flight
· The aircraft has a constant L/D at all times
· The aircraft has a constant cruise speed at all times
· The aircraft has a constant fuel consumption at all times

For the design weight, the L/D for cruise is 12.7. The L/D for loiter is 0.866(L/Dcruise) (Raymer 2006).
Thus,

$L/D_{CRUISE} = 12.7$ and $L/D_{LOITER} = 14.67$

5.5.1 Mission Profile

The following section outlines the mission profile and its associated requirements.

5.5.1.1 Mission Profile Diagram

Figure below diagrammatically illustrates the mission profile for the fire-fighting aircraft.

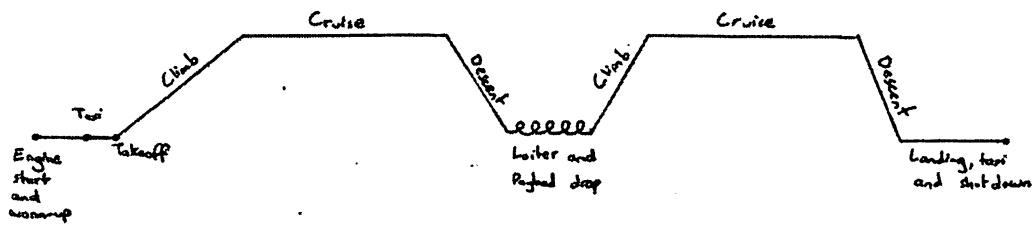


Figure: 5.2 Mission profiles [Raymer 2006]

5.5.1.2 Mission Profile Requirements

The phases of the mission profile and associated relevant details are given in Table 5.3

Table 5.3 - Mission Profile Summary

	Phase	Details
1	Engine start and warm-up	
2	Taxi	
3	Takeoff	
4	Climb	Climb to 8000 ft
5	Cruise	150 km (335.54 sm) at 375 km/h
6	Descent	To assumed payload drop altitude of 100 ft
7	Loiter and Payload drop	20 minutes (E=0.33 hrs) at 1.1 Vstall
8	Climb	Climb to 8000 ft
9	Cruise	150km (335.54 sm) at 375 km/h
10	Descent	To sea level
11	Landing, taxi and shut down	

5.6 Weight Estimation

The takeoff weight and empty weight of the fire-fighting aircraft can be estimated from the mission profile, the requirement and the results of the statistical analysis. The Technical Task Requirements

The technical task requirements are summarized below:

- Payload: 2000 kg (4444.4 lbs)
- Single pilot and baggage design weight: 86kg + 15kg = 101 kg
- Cruise speed: 375 km/h = 341.7542 ft/s
- Radius: 150 km
- Loiter time for payload drop: 20 minutes
- Cruising altitude: 8000ft

- Stall speed: 170km/hr = 154ft/s
- L/D $_{cruise}$=12.7 and L/D $_{loiter}$ = 14.67

5.7 Remaining Sizing Requirements

Several parameters were not defined by the stages above, and were estimated from prototypes and literature. Values for these parameters and the corresponding prototypes are shown in Table

Table 5.4 - Parameters Estimated from Prototypes and Literature

Parameter.	Value	Source
Rate of Climb	850 fpm = 14.167 ft/s	Air Tractor 802F (Air Tractor 2007)
Propeller Efficiency	0.88	(Raymer 2006)
Cruise Power SFC	0.471 lbs/hp/hr	(Honeywell 2009)
Loiter Power SFC	0.571 lbs/hp/hr	cp(loiter) = 0.1 + cp(cruise) (Raymer 2006)
Reserve Fuel Fraction	0.06	(Roskam 2005)

5.8 Configuration Selection

Fire-fighting aircraft can be classified by their payload capability, propulsion system and landing system. Payload capacity for the aircraft was specified by the technical task as 8,820 lb. This payload is heavier than that carried by agricultural or existing single engine turboprop aircraft. However, the payload is much less than that carried by twin-engine aircraft. Consequently, both configurations were investigated.

Common propulsion systems include jet, turboprop, piston or radial engine. Aircraft that use a jet propulsion system are significantly faster than those powered by radial or piston engines. However, large aircraft have reduced aerobatic capabilities and are hence, rarely used for fire-fighting aircraft. Turboprop and piston engines are regularly used for fire-fighting aircraft. Both propulsion methods are further investigated.

5.8.1 Concept 1 (flying boat)

The first concept considered was a flying boat configuration, where the fuselage can be used as a hull so that the aircraft can take off and land on water .Due to lack of large inland bodies of water, which makes this concept a suitable. This configuration allows rapid water collection. A sketch of concept 1 can be seen in Figure

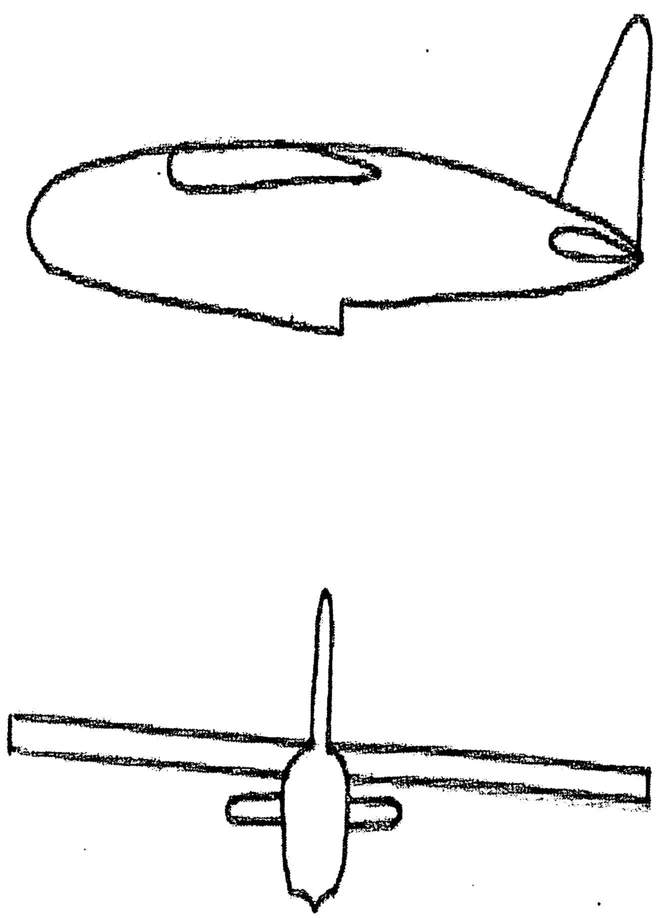


Figure-5.3 Concept- 1 Flying boat [Australian fire fighting report]

5.8.2 Concept 2(Floatplane)

The second concept considered was a floatplane configuration, where floats are attached to the fuselage of the aircraft to allow the aircraft to take off and land on water. Due to lack of large in land bodies of water, which makes this concept unsuitable .A sketch of concept 2 can be seen in Figure.

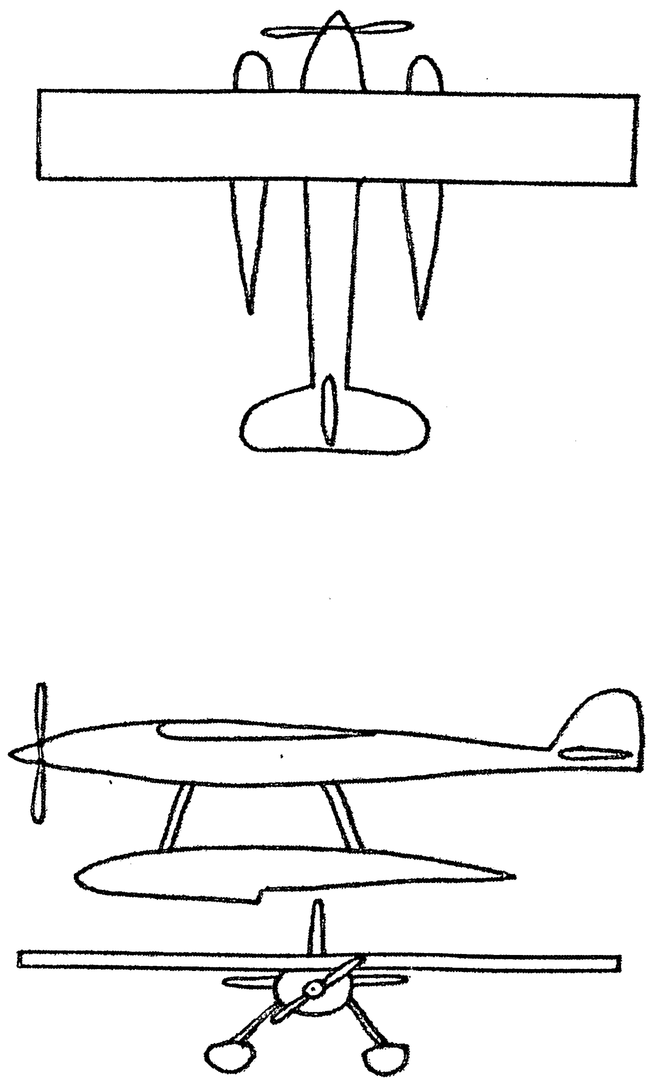


Figure-5.4 concept 2 (Floatplane) [Australian fire fighting report]

5.8.3 Concept 3 (Twin- engine aircraft)

The third concept considered was a twin-engine aircraft. Two engines increase the reliability of an aircraft, but the maintenance and running costs are higher than a single engine aircraft. A single turboprop can produce the required thrust for the aircraft, so a twin-engine aircraft was disregarded. A sketch of concept 3 can be seen in Figure

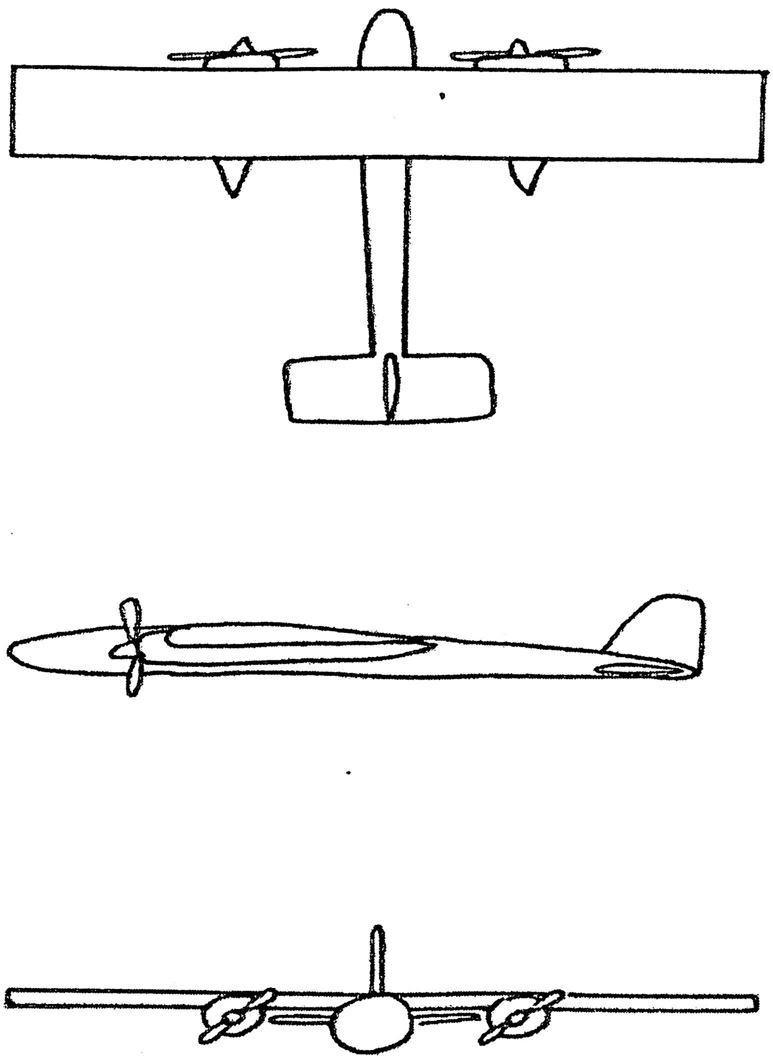


Figure-5.5 concept 3 (Twin engine aircraft) [Australian fire fighting report]

5.8.4 Concept 4 (Conventional aircraft)

The fourth concept considered was a conventional aircraft with a low wing configuration. Although most agricultural aircraft have a low wing configuration, the wing location decreases stability and ground visibility. Hence, a low wing configuration was disregarded. A sketch of concept 4 can be seen in Figure

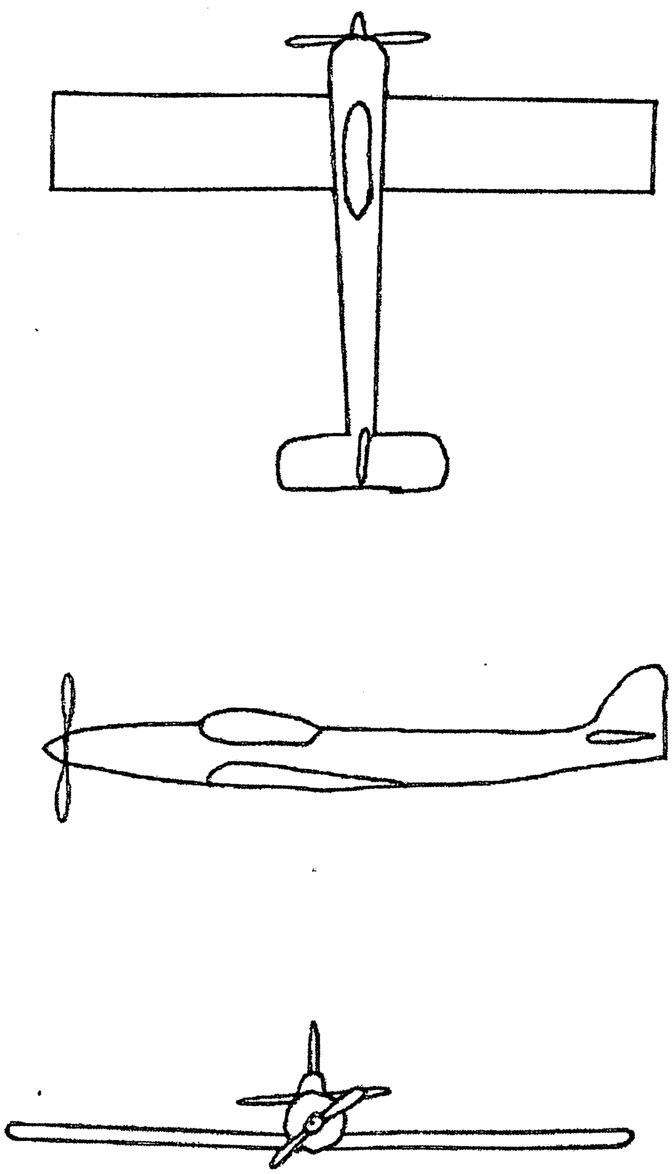


Figure-5.6 Concept-4 (Conventional aircraft)[Australian fire fighting report]

5.8.5 Concept 5 (Conventional aircraft with high wing)

The final concept that was considered by the group was a conventional aircraft with a high wing configuration. This design has high stability and ground visibility, which are two important considerations for a fire-fighting aircraft. A sketch of concept 5 can be seen in Figure

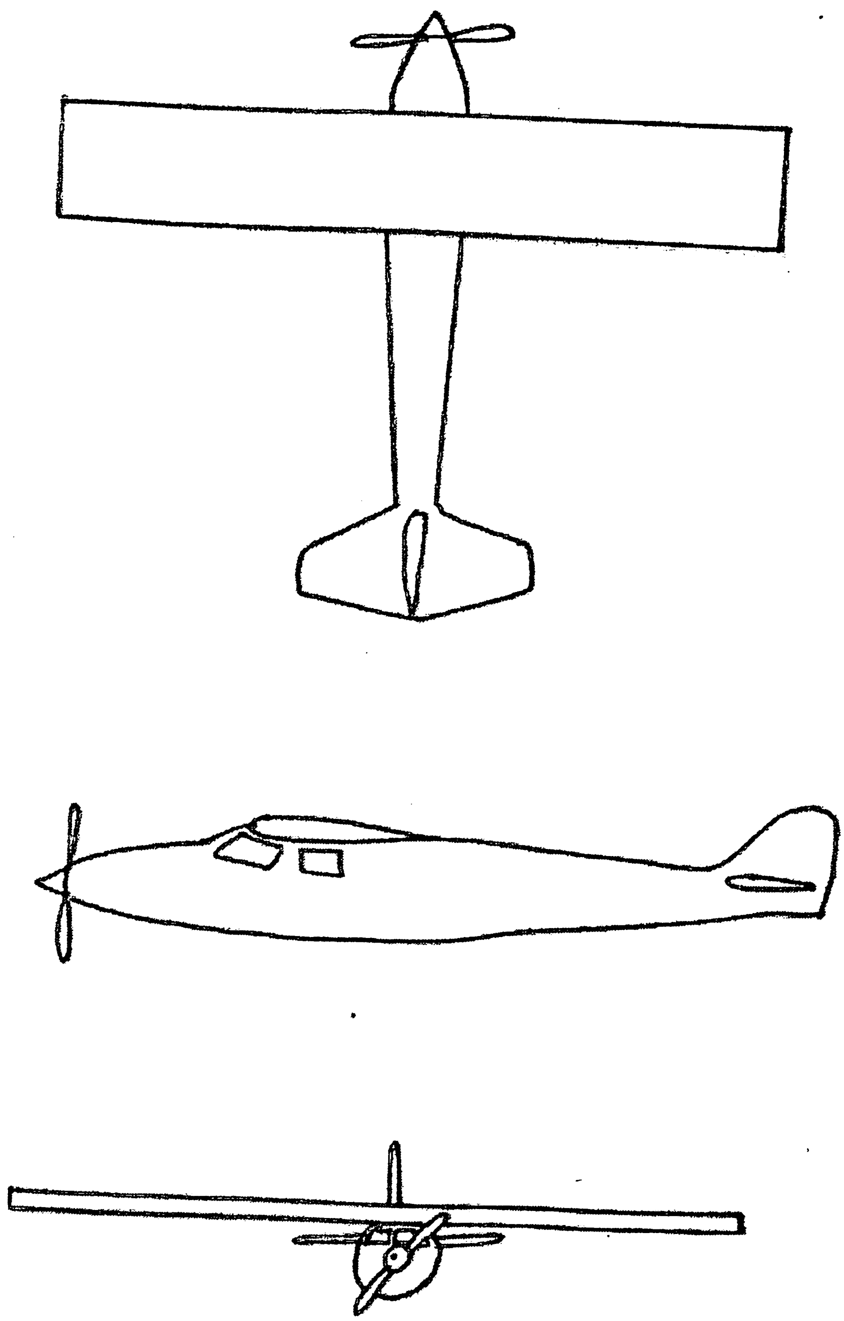


Figure-5.7 Concept-5 (Conventional aircraft with high wing) [Australian fire fighting report]

CHAPTER 6

DESIGN CONSIDERATIONS

Table presents the design considerations that were considered in the first step of the aircraft Configuration design.

Table 6.1 - Design Considerations

Consideration	Reasoning
Low aerodynamic efficiency	A strong structure is more important than aerodynamic efficiency
Metallic structure	Exposure to high temperatures which can damage composite materials
Operation in harsh environments	Exposure to high temperature, humidity and wind speeds
High cruise velocity	Required to reach the fire quickly
High manoeuvrability	Required to avoid obstacles, negotiate undulating terrain and line up for release of payload
Ability to fly at low altitude	Payload is released at low altitude
Retractable landing gear	Cruise speed is greater than 150 knots
Single tractor turboprop propulsion configuration	Ease of maintenance, reduced weight, increased reliability and reduce cost
Simple wing planform	Light weight, and cheap and easy to manufacture
High wing configuration	High ground visibility, ease of payload loading, high lateral stability, good structure
Raised cockpit	Increased ground visibility
Long nose	Payload placement and engine integration
Conventional empennage configuration	Light weight, and cheap and easy to manufacture

CHAPTER 7

AIRCRAFT SIZING

7.1 Fuel Fraction Estimates

Fuel fractions for phases 0-1, 1-2, 3-4, 8-9 were estimated using statistics for agricultural aircraft. Fuel fractions for phases 2-3, 4-5, 6-7,7-8 were calculated based on mission profile requirements. The mission fuel fraction was then calculated from the individual phase fuel fractions. The results are shown in Table 7.1

Table 7.1-fuel fraction

Phase	fuel fraction
Take off (phase 1)	W_1/W_0= .97 (raymer)
Climb (phase 2)	W_2/W_1 = .985 (raymer)
Cruise (phase 3)	W_3/W_2 = .987 (calculated)
Decent (phase 4)	W_4/W_3= .999(assume)
Loiter (phase 4)	W_5/W_4= .988 (calculated)
Climb (phase 6)	W_6/W_5= .985
Cruise (phase 7)	W_7/W_6= .987
Loiter (phase 8)	W_8/W_7= .997
Landing (phase 9)	W_9/W_8 = .995
Total fuel fraction	W_x/W_0 = .8976

7.2 Takeoff weight and empty weight estimation

The takeoff weight of the aircraft is estimated from a takeoff weight component breakdown and the technology diagram. This is achieved by solving Equation 2 and Equation 3 simultaneously for takeoff weight.

$W_{takeoff} = W\ crew + W\ payload + W\ fuel + W\ empty$

Hence ,

$W_0 = (W_{crew} + W_{pl}) / [1 - (W_f / W_0) - (W_e / W_0)]$ (eq. 7.1)

$W_{crew} = 85\ kg + 15\ kg = 100\ kg = 222.2\ lbs$

$W_{payload} = 2000kg = 4444.44\ lbs$

$W_{fuel} (W_f / W_0) = 1.06 (1 - W_x / W_0)$
$= 1.06 (1 - .8976)$
$= .1085$

$W_{empty} (W_e / W_0) = A\ W_0{}^{c}\ kvs$ (eq. 7.2)

Where,
A = .74 (raymer)
C = -.03(raymer)
Kvs = 1 (for fixed wing)

After putting all the values in eq. 7.1and doing iteration various time

Takeoff weight and Empty weight was estimated –

W_0 (takeoff weight) = 7577kg = 16726.26 lbs

$W_{e\ (Empty\ weight)}$ = 4288 kg = 9454.54 lbs

CHAPTER 8

FUSELAGE DESIGN

The purpose of the fuselage is to attach the wings and empennage, as well as the cockpit, motor, payload, and landing gear. The challenge with designing a fire-fighting aircraft is the requirement for the payload to be located directly on the centre of gravity to ensure that when the payload is released, there are no significant changes in the stability of the aircraft. The other design consideration is to ensure that the required components of the aircraft can all fit within the fuselage. For the fire-fighting aircraft, the required components include the cockpit, the motor, the front and rear landing gear, the wing attachment, tail attachments, and the payload and payload distribution system. As the landing gear, wing location and the payload location are all determined by the location of the centre of gravity, determining the size and layout of the fuselage is an iterative process.

8.1 Total length of fuselage

$L = a\ W_0\ ^c$(raymer)

Where,

$a = 4.04$

$c = 0.23$

hence total length = 37.5 ft

8.2 Overall Design of the Fuselage

The final layout of the fuselage is shown in Figure below.

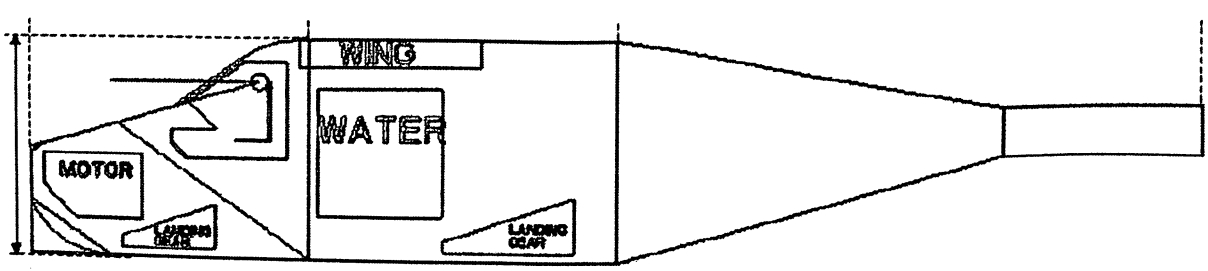


Figure 8.1 Fuselage design [Australian fire fighting report]

Using these dimensions, the maximum width of the fuselage was determined to be 7 ft as shown in Figure This value was selected based upon the required space for the storage of the retardant, the width of the cockpit required for the comfort of the pilot, and also based upon the aesthetics of the aircraft.

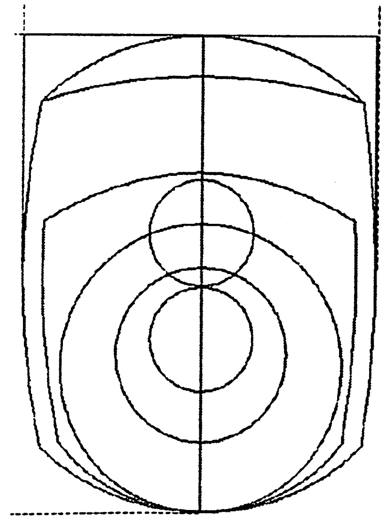


Figure 8.2 front view of the fuselage [Australian fire fighting report]

The overall length of the fuselage, and the length of the nose and tail sections, is dictated by the fineness ratio. It is desirable to adhere to these recommended fuselage parameters to reduce friction drag. The recommended fineness ratios for sub-sonic flight are given by Roskam (2004), and are shown in Table 8.1

Table 8.1 - Fineness Ratio [Roksham 2004]

	Fineness Ratio	Recommended Range
Total Fuselage	L_F/ D_F F	6-9
Cone	L_{FC}/D_F	2-3
Nose	L_{FN} / D_F	1.2-2

- The desired length of the fuselage and fuselage sections is dependent on the diameter of the aircraft. This implies that an iterative process is required to determine the optimum solution. The main driving parameter in determining these dimensions is the aircraft nose. The nose section of the aircraft contains the majority of the aircraft components including the cockpit, the nose landing gear, the motor (and associated air intake and outlet pipes), and a firewall to separate the cockpit from the engine. Once this layout was sufficiently established, the height of the aircraft could be determined, and using this along with a reasonable aircraft width, the fuselage proportions could be determined.

The final dimensions were determined to be as follows:

- Fuselage height: 84 inches (7 ft)
- Fuselage width: 66 inches (5.5 ft)
- Fuselage overall length (LF) : 450 inches (37.5 ft)
- Nose length (LN) : 102 inches (8.5 ft)
- Cone length (LC): 225 inches (18.75 ft)

The nose length was dictated by the constraints of the motor, nose landing gear and the cockpit, whilst the overall length was kept to a minimum and the cone length maximized and to minimize the weight of the aircraft. This was possible as all loads, excluding the structure and the empennage, are located in the foremost half of the aircraft.

The 'diameter' DF used to determine the fineness ratio was taken to be the average of the fuselage height and the width.

$D_F = (7+5.5)/2 = 6.25$ ft

Table 8.2 - Comparison of the Fineness Coefficient for the Designed Aircraft Compared with the recommended values [Roksham 2004]

	Fineness ratio	Recommended range
Fuselage	37.5/6.25 = 6	6-9
Cone	1 8.25/6.25 =3	2-3
Cone	8.5/6.25= 1.36	1.2-2

8.3 Fire Retardant Tanks and Distribution System

The retardant tanks are located on the centre of gravity. The system itself is required to drop 1873.25L of retardant, which equates to a total space envelope within the fuselage of $2*1*1m^3$ for the retardant alone. The space envelope within the fuselage allowed is distributed about the centre of gravity, is $2.2*2.8*4.1m^3$ to allow for sufficient room for tank structure and baffles to prevent the effects of sloshing. To further allow for the distribution system, including the payload bay doors to release the retardant, additional space has been left around the fuselage tank. This is shown schematically in Figure 28.

It is intended that the tank can be split into components to allow for the distribution of the retardant as required. Either an off-the-shelf or custom built distribution system could be accommodated within the provided space.

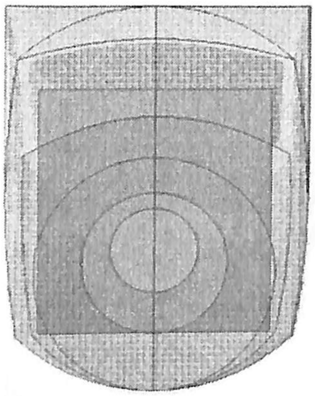


Figure 8.3 - Tank Location in the Fuselage

8.4 Fuselage Structure

Table 8.3 shows the frame depths, frame spacing and longeron spacing for a small commercial aircraft as specified by Arjomandi (2009).

Table 8.3- Recommended Frame and Longeron Spacing, and Frame Depth for a Small Commercial Aircraft as specified by Arjomandi (2009)

Frame depth (inches)	1.25-1.75
Frame spacing (inches)	24-30
Longeron spacing (inches)	10-15

By considering each section of the fuselage separately, the appropriate frame spacing could be determined. The frame spacing in the foremost half of the fuselage are primarily dictated by the locations of the wing leading and trailing spars, as well as the fire wall. The spacing of the formers around these components was designed to remain within the range specified above.

A firewall is located on an angle of approximately 35 degrees from the horizontal. This angle is required to allow sufficient room for the air outlets for the motor, and to accommodate the landing gear location. The main components in the nose of the fuselage (the motor and the landing gear) are usually attached to the firewall and supported using truss structures. When designing the nose of the aircraft, sufficient space was required to ensure the structure would fit inside the fuselage. The longerons were similarly placed depending on the size and shape of the fuselage.

CHAPTER 9

WING DESIGN

The following section of the report details the wing design. The geometry of the wing, including vertical position, sweep, aspect ratio, thickness ratio, taper ratio, twist, dihedral, wing loading, incidence angle and longitudinal position are considered. An aerofoil selection is summarized, followed by control surface sizing, wing tip selection and a summary of the wing structure.

9.1 Vertical Position

An aircraft can have three main vertical positions for the wing. A high wing is mounted above the fuselage, a low wing is mounted below the fuselage and a mid wing is mounted through the centre of the fuselage. An important consideration for a fire-fighting aircraft is ground visibility. A high wing configuration offers the best ground visibility. A further consideration is the loading and unloading of fire retardant. High wing aircraft are preferred for cargo applications, as no special equipment is needed for loading and unloading. A high wing configuration has high lateral stability and a lighter structure, as the internal volume of the fuselage is not cut by wing spars and other structural elements.

Incorporating landing gear into a high wing aircraft is often difficult as a large bay is required inside the fuselage for the retractable landing gear. The problem can be overcome by designing an appropriately sized area within the fuselage for the retracted landing gear. High wing aircraft are also less survivable during crashes in comparison to low wing aircraft. However, there are no passengers on board a fire-fighting aircraft and the aircraft is flown by an experienced pilot. Hence, crashworthiness is considered a minor issue. The aircraft fuselage will be designed to bear the impact loads generated by the wing in a crash.

The high wing configuration has many advantages over a mid wing configuration and a low wing configuration. The disadvantages of a high wing configuration were considered reasonable for the application. Hence, a high wing configuration was chosen for the fire-fighting aircraft.

9.2 Aspect Ratio

Aspect ratio is defined as the square of the wing span divided by the wing area. A high aspect ratio wing has low induced drag, a high lift-curve slope, good runway visibility from the cockpit and a higher span. However, a high aspect ratio wing has decreased ride quality through turbulence. High aspect ratios lead to steeper lift-curve slopes such that aircraft are more sensitive to changes in angle of attack. Hence, the ride quality of the aircraft is reduced. However, the aircraft is not a passenger aircraft. Hence, ride quality is considered a minor issue.

High aspect ratio wing require longer structural supports which corresponds to a higher overall wing weight, and experience low aero elastic stability. The aircraft has a cruise velocity such that aero elastic stability effects would be low. Additionally, high aspect ratio wings have low lateral stability. However, the fire-fighting aircraft has a high wing configuration, and as such, has a high lateral stability.

The advantages of low induced drag and good runway visibility were seen as significant. Hence, a high aspect ratio was chosen for the fire-fighting aircraft.

An average aspect ratio of 7.5 was calculated from the statistical analysis.

9.3 Sweep

Wing sweep is defined as the angle between the leading edge of the wing and the perpendicular to the fuselage. Wings can either be swept or un swept, depending on the application. An unwept wing has low weight, as the wing does not require additional structural supports, and exhibits good stall behavior. A un swept wing also has good runway visibility, as sweep reduces the lift-curve slope, which causes the aircraft to have more pitch attitude. Additionally, unswept wings are cheap and easy to manufacture, as all structural components are simple and all wing ribs can be made the same.

A swept wing reduces compressibility drag. However, the fire-fighting aircraft only has a cruise velocity of 350 kph, and such, compressibility effects would be marginal. A swept wing has higher longitudinal stability, as the sweep allows the aerodynamic centre to move faster than the centre of gravity. Additionally, sweep changes the longitudinal moment arm, which has a beneficial effect on the inherent longitudinal damping characteristics of the aircraft. Swept wings have increased ride quality. However, there are no passengers on board the fire-fighting aircraft and the aircraft is flown by an experienced pilot. Hence, ride quality is considered a minor issue.

The advantages of low weight, good structure, good stall behaviour and ease of manufacture were seen as significant. Hence, an unswept wing was chosen for the fire-fighting aircraft.

9.4 Thickness Ratio

Thickness ratio is defined as the maximum thickness of the wing divided by the chord length of the wing. A thick wing is lightweight due to the increased bending and torsional stiffness, and provides maximum lift coefficients. A thick wing can accommodate more fuel volume but has higher profile drag in the subsonic flight regime.

The advantages of a lightweight and maximum lift were seen as significant. A high thickness NACA

4415 aerofoil was chosen for the fire-fighting aircraft .This aerofoil has a thickness of 15%, which is a suitable value for obtaining maximum lift coefficients.

9.5 Taper Ratio

Taper ratio is defined as the ratio of the tip chord to the root chord. Low taper ratio reduces the weight of the wing as the wing lift distribution tends to zero at the wing tip and the area of the wing near the wing tip is not fully loaded. A wing with a taper ratio of one is also cheap and easy to manufacture, as all structural components are simple and all wing ribs are the same. The wing tip of a low taper ratio wing tends to stall sooner as it flies on lower Reynolds's number airflows, and has a lower maximum lift coefficient. Additionally, a high taper ratio increases the amount of fuel that can be stored in the wings. However, a thick wing was chosen to negate these issues.

The advantages of reduced wing tip stall and ease of manufacture were seen as significant. Hence, no taper was chosen for the fire-fighting aircraft.

9.6 Twist

Wing twist occurs when the tip aerofoil has a lower or higher angle of incidence than the root aerofoil. Wings that have no twist are easy and cheap to manufacture, as all structural components are simple and all wing ribs can be the same. Wings that have no twist have decreased induced drag. However, wings that have no twist experience wing tip stall that can generally occur in an asymmetric manner and cause serious roll control problems. However, a thick wing was chosen to provide high maximum lift coefficients to negate this problem.

The advantages of decreased induced drag and ease of manufacture were seen as significant. Hence, no wing twist was chosen for the fire-fighting aircraft

9.7 Dihedral

A high wing configuration has an inherent dihedral effect that causes the rolling moment due to the sideslip derivative to be negative. This means that the aircraft has more spiral stability and less dutch roll stability. Hence, no dihedral was chosen for the fire-fighting aircraft as it has a high wing configuration.

Low wing loading provides a shorter takeoff and landing distance, but requires a larger wing area that increases the weight of the wing. Short takeoff and landing distance is not considered an important issue as the aircraft is designed to operate out of paved runways. Low wing loading is used for aircraft that are required to fly at high altitude, which is not an important parameter in the design of a fire fighting aircraft. Also, low wing loading results in a higher response to changing angle of attack which corresponds to poor ride quality. However, there are no passengers on board the aircraft and the aircraft is flown by an experienced pilot. Hence, ride quality is not considered a major issue

9.8 Wing Loading

High wing loading allows the cruise lift coefficient to be similar to that at (L/D)max. A high wing loading also requires the aircraft to resist higher accompanying stresses. Hence, high wing loading increases the cost and complexity of manufacture as it requires materials that are more expensive and more complex manufacturing methods.

Wing loading at max. C_L was calculated at stall speed –

$W/S = .5 \square V^2_{stall} C_{lmax}$

Now,

Stalling speed = 154 ft/s (prototype)

Staling altitude = 5000ft

Density = .00189 slug / ft^3

C_{lmax} = 1.6 (estimated from raymer)

After putting all the values we get

$W/S = 35.8$ lb / ft^2

9.9 Wing Longitudinal Location

From the statistical analysis, the average wing leading edge location as a percentage of the fuselage length was determined to be 26%. Hence, this value was used for preliminary sizing. The initial fuselage length was 37.5 ft, which gives the wing leading edge location from the nose of the aircraft as 9.75 ft. throughout the design process. this was modified to correspond with the geometry of the aircraft. Hence, the wing longitudinal located was adjusted to be 8 ft from the nose of the aircraft

9.10 Aerofoil Selection

The design of the aerofoil section for the wing is critical for ensuring the aircraft can achieve the required performance. The shape of the aerofoil affects the lift and performance of the aircraft in all flight regimes, including cruise, takeoff and descent (Raymer 2006).

9.10.1 Operational Reynolds Number

L = Cwing = 8.59 ft. VTO = 153.17 ft/s.

v = 1.57 x 10-4 ft2/s at sea level.

Re = VTO L/v = (153.17)(8.59)/(1.57 x 10-4).

→ Re = 8.38 x 106.

The aerofoil must be suitable for operation in airflow with a Reynolds's Number Re = 8.38 x 106.

9.10.2 Maximum Lift Coefficients-

For an agricultural aircraft, CL max = 1.3 - 1.9 (Roskam 2005). Hence, an average value of 1.6 will be chosen as the preliminary CL max.

For an untwisted, constant-aerofoil-section wing, CL max/Cl max = 0.9 (Raymer 1992).

→ Cl max = CL max/0.9 = 1.6/0.9.

→ Cl max = 1.78.

The aerofoil must be selected to provide the desired maximum wing lift coefficient CL max = 1.6 and the desired maximum aerofoil lift coefficient Cl max = 1.78. There are also some additional considerations as outlines below.

The aerofoil must have the highest possible (L/D)wing compared with similar aerofoils to allow the aircraft to achieve the highest possible (L/D)aircraft.

The aerofoil must have a low pitching moment coefficient Cm to reduce the torsional loads and induced drag from trimming

9.10.3 Design Lift Coefficient -

(W/S) = 35.8 lbs/ft2.

ρ_c = 0.00116355 slugs/ft3 at a cruise altitude of 22,500 ft. Vcr = 341.75 ft/s.

L = W = 0.5ρcrVc 2SC.

→ CL = (W/S)(1/0.5ρV2).

→ CL = (35.8)(1/(0.5*0.00116355*341.752).

→ CL = .32

The design lift coefficient CL = 0.32

9.10.4 Aerofoil Selection Process

The aerofoil selection process compared the two-dimensional flow performance of the aerofoil candidates over the range 0o < α < 20o. The two dimensional performance of the aerofoils differ from a three dimensional wing. However, a suitable indication of (L/D)wing can be obtained from two- dimensional data. For the purpose of aerofoil comparison, it was assumed that the aerofoil with the highest (L/D)aerofoil would produce the wing with the highest (L/D)wing.

Similarly, the aerofoil with the lowest, most constant section pitching moment coefficient Cm would produce the wing with the lowest, most constant pitching moment coefficient CM. JavaFoil (2009) was used to compare the performance and suitability of each of the selected aerofoils. The selected aerofoil profile was to have the properties as outlined below.

- High (L/D)aerofoil
- Low, constant Cm
- Cl max > 1.78 such that CL max > 1.78 after three dimensional correction

9.10.5 Aerofoil Candidates

Three possible aerofoil profiles were identified from research of the aerofoils used on existing agricultural aircraft. The aero foils are presented in Table 9.1 below.

Table 9.1 - Aerofoil Candidates

Aerofoil	Aircraft	Reference
NACA 4415	Air Tractor AT-301 through AT-802	UIUC 2008
NACA 4416	M-18A Dromader	UIUC 2008
NACA 4412	Grumman G-164 Ag-Cat	UIUC 2008
NACA 4415	Pacific Aerospace Cresco 08-600	UIUC 2008

9.10.6 2D Analysis

Table 9.2 shows a comparison between the selected 2D aerofoils.

Table 9.2- 2D Aerofoil Comparison Table

Aerofoil	Cl max	Cd	Approximately	Average Cm
NACA 4412	1.904	0.01340	Yes	-0.12
NACA 4415	2.184	0.01407	Yes	-0.13
NACA 4416	2.306	0.01696	Yes	-0.13

From Table 9.2, all the aerofoils provide the minimum desired Cl max value of 1.78, and they all have similar values for the pitching moment coefficient Cm. Hence, a 3D analysis is required to determine the most suitable aerofoil.

9.10.7 3D Analysis

3D flow effects cause wings to have lower lift coefficients than the 2D aerofoil lift coefficients. Consequently, a correction for 3D flows will be considered.

Table 9.3 shows the results for the 3D aerofoil analysis.

Table 9.3- 3D Aerofoil Comparison Table

Aerofoil	Cl max	Cd	(L/D)max	Approximately constant Cm	Average Cm
NACA 4412	1.535	0.10715	14.33	Yes	-0.12
NACA 4415	1.78	0.14341	12.42	Yes	-0.13
NACA 4416	1.859	0.15443	12.04	Yes	-0.13

The NACA 4412 does not achieve the desired Cl max value of 1.78, whereas the NACA 4415 and NACA 4416 both achieve the desired Cl max value. Both the aerofoils have similar values for the average Cm, but the NACA 4415 has a higher value for (L/D)max.

The NACA 4415 is the most appropriate aerofoil to choose for the fire-fighting aircraft, as it provides an appropriate value for Cl max, has a small Cm value and has a high L/D at a low angle of attack.

Cl max = 1.78.

For an untwisted, constant-aerofoil-section wing, CL max/Cl max = 0.9 (Raymer 1992).

→ CL max = 0.9*Cl max.

→ CL max = 0.9*1.78.

→ CL max = 1.6.

9.11 Wing plan form area

The wing area is calculated based on the weight and wing loading at the start of the cruise.

Hence

$(W)_{cruise} = .97*0.985*W_0$

$= 16000.309$ lbs

$(W/S)_{cruise} = q(3.14*A*e*C_{D0}/3)^{1/2}$ (from Raymer)

Now

A= 7.5

e = .8

C_{D0}=.03

putting all the values in eq. we get

$W/S_{cruise} = 46.78 lb/ft^2$

S = W /(W/S)

$= 16000/ 46.76 = 342.57 ft^2$

9.11.1 Wing span and chord

Wing span and chord can be calculated by using formula

$A.R. = b^2/S$

$b = (A * S)^{½}$

$b = (7.5 * 342.57)^{½} = 50.688 ft$

S = b*c

C = S/b = 6.75ft

9.12 Flap Selection

Most agricultural aircraft in operation utilize Fowler flaps. Hence, Fowler flaps were selected for the firefighting aircraft.

Fowler flaps provide ΔCl max = 1.3.

9.13 Incidence Angle

The wing incidence angle is calculated based on the two factors: the cruise drag and the floor attitude at cruise. The incidence angle should be chosen so that during the main part of cruise, the fuselage has no angle relative to the oncoming airstream. If the fuselage cruises nose up or nose down, the total drag of the fuselage is increased. The floor attitude in cruise is also influenced by the choice of incidence angle.

The following calculations show the process used to determine the wing angle of incidence. WTO = 16726.54lbs.

S = 342.56 ft2.

ρcr = 0.0015455 slugs/ft3. Vcr = 341.75 ft/s.

L = WTO = 0.5ρV2SCL. CL = WTO/(0.5ρcrVc 2S).

→ CL = (16726.54)/(0.5*0.0015455*(341.752)*342.56).

→ CL = 0.54

From the CLα curve, CL = 0.54 corresponds to 0 degrees angle of attack. Hence, the wing

9.14 Flow Control Devices

The aircraft wing has no sweep, so no loss of stability occurs at the wing tips due to the thickening of the boundary layer and airflow separation. Hence, no overall lift is lost at the wing tips, and ailerons are not affected. Hence, flow control devices are not required angle of **incidence is 0 degree**

9.15 Wing Tips

Table 9.4 summarises the advantages and disadvantages of different wingtips.

Table 9.4 - Wing Tip

Wing Tip	Advantage	Disadvantages
Rounded	Aesthetic	High induced drag
Sharp	Low induced drag	Difficult to manufacture
Cut off	Low induced drag, simple and cheap to	None
Hoerner	Low induced drag	Difficult to manufacture
Dropped	Increases effective span without	Difficult to manufacture
Upswept	Increases effective span without	Difficult to manufacture
Aft-swept	Low drag	Increases wing torsional
End plate	Prevents air flowing beneath the wing	High drag
Winglet	High drag reduction	Flutter, twist and camber must

Cut off wing tips are the simplest, cheapest and easiest wing tips to manufacture, and do not increase induced drag. Hence, the fire-fighting aircraft shall be designed with cut-off wing tips.

9.16 Centre of Gravity

The centre of gravity of the wing was determined by calculating the centroid of the NACA 4415 aerofoil, and was determined to be 42% of the wing chord.

$C_{wing} = 8.18$ ft.
$CG_{wing} = (0.42)(8.18)$.

$\rightarrow CG_{wing} = 3.4356$ ft from the leading edge of the wing.

9.17 Wing Design Summary

Table 9.6 summarizes the above wing design.

Table 9.5 - Wing Design Summary

Parameter	Value
Vertical position	High
Wing loading	35.8 lbs/ft^2
Area	342.56ft^2
Span	50.68 ft
Chord	6.75 ft
Sweep	0 degrees
Aspect ratio	7.5
Thickness ratio	15%
Taper ratio	1
twist	None
Dihedral angle	0 degrees
Incidence angle	0 degrees
MAC	6.75 ft
Aerofoil	NACA 4415

CHAPTER 10

EMPENNAGE DESIGN

10.1 Empennage sizing

Figure 10.1below shows some possible configurations for the empennage design. Raymer (2006) recommends the use of the conventional arrangement for conventional aircraft as the configuration will provide adequate stability and control at the lightest weight. Other configurations considered were the T-tail, the cruciform, the V-tail and the H-tail. The T-tail and the H-tail were not chosen, as they are heavier than the conventional configuration for an unnecessary gain in stability. The cruciform was not chosen, as it was not as stable as the conventional configuration. A conventional
tail configuration was chosen for the fire-fighting aircraft application.

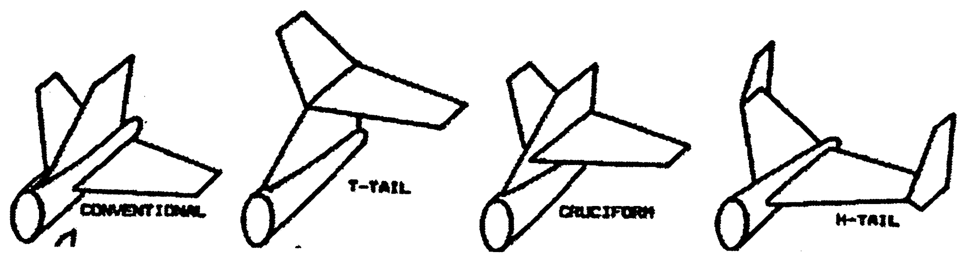


Figure 10.1 - Empennage Configurations [Raymer 2006]

The horizontal stabilizer is the component of the empennage that lies in the horizontal plane. A statistical approach was used to calculate the area of the horizontal stabilizer. The statistical approach involves the use of a tail volume coefficient and Raymer (2006) provides data for the parameters used. The aircraft is modeled as an agricultural aircraft and the volume coefficient VH was determined to be 0.5. The formula involves the reference area S, which is calculated from the aspect ratio. The distance between the MAC of the tail and the MAC of the aircraft was calculated from the stability analysis as 35ft and the chord of the wing as 8.18ft. The horizontal area was calculated as follows-

L_{HT} = 25 ft (from analysis)

$S_{HT} = (C_{HT}\,C_w\,S_w)/\,L_{HT}$ (eq. 10.1)

C_{HT} =0 .5 (from raymer)

C_w = 6.35 ft

S_w = 342.56 ft

Now putting the values in eq. 10.1 we get

S_{HT} = 46.246 ft^2

The vertical stabiliser is the component of the empennage that lies in the vertical plane. The vertical stabiliser was calculated in similar way. The tail volume coefficient was determined for an agricultural aircraft from Raymer (2006) to be 0.04.

The vertical area was calculated using formula

$S_{VT} = (C_{VT}\, b_w\, S_w)/ L_{vt}$

$C_{vt} = .04$

Putting all the values we get

$S_{VT} = 27.78$ ft

10.2Horizontal Stabilizer Geometry

The calculation of the horizontal stabilizer dimensions incorporates the tail aspect ratio and taper ratio. Raymer (2006) recommends that a horizontal stabilizer have an aspect ratio of 4.0 and a taper ratio of 0.4. The horizontal stabilizer will be configured as shown in Figure 10.2 below.

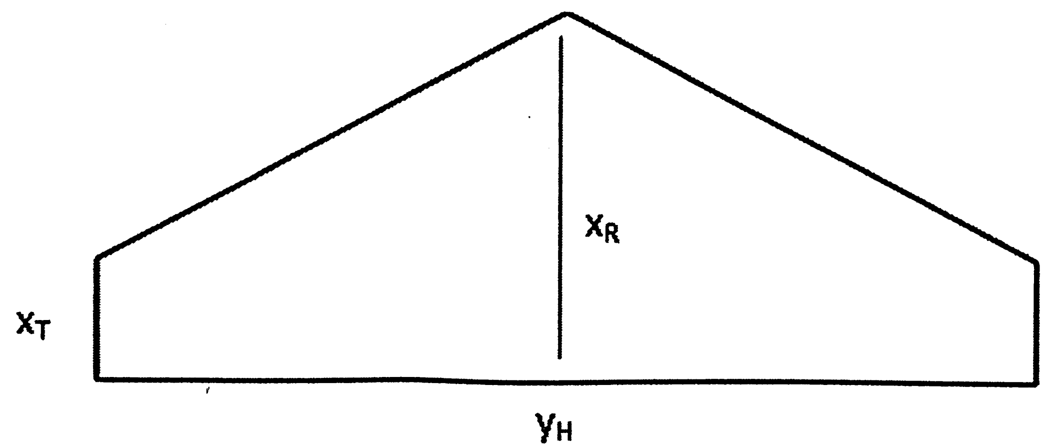


Figure 10.2 - Horizontal Stabilizer Arrangement [Roksham 2004]

The total area of the horizontal stabilizer is calculated as follows

$S_H = x_T y_H + 0.25\,(x_R - x_T)\, y_H$

$X_T = 0.4\, X_R$

$S_H = 0.55 X_R Y_H$

$X_{R} = 6.77$ft

$X_{T} = 0.4 * 6.77$

$= 2.77$ ft

$A.R = Y^{2}{}_{h}/ S_h$

$Y_H = 13.60$ f

10.3Vertical Stabilizer Geometry

The calculation of the vertical stabilizer dimensions incorporates the tail aspect ratio and taper ratio. Raymer (2006) recommends that a vertical stabilizer have an aspect ratio of 1.2 and a taper ratio of 0.4. The vertical stabilizer will be configured as shown in Figure 10.3 below.

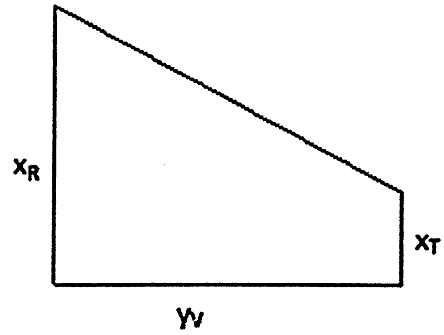


Figure 10.3 - Vertical Stabilizer Arrangement [Roksham 2004]

The total area of the vertical stabilizer is calculated as follows:

$S_V = X_T Y_V + 0.5\ (X_R - X_T)\ Y_H$
$X_T = 0.4\ X_R$
$S_V = 0.7\ X_R Y_V$

$A.R. = Y^2_v / S_v$
$Y_v = 5.773ft$
$X_R = 6.87ft$
$X_T = 2.74ft$

10.4. Elevator Sizing and Geometry

Raymer (2006) states that the ratio of the area of the elevators to the area of the horizontal tail is between 0.25 (for a jet transport) and 0.45 (for a general aviation aircraft). The ratio for this aircraft is 0.3, as fire-fighting aircraft require somewhat more control authority than a jet transport but less than a general aviation aircraft. The area of the elevators can now be calculated.

$S_{elve}/ S_H = 0.3$
$S_{elv} = 0.3 * 46.246$
$= 13.87$ ft

A trim tab will be placed in the elevator arrangement, and will be sized by a similar volume coefficient method. The volume coefficient for the elevator trim tab is 0.09 (Raymer2006).

$S_{trim} / Selve = .09$

$S_{trim} = 1.24ft^2$

Due to the position of the vertical stabilizer, there is a span wise area on the horizontal where an elevator cannot be placed. The thickness of the vertical stabilizer is chosen later in this section and the thickness to chord ratio is 13%.

The chord of the vertical stabilizer at the root was found to be 8.34 ft. This results in a width of 1.40 ft where no elevator can be placed. A gap of 6 inches is placed between the vertical tail infringement and the start of the elevator. The chord at this location is 5.21 ft.
The geometry of the elevator is shown in Figure 10.4 below:

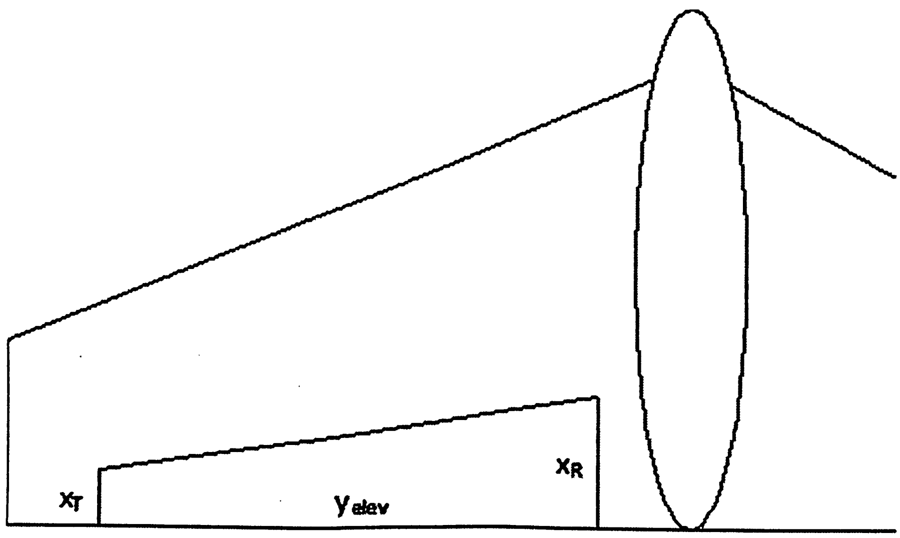


Figure 10.4 - Elevator Geometry [Roksham 2004]

The elevator is chosen to be 40% of the chord of the horizontal stabiliser. Using a similar approach to the stabiliser sizing, the elevator dimensions are now calculated.

$X_R = 0.4 * 6.77 = 2.71$ ft
$X_T = .4 * 2.77 = 1.09$ ft
$Y_{elev} = 4.174$ ft

10.5 Rudder Sizing and Geometry

Similar to the elevator sizing, Raymer (2006) states that the ratio of the area of the rudders to the area of the vertical tail is between 0.35 and 0.45. The ratio for this aircraft is chosen to be 0.4. The area of the rudders can now be calculated.

$S_{rud} / S_v = .4 * S_v$
$S_{rud} = 11.112 ft^2$

A trim tab will be placed in the rudder arrangement, and will be sized by a similar volume coefficient method. The volume coefficient for the rudder trim tab is 0.09 (Arjomandi 2009).

$S_{trim} / S_{rud} = .09$
$S_{trim} = 1.0008 ft^2$

The geometry of the rudder is shown below

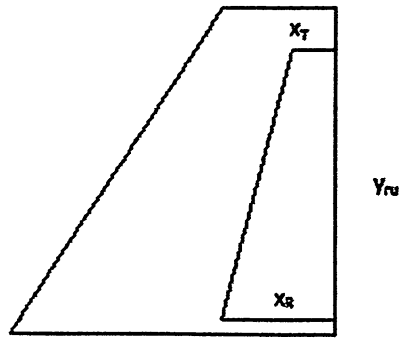


Figure 10.5 - Rudder Geometry[Roksham 2004]

The rudder is calculated to be 40% of the chord of the vertical stabiliser. Using a similar approach to the stabilizer sizing, the rudder dimensions are now calculated.

$X_R = .4 * 6.87 = 2.74$ ft
$X_T = .4 * 2.74 = 1.096$ft
$Y_{rud} = 6.64$ft

CHAPTER 11

LANDING GEAR DESIGN

Landing gear placement is essential for ground stability and controllability. A good landing gear position must provide superior handling characteristics and must not allow over-balancing during takeoff or landing.

11.1 Landing gear arrangement

Landing gear arrangements are included in Figure 11.1 below. The two most common landing gear arrangements for high-wing designs are the tail-dragger and tricycle arrangements (Raymer 2006).

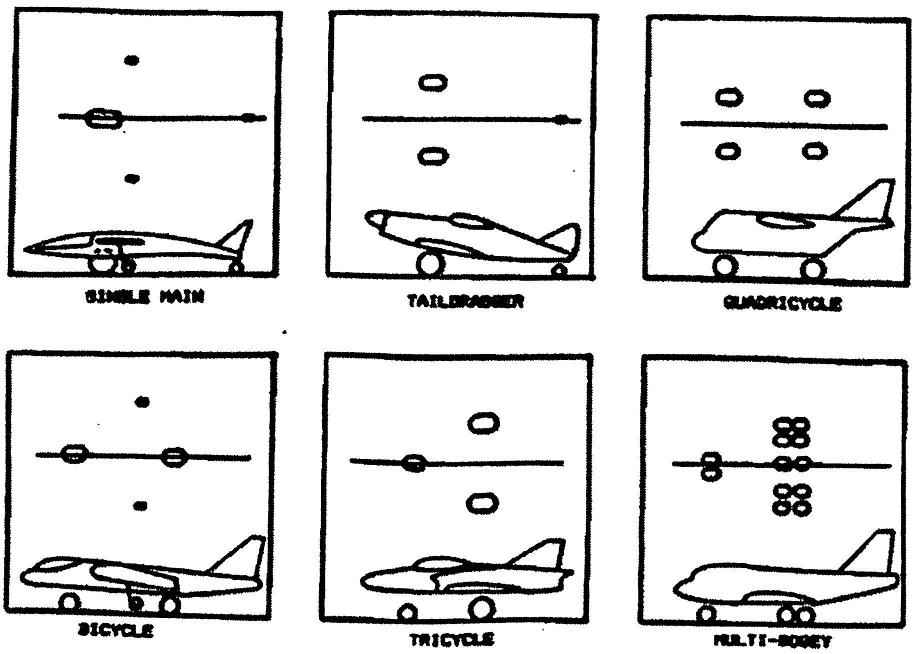


Figure 11.1 – Landing Gear Configurations (Raymer 2006)

Bicycle and single main landing gear arrangements are less preferable due to the inherent instability on the ground. Outrigger wheels are required on the extremes of the aircraft, and the high-wing configuration makes the placement of these difficult (Raymer 2006). The outrigger wheels would need to be long to reach from the wing to the ground. The weight of these outrigger wheels would be significant, an the storage of them difficult. The quadricycle arrangement would involve a significant increase in weight in comparison to the tricycle and tail-dragger arrangements. The stability is increased significantly due to the wheel locations and the loads on each wheel are reduced due to the added wheel (Raymer 2009). The quadricycle arrangement is not considered due to the width required in storing the landing gear in the fuselage when the gear is retracted. The fuselage design is not of sufficient width to house all four landing gear.

Both the tricycle and the tail-dragger arrangements are used for high wing aircraft. The tricycle gear arrangement provides good steering and ground stability characteristics. The advantage of a flat cabin floor allows for good visibility take-off and during approach as well as the ability to store and load cargo horizontally. The advantages of flat storage and loading of cargo are not applicable to the fire-fighting application. The tail-dragger allows an increased angle of attack at take-off and landing (Torenbeek 1982). This decreases the take-off and landing distances for the aircraft in comparison to a tricycle gear. Tail-dragger gears are typically smaller, are thus lighter, and require less storage space in the fuselage (Raymer 2006). Tail-dragger arrangements are unstable during turning man oeuvres on the ground, due to the centre-of-gravity being located behind the main landing gear. This significant decrease in stability was considered prohibitive to this design.

A tricycle arrangement was chosen for this configuration due to its good stability and steering, as well as good visibility.

11.2Landing Gear Sizing Nomenclature

Figure 11.2below shows the nomenclature used throughout the landing gear sizing section of this report. All symbols are defined in the nomenclature list at the beginning of this report.

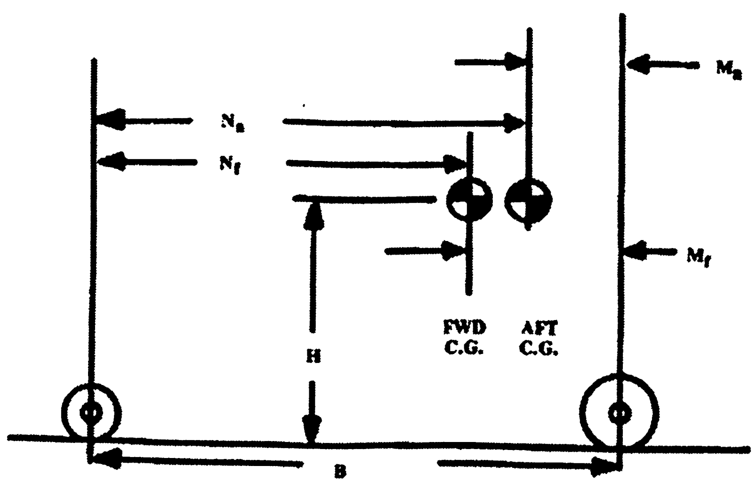


Figure 11.2 - Landing Gear Nomenclature (Roskam 2006)

11.3Landing Gear Placement Criteria

Raymer (2006) gives five criteria for locating the landing gear on the aircraft. These criteria are outlined below:

- ➢ The nose weight criterion
- ➢ The height criterion

11.4Nose weight criterion

The nose weight criterion ensures that the correct proportion of weight is carried by the nose gear. The nose wheel is required to carry more than 5% of the aircraft weight at take-off and after landing. This allows enough traction on the tire of the nose-wheel to permit nose-wheel steering (Raymer2006). The proportion of loads on the nose wheel should be less than 20%. An increased proportion of weight on the nose wheel results in a more difficult take-off as a larger speed is required to create the lift required for takeoff rotation (Torenbeek 1982). The upper limit of 20% on this criterion allows for a reasonable takeoff speed (Raymer 2006).

11.5 Height Criterion

The height criterion ensures that there is sufficient clearance for the fuselage and propeller including required safety clearances. The landing gear calculations can determine the vertical height of each gear. This height is measured from the ground to the centre of gravity of the fuselage. The height of the landing gear must be greater than the vertical distance between the centre of gravity of the fuselage and the bottom of the fuselage at the landing gear attachment point. Further, it is required that the height of the nose landing gear allow enough height for proper rotation of the propeller. The propeller diameter will be 10ft but is not located at the vertical centre of the fuselage. A propeller clearance of at least 7” is required for safety purposes (Arjomandi 2009). From preliminary drawings, the distance from the ground to the centre of the propeller disc is 2.85 ft.

The over-turn angle criterion regards ground stability during taxiing. According to Raymer (2006, pg232), the over-turn angle is “measured as the angle from the [centre of gravity] to the main wheel, seen from the rear at a location where the main wheel is aligned with the nose wheel”. This dimension is illustrated in Figure 21 below.

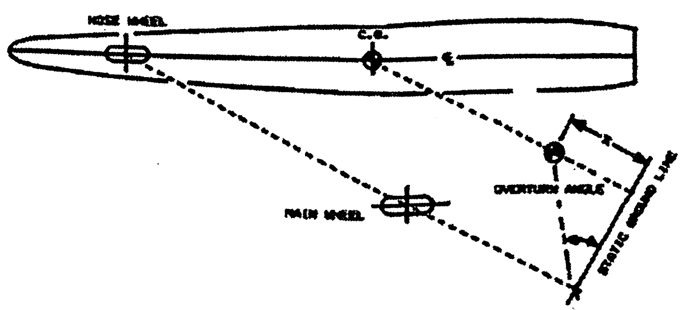


Figure 11.3 - Over-turn Angle Criterion (Raymer 2006 p. 232)

11.6Landing Gear Position

The correct landing gear position is found by beginning with an initial configuration, and iterating the calculations until all four criteria are met. The following design parameters are used in verifying the initial configuration.

An iterative approach was used to find a landing gear position that meets all four criteria. The following design point was proposed:

Distance from nose to nose gear = 4.1 ft

Distance from nose to landing gear = 18 ft

The following calculations verify that this design point meets all two criteria.

11.7Nose Weight Criterion

2.35

$M_F / B = 2.37 / 13.9 = 0.17$

2.1

$M_A / B = 2.1/13.9 = .15$

Thus, the nose weight criterion is satisfied.

11.8Height Criterion

The height of the aircraft is determined by solving a quadratic equation, which can be derived by simultaneously solving two equations that are a result of the geometry. The quadratic is as follows:

$$\left(\frac{D_{fuselage}}{2} + H\right)(D_{fuselage} + H) - M_f(L_{fuselage} - B) = 0$$

Using our parameters, and taking the only positive root:

$H^2 + 12.45H - 47.32 = 0$

$H = 3.04$

The height is less than the height required by the height criterion, so the minimum height is used. H = 3.43

11.9Number, Type and Size of Tires

The tricycle configuration has three contact points on the ground. Two wheels will be used at each contact point to minimize the effect of a flat tire. It is common to use two wheels at each point for this reason (Torenbeek 1982). For a fire-fighting application, the heat involved will reduce the life of the tires, and the instances of flat tires may be more numerous.

The weight that each tire will need to support can be determined from the following equation:

Weight on wheel (main) = maximum static load / no. of main gears wheel = 17929/ 4

= 4482 lb

Weight on wheel (nose) =(maximum static load + breaking load) / 1.4 * no. nose gear wheel

= (3801 + 1501) / 1.4 * 2 = 1894 lb

Wheel diameter is calculated by using formula –

$D = AW^B{}_w$ (Raymer)

For main wheel

$W_{w} = 4482lb$

A = 1.59 (table in Raymer)

B = .302 (table in Raymer)

Putting the value we get

D = 20.14 inch

For nose wheel

$W_w = 1894$

A = 1.59

B = .302

D = 15.52 inch

Wheel width is calculated by the formula –

$w = AW^{B}{}_{w}$ (raymer)

Where

A = .098

B = .407

Hence

Putting the values in above equation

$Width_{main} = 4.97inch$

$Width_{nose} = 3.32$ inch

Based on the above result following table was selected from raymer

Raymer (2006) recommends the use of Type III or Type VII tires for traditional aircraft. Type III tires are used on aircraft with piston engines and Type VII tires are used on aircraft with jet engines. Type VII tires will be used for this application and are selected from Raymer (pg 235, 2006).

Table 11.1 - Tire Selection Table

Nose wheel tyres (2 of)								
Size	Speed (knots)	Max load (lb)	Inflation (psi)	Max Width (in)	Max Diameter (in)	Rolling Radius (in)	Wheel Diameter (in)	Number of plies
18x4.4	174	2100	100	4.45	17.90	7.9	10.0	12
Main gear tyres (4 of)								
Size	Speed (knots)	Max load (lb)	Inflation (psi)	Max Width (in)	Max Diameter (in)	Rolling Radius (in)	Wheel Diameter (in)	Number of plies
24x5.5	174	11500	355	5.75	24.15	10.6	14.0	16

CHAPTER 12

PROPUSION SYSTEM DESIGN

Propulsion system design is an essential component of aircraft design. Propulsion system design involves the decision to manufacture or purchase a pre-existing engine, followed by the selection of the engine model and design integration. This process may flow systematically, but the conflicting input from many subsystems often causes the process to be iterative. This iterative process is amplified by the sensitivity of the propulsion system to weight. Increases in weight may result in the selection of a different engine model or even an increase in the number of engines at later stages in the design.

12.1 Propulsion System Type Selection

The selection of an optimal engine is fundamental for a successful propulsion system design. Engines available for selection include piston, Wankel, rotary, radial, electric, turboprop, turbojet, turbofan, ramjet and scramjet engines. The cruise speed of the aircraft critically affects the selected engine type and is specified by the technical task. The selected engine type is largely independent of the design of other systems such as weight, aerodynamics and structures, and consequently these factors will be neglected when investigating engine type. Hence, engine type can be selected considering only constraints from the technical task. Constraints due to other systems or aircraft configurations can be neglected.

The technical task specifies a maximum speed of no less than Vmax = 202.5 knots (341.8 ft/s) and a cruise altitude of 8000 ft. At this altitude, the speed of sound a = 1085.3ft/s.

Therefore, the Mach number can calculated as follows:

$M = v/a$

$M = 341.8 / 1085.3$

$M = 0.322$

The primary selection criteria for engine type include the following:

- Suitability to aircraft operating envelope (including technology level, required power, operating ceiling and cruising speed)
- High thrust to weight ratio at flight mach number.
- Low Thrust Specific Fuel Consumption (TSFC) at flight mach number.

These criteria will be addressed in the following sub-sections.

Technical Task Requirements

The technical task does not outline any requirements regarding the propulsion system type or number of engines.

Suitability to Aircraft Operating Envelope -

Some engines listed in Section 12.1 can be eliminated, as they do not satisfy the conditions outlined by the operating envelope of the aircraft. These are listed below:

- Rotary Engine: Technology level has been surpassed, and are considered very heavy and aerodynamically inefficient
- Electric Engine: Does not satisfy the power requirement for a fire-fighting aircraft, and are best suited for UAV or RC aircraft
- Ramjet Engine: Requires the aircraft to be travelling at Mach numbers, M > 3 to initiate combustion. As the maximum speed of the aircraft is orders of magnitude below the initiation speed, a ramjet engine will not be considered for this application.
- Scramjet Engine: Requires the aircraft to be travelling at Mach numbers, M > 5 to initiate combustion. As the maximum speed of the aircraft is orders of magnitude below the initiation speed, a ramjet engine will not be considered for this application.
- Therefore, the remaining engines to be considered are Wankel, radial, turboprop, turbojet and turbofan. Figure 5.4 in Brandt (2004, p. 178) shows that for a Mach number, $M \approx 0.3$ and altitude h=8000 ft, a reciprocating propeller is the preferred engine type followed by turboprop, turbofan and turbojet engine.

Thrust to weight ratio -

The highest thrust to weight ratio is desired. Figure 5.2 in Brandt (2004 p. 176) shows that for a Mach number $M \approx 0.3$, an afterburning turbofan achieves the highest thrust to weight ratio. This is followed by an afterburning turbojet, turboprop and low bypass ratio turbofan.

The lowest TSFC is desired. Figure 5.3 in Brandt (2004 p. 177) shows that for a Mach number, $M \approx 0.3$, a piston engine with propeller gives the lowest TSFC followed by turboprop, high by pass ratio turbofan and low bypass ratio turbofan.

Recommendations-

Initial analysis suggests the use of a piston engine with a propeller. A secondary recommendation exists for a turboprop engine, followed by a low bypass ratio turbofan. Further investigations of existing piston engines were conducted. Approximately 350 piston engines are listed by Jackson (2008). Of these, only six provide a power output greater than 500hp. Initial design suggests that the required power output would lie between 1250 – 3000 hp. Only one engine, the CRM 18DD/SS provided a power output greater than 1,250hp. However, the CRM 18DD/SS weighed 3,745 lb, which was considered prohibitive to use on the aircraft. Consequently, piston engines were not selected as the engine type for the aircraft. An investigation of available turboprop aircraft was undertaken. Eighteen of the fifty engines listed by Jackson (2008) provide a power output with the desired 1500 – 3000 hp range. As such, enough variety existed within the turboprop range to allow for design optimization. Consequently, a turboprop engine was selected as the propulsion system type.

12.2 Number of Engines and the Power Required per Engine

Initial Design

Initial estimation suggested a total required power output between 1250 – 3000 hp. The large range in required power existed to encompass both the agricultural and regional jet prototypes. Early analysis of current aircraft showed that both single engine and twin-engine aircraft existed within this range. Engine number has a significant effect on configuration design. Consequently, it was important to identify the point in regards to both power output and engine weight at which the optimal design switches from single to twin engine. Data for the uninstalled power output and dry engine weight data for several engines was obtained from Jackson (2008). Installation effects were also considered. This required the reduction of output power and increase in engine weight.

Installed power output is defined below:

$THP = \eta p \times SHP$ (Roskam III 2002) Roskam III (2002) defines $\eta p = 0.88$ for a turboprop.

Maximum Power Requirement

$T/W_{cruise} = .11$ (calculated)

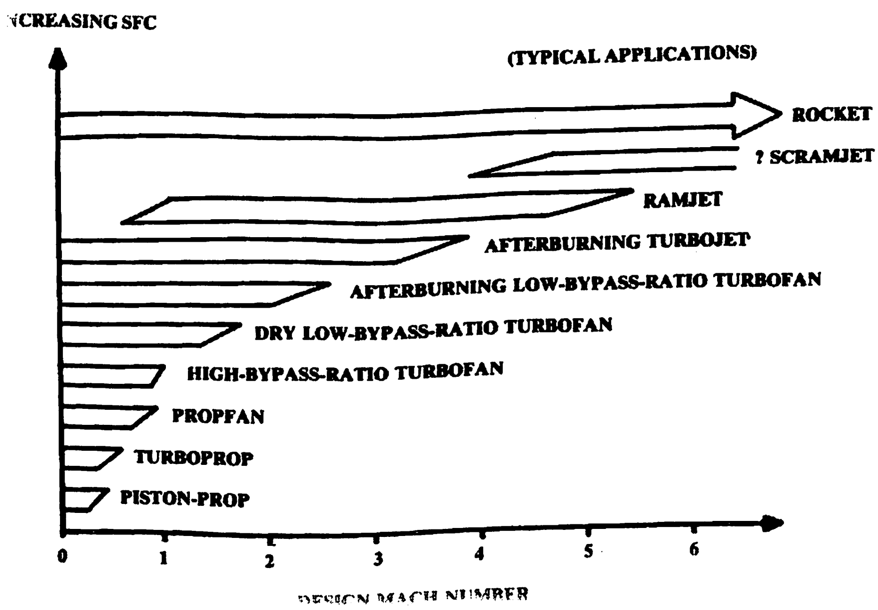
$T = .11 * W_{cruise}$

$T = .11 * 16013.305 = 1757lb$

$P = TV/ \square_{cruise}$

$P = 682792.71 ft\text{-}lb/s$

$= 1246.89 hp$



Plot 12.1 specific fuel consumption [Raymer 2006]

Engine Model Selection

Engine models that provided a power output similar to 1246.89hp were further investigated. Selection was limited to single engines, as configuration design preferred this arrangement. Table 12.1 below shoes data for suggested engine models

Table 12.1- Suggested Engine Models [Jackson 2008]

Manufacturer	Designation	Number of Required	Installed Power (hp)	Installed Weight (lb)	Installed Power to Weight	Specific Fuel Consumption (lb/(shp hr))
General Electric	CT7-5A	1	1526.8	1252.8	1.22	Not available.
General Electric	CT7-9	1	1707.2	1288	1.33	0.47
Pratt & Whitney	PT6A-67AG	1	1350	992	1.74	0.51
P&WC	PW121	1	1848	1473.6	1.25	0.48
Klimov	TV3-117VMA-SB2	1	2200	2011.2	1.09	Not available

The General Electric CT7-9, the Honeywell TPE331-14GR, P&WC 121, and Klimov TV3-117VMA-SB2 all satisfy the installed power requirements. The Pratt & Whitney has a significantly higher power-to-weight ratio than the other engines. From the available data, the General Electric CT7-9 has the lowest specific fuel constant. Although low specific fuel consumption was seen as a desirable characteristic, it was not considered as critical as power-to-weight ratio. Consequently, New Pratt & Whitney PT6A-67AG one will be used to power the aircraft

12.3 Propeller Sizing

The required propeller diameter can be determined from the following equation:

$$DP = ((4 \times Pmax) / (\pi \times np \times Pbl))1/2$$

where DP is the propeller diameter and the maximum power per engine (installed) is Pmax = 1350hp.The blade power loading, Pbl, and the required number of blades, np, is determined from statistical analysis of similar aircraft, and is summarized in Table 12.2 below.

Table 12.2- Statistical Analysis of Relevant Engines [Roskam III 2002]

Aircraft	Maximum Power per Engine,	Propeller Diameter (ft)	Number of Propeller	Blade power loading
Air Tractor AT-	600	9.1	2	4.6
PZL-M18A	1000	10.8	4	2.7
Beech 1900	1100	9.1	4	4.2
EMB-110	750	7.8	3	5.2
SF-340	1630	10.5	4	4.7

From the above statistical analysis,

Number of blades, np = 4 and blade power loading, Pbl = 4.5

Therefore,

$D_P = ((4 \times 1350) / (\pi \times 4.5 \times 4))^{1/2}$

$D_P = 9.7$ ft

Larger propellers are more efficient. However, the propeller tip speed must remain subsonic. The propeller tip speed can be calculated as the vector sum of the rotational tip speed and the aircraft forward speed.

$V_{rot} = \pi \times n \times D$

where n is the rotational speed of the engine, n = 1540 rpm = 25.66 rev/sec and D is the proposed propeller diameter, D = 9.7ft.

$V\ rot = \pi \times n \times D$

$V\ rot = \pi \times 25.66 \times 9.7$

V rot = 781.55 ft/s

The tip velocity can then be calculated using the following e quation:

$Vtip = \sqrt{(Vro^{2} + V^{2})}$
t

The aircraft cruise velocity is V = 341.7 ft/s and the engine rotational speed is as calculated above.

Therefore,

$V_{tip} = \sqrt{(781.55\ + 341.72)}\ V_{tip} = 852.99$ ft/s

This speed is below the speed of sound (a = 1061.4 ft/s) at the specified cruise altitude. Therefore, the propeller tip speed maintains subsonic.

12.3.1Propeller Material Selection

The maximum propeller tip speed dictates the material selection of the propeller. Metallic propellers should be used for applications with a maximum propeller tip speed of Vtip = 950 ft/s, whilst wooden propellers have a maximum propeller tip speed Vtip = 850 ft/s. The aircraft has Vtip = 853 ft/s, and consequently, a metallic propeller will be used.

12.3.2Propeller Type Selection

There are three main propeller types as outlined below:

Variable pitch: Blade pitch is varied to maintain an optimal lift-drag ratio with speed, which results in increased thrust across a range of speeds

Constant speed: Blade pitch angle is varied to maintain constant speed, which improves fuel efficiency

Controllable pitch: Pilot can override constant speed mechanism, which is useful to reverse the blade pitch angle to slow the aircraft down the additional drag produced by a controllable pitch propeller is not required for the relatively light aircraft designed in this project. Increased thrust is considered advantageous over increased fuel efficiency, as it will improve the Maneuverability of the aircraft. Consequently, a variable pitch propeller will be selected for the aircraft.

12.3.3Specific Propeller Selection

The Propeller (Hartzell) HC-B5MA-3D/M11691NS was selected for this application .This propeller has a diameter of 9.8 ft which meets the requirements.

12.4 Propulsion System Integration

The following section of the report focuses on the integration of the propulsion system into the overall aircraft design. Integration includes the selection of the installation configuration, location and the mounting of the engine. Finally, checks are performed to ensure complete compatibility with other aircraft systems.

Pusher/Tractor Selection

Three options exist for the configuration of propeller engines: tractor, pusher and mixed, as shown in Figure 22 below.

Figure 12.2 - Propeller Engine Configurations: Tractor and Pusher [Raymer 2006 p.25]

A mixed installation requires two engines, one located as a pusher and the other as a tractor. This is not appropriate for this design as it requires at least two engines, and hence, will not be discussed further. Tractor installations place the inlet in the free airstream, resulting in improved engine cooling. Furthermore, this layout improves the stability of the aircraft, allowing shortening of the fuselage and a reduction in tail size. Pusher installations reduce the flow disturbance over the wing, decreasing the skin friction drag and allowing the wetted area of the aircraft to be reduced. Other benefits of the pusher configuration include improved visibility for the pilot and reduced cabin noise. However, in a pusher configuration, the propeller receives disturbed airflow, substantially reducing its efficiency. Additionally, pusher configurations may require larger tail areas, longer landing gear and are more likely to suffer from FOD damage. These disadvantages are significant and resulted in the selection of a tractor configuration for the aircraft.

12.4.1Engine Mounting Selection

Figure 12.3 below shows the possible mounting locations for aircraft engines, including the fuselage, wings, tail or as part of an upper fuselage pod.

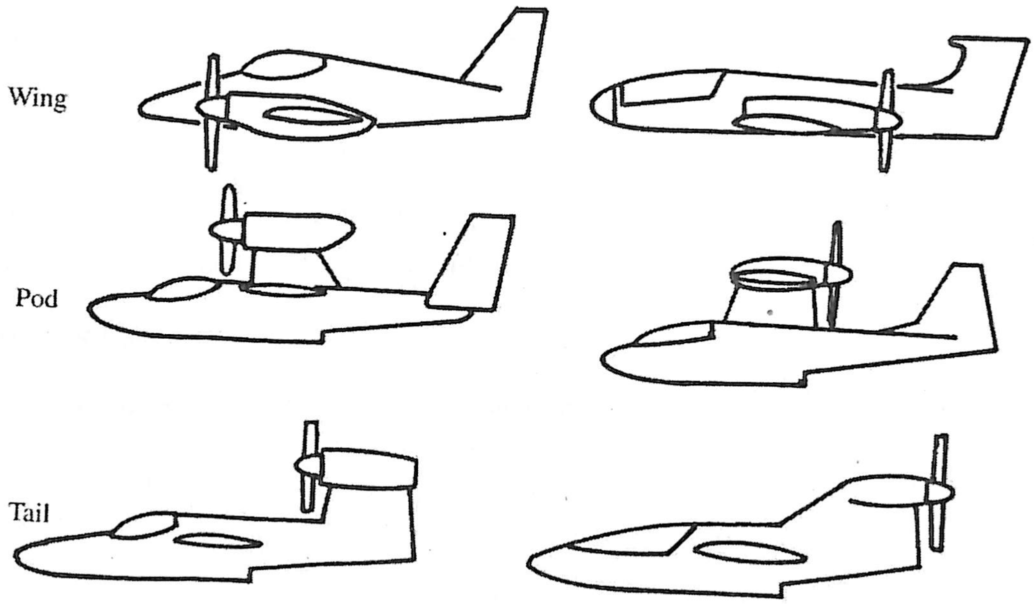


Figure 12.3 - Engine Mounting Locations: Fuselage, Wings, Tail or as Upper Fuselage Pod (Raymer 2006, p.252)

Wing mounting is not appropriate for this design as only one engine is used. Mounting engines on the tail or as part of upper fuselage pods results in a high thrust line that degrades the control characteristics of the aircraft. Consequently, this engine arrangement is used only for applications that require significant engine clearance, notably, amphibious aircraft. The aircraft does not require this level of clearance, and hence, the engines will not be mounted in a tail or upper fuselage pod.

12.4.2 Pratt & Whitney PT6A-67AG Specifications

The Pratt & Whitney PT6A-67AG is shown in Figure below.

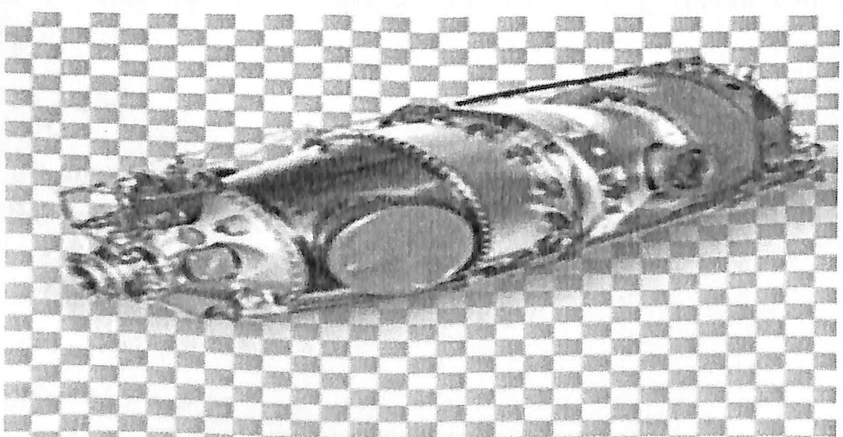


Figure 12.4 Pratt & Whitney PT6A-67AG [Wikipedia]

The Pratt & Whitney PT6A-67AG has the following dimensions:

Power	1350 SHP
Diameter	19 "
Length	76 "
RPM	1700

The engine has five mounting points. The locations of these are shown in Figure 23. The centre of gravity of the engine was not.stated by the manufacturer. Consequently, it was assumed that the centre of gravity was located at the geometric centre of the engine.

Cooling System Configuration

Cooling systems can be configured in an updraft or downdraft arrangement as shown in Figure 24 below.

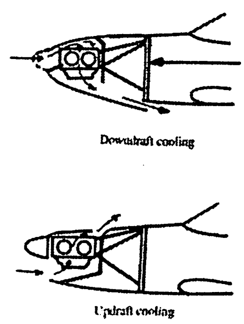


Figure 12.5 - Cooling System Configuration (Raymer 2006, p.256)

Updraft arrangements have maximum cooling efficiency but exhaust hot dirty air in front of the windscreen. This can cause the cabin to heat up, and in the event of an oil leak, can reduce pilot visibility. Downdraft arrangements do not suffer from these problems, but have reduced cool efficiency. As ground visibility is considered critical, a downdraft configuration will be utilized.

Air Intake Sizing

Raymer (2006) states that area required for the cooling intake can be determined using the following equation:

$A_{intake} = P / (2.2 \times V_{climb})$

where P is the installed power, (1354.54 hp). The climb speed Vclimb is assumed to be the average of the cruise and takeoff speeds. From the technical task, Vclimb =187.5 mph, (275 ft/s).

Therefore,

A intake = 1350 / (2.2 x 275)

A intake = 2.23ft^2

Air Exhaust Sizing

Roskam (2006) suggest that air exhausts should be sized as follows:

A exhaust / A intake = 0.8

Therefore, the recommend exhaust area is A exhaust = 1.79ft2

Firewall

Firewalls prevent the spread of heat or fire from the engine into the cockpit. Raymer (2006) states the requirement of a 0.015 inch thick sheet of stainless steel with no cut out to act as a firewall. This sheet should be attached to the first structural bulkhead of the fuselage. Any wires that pass through the firewall must have a fireproof sealing.

Fuel Tank Type

The three fuel tanks types are discrete, integral, and bladder. Discrete tanks are fabricated separately and then mounted to the aircraft. Discrete tanks are used predominantly for general aviation aircraft. Bladder tanks are a thick rubber bag stuffed into a cavity of the structure. Bladder tanks are self-sealing, but significantly reduce the available volume for fuel, and hence, are preferred for military applications, which benefit from the self-sealing capability. Integral tanks are part of the aircraft structure that has been sealed to form a tank. Consequently, an integral tank will be used for the aircraft.

Some other features of **Pratt & Whitney PT6A-67AG**

Multi- stage axial and single-stage centrifugal compressor

- Reverse flow, radial inlet with screen for FOD (Foreign Object Damage) protection
- Large high power PT6A models incorporate 4-stage axial and 1-stage centrifugal
- Small and Medium PT6A models incorporate 3-stage axial and 1-stage centrifugal

Reverse flow combustor

- Low emissions, high stability, easy starting, durable

Single-stage compressor turbine

- Cooled vanes in some models to maintain high durability

Independent 'free' power turbine with shrouded blades

- Large and Medium PT6A models incorporate 2-stage axial power turbine
- Small PT6A models incorporate 1-stage axial power turbine
- Forward facing output for fast hot section refurbishment

Epicyclic speed reduction gearbox

- Enables compact installation
- Output speed optimized for highest power and low propeller noise

CHAPTER 13

AERODYNAMIC AND PERFORMANCE ANALYSIS

The final conceptual fire fighting aircraft design involved a wing area of $342.57ft^2$ and an engine power of 1350 hp. An aerodynamic analysis was performed on the design to determine the lift to drag ratios for the main mission phases. These new aerodynamic properties and engine data were used to calculate a final estimated aircraft weight. This aircraft weight, in combination with the known wing area and engine power, was used to determine whether the design point remained within the met area of the matching diagram.

13.1 Aerodynamic Analysis

The lift to drag ratio of the aircraft in cruise and loiter phases can be calculated from the ratio of the respective lift and drag coefficients. These values can then be used to perform a new weight estimate.

13.1.2 Required Lift Coefficients in Cruise

The mission profile requires a cruise speed of 341.75 ft/s (375 km/h) At these speeds, the required wing lift coefficient was calculated

$C_{L} = 2W/ \square V^2S$

For cruise

$W_{cruise} = 16000.319bs$

$V_{cruise} = 341.8$ ft/s

putting all the values in eq. we get

$C_{Lcruise} = 0.517$

13.1.3Drag Coefficient in Cruiser Phase

During cruise and loiter, the aircraft is in the clean configuration. Hence, it has a zero-lift drag coefficient of .03

$C_D = Cd_0 + K\ C_L^2$

$C_L^2 = .51^2 = .26$

$K = 1/(3.14\ A\ e)$

$= .05$

$C_D = .03 + .05*(.26)$

$= .03 + .013$

$= .043$

13.1.4Lift to Drag Ratio Calculation

The lift to drag ratio for each phase was calculated by dividing the phase lift coefficient by the phase drag coefficient. The lift to drag ratios were calculated to be 14.17 for cruise . These lift to drag ratios are compared to the assumed lift to drag ratios in Table 13.1. This comparison shows that the aerodynamic performance of the aircraft in cruise has improved significantly upon the assumed performance. The aerodynamic performance of the aircraft in loiter has decreased slightly from the assumed performance.

For cruise

$C_D = 0.043$

$C_L = 0.517$

$L/D = C_L / C_D$

$L/D = 12.02$

Table 13.1- Comparison of Assumed and Estimated Lift to Drag Ratios

Phase	Assumed L/D	Estimated L/D
Cruise	12.7	12.02

CHAPTER 14

WEIGHT AND BALANCE ANALYSIS

The aircraft takeoff weight of 16726.26 lbs can be distributed to different groups and components within the aircraft using statistics, except when the weight of actual components or systems is available in which case actual weights are used. Weight distribution percentages, shown in Table 21, suggested by Arjomandi (2009), were used as a guide due to the absence of more specific data in Roskam (1985). System weight was distributed evenly between cockpit systems and payload systems. Landing gear weight was distributed with 25% at the nose gear and 75% at the main landing gear.

Table 14.1 - Suggested Weight Distribution as Percentages (Eger 1983; Arjomandi 2009)

Component	Percentage	Reference weight
System	12-15%	Takeoff weight
Fuselage	30-40%	Structural weight
Wings	30-40%	Structural weight
Empennage	5-10%	Structural weight
Landing gear	10-15%	Structural weight

Actual weight distribution is shown in table 14.2

Takeoff weight = 16726.26

Structural weight = 9454.54lb

Table 14.2 Actual weight distribution

Component	Percentage %	Actual weight (lbs)
System	13	2174.4
Fuselage	35	3309.08
wing	35	3309.08
Empennage	8	756
Landing gear	12	1134

CHAPTER 15

RESULT AND CONCLUSION

TABLE 5.1 Complete specifications and performance of aircraft

Engine type	P&W PT6A-67AG
Engine SHP & RPM	1,350 @ 1,700 RPM
Propeller	Hartzell HC-B5MA-3D/M11691NS
Propeller diameter	9.8 ft
Takeoff weight	16726.26 lbs (7577kg)
Landing weight	16726.26 lbs (7577kg)
Empty weight	9454 lbs (4288.580)
Pay load weight	4444.4 lbs (2000kg)
Crew weight	222.22 lbs (100 kg)
Total length	37.5 ft
Height	7 ft
Wing span	50.688 ft
Wing area	342.57 ft
Chord length	6.75 ft
Aspect ratio	7.5
Main wheel size	24 * 5.5 in.
Tail wheel size	18 * 4.4 in.
Cruise speed (8000 ft)	375 km /hr (341 ft/sec)
Stall speed	170 km/ hr (154 ft / sec)
Range (8000ft)	150 km
Rate of climb	14.67 ft /s

15.1 Final solid works design

15.1.1 Isometric view

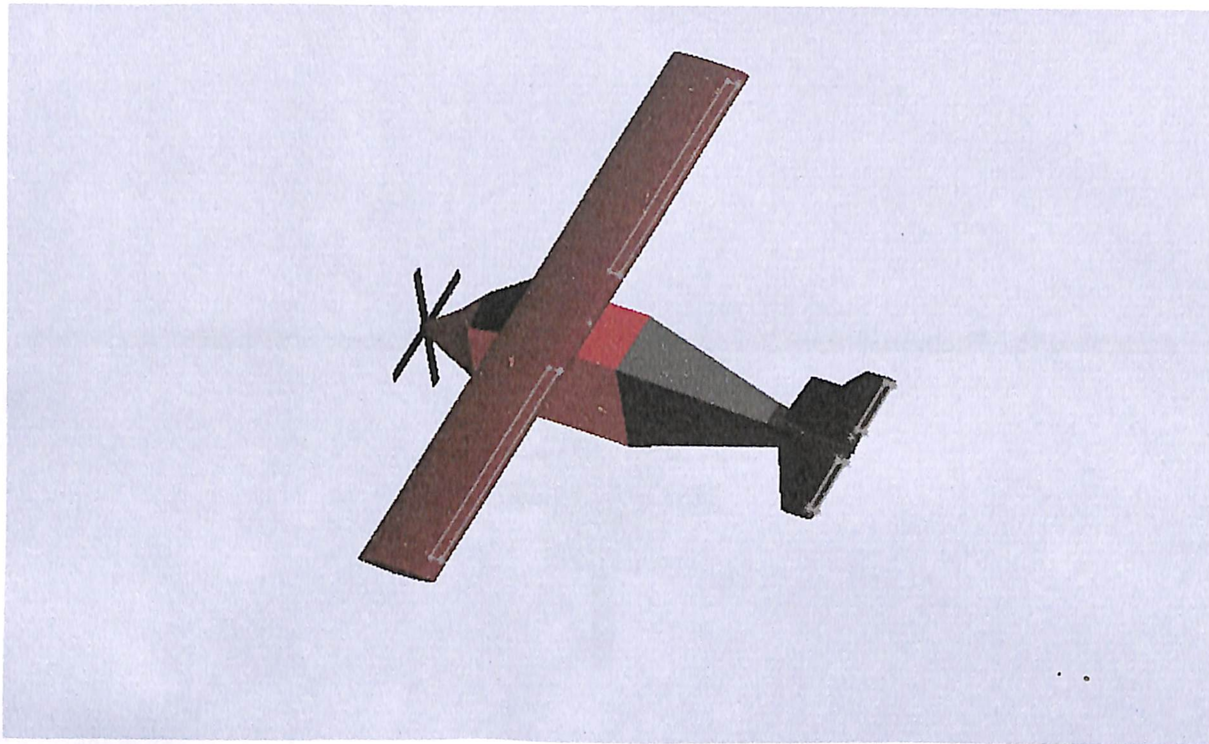


Figure 15.1 isometric view 1

Figure 15.2 isometric view 2

15.1.2 Front view

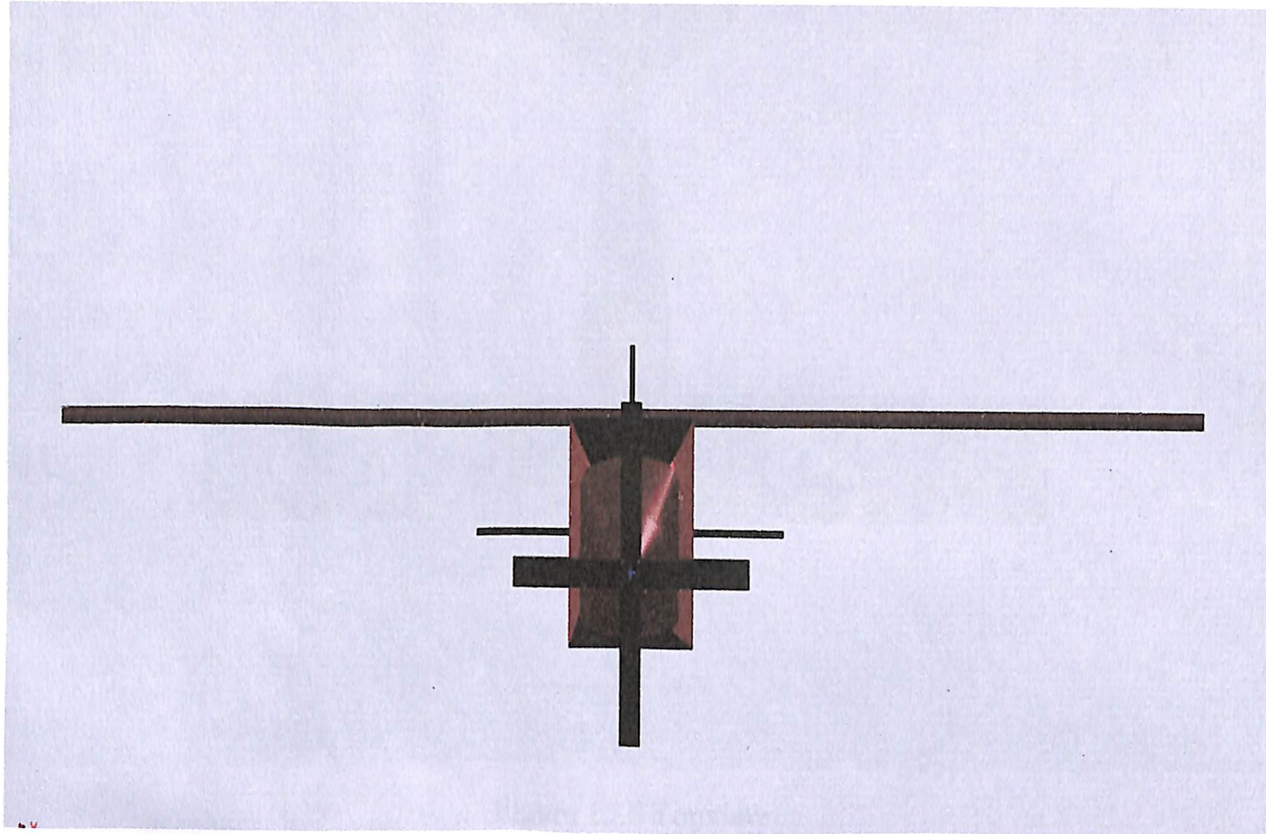


Figure 15.3 front view

15.1.3 Side view

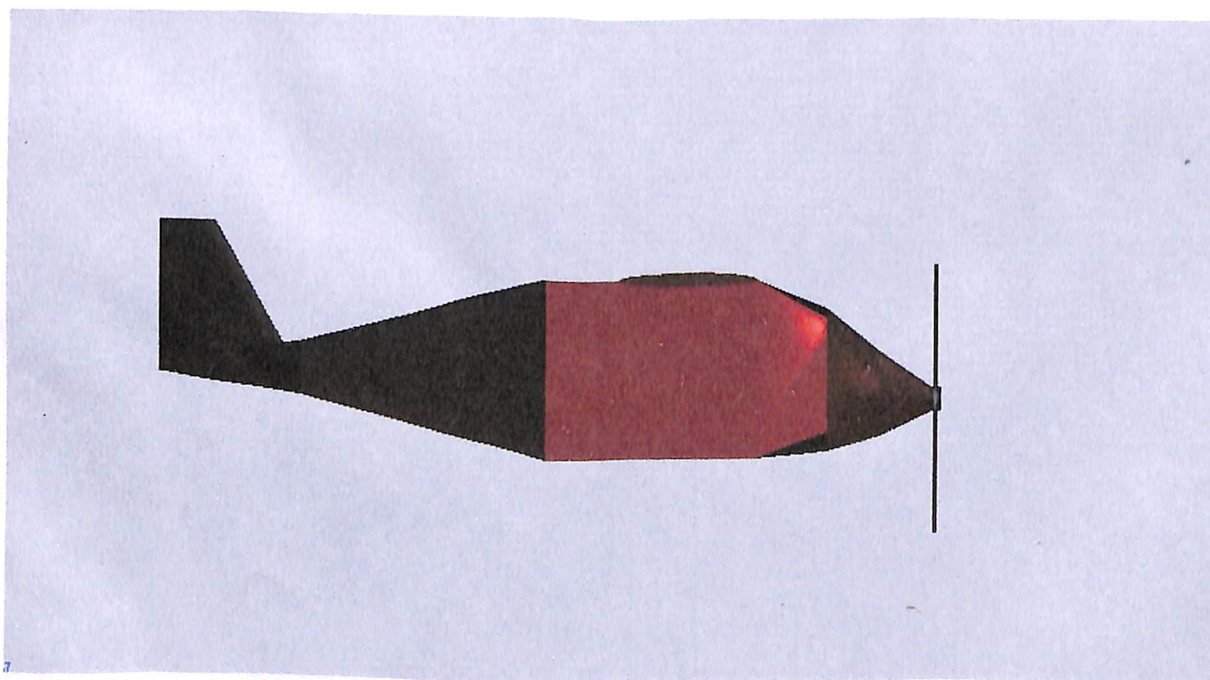


Figure 15.4 side view

14.1.4 Top view

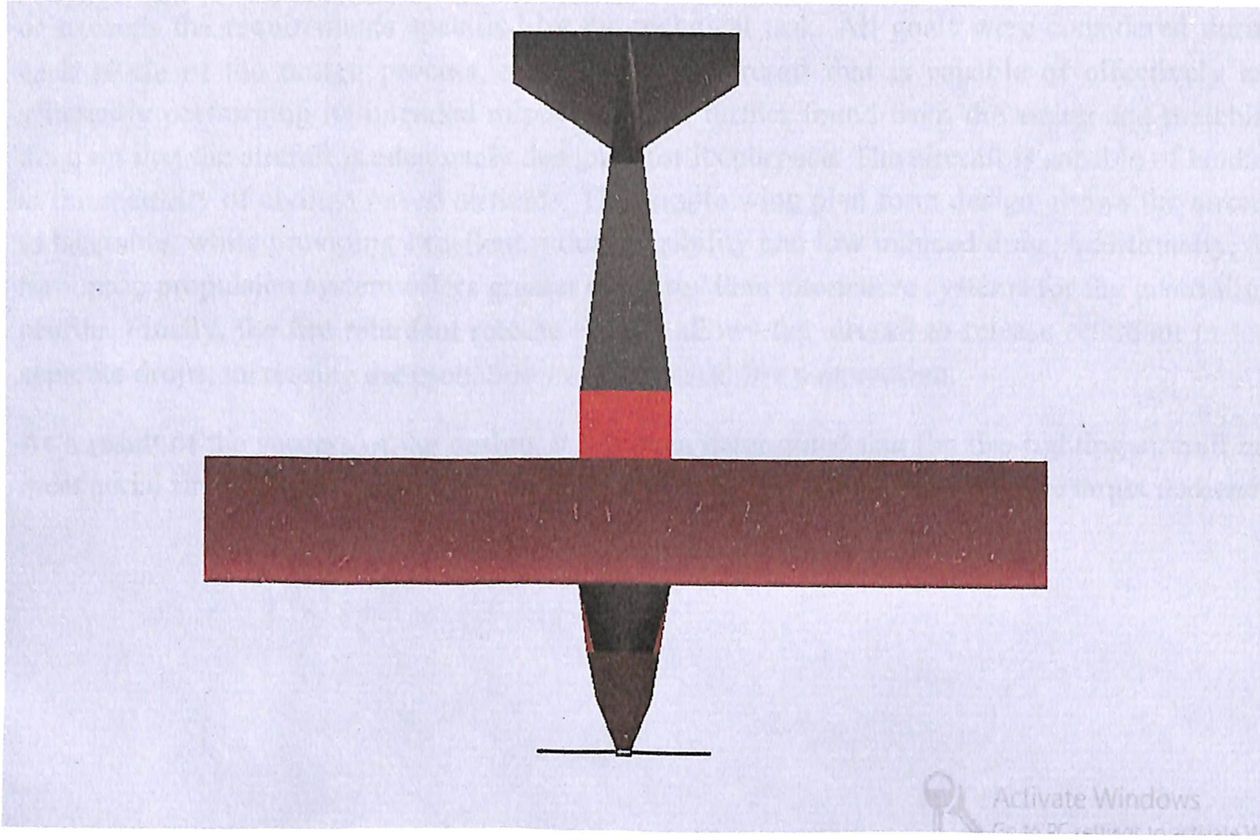


Figure 15.5 Topview

Conclusion

The conceptual design process for the fire-fighting aircraft has resulted in an aircraft that meets or exceeds the requirements specified by the technical task. All goals were considered during each phase of the design process, resulting in an aircraft that is capable of effectively and efficiently performing its intended mission. It was further found from the sizing and matching diagram that the aircraft is adequately designed for its purpose. The aircraft is capable of landing at the majority of civilian paved airfields. The simple wing plan form design allows the aircraft to be stable, while providing excellent ground visibility and low induced drag. Additionally, the turboprop propulsion system offers greater economy than alternative systems for the given flight profile. Finally, the fire retardant release system allows the aircraft to release retardant in four separate drops, increasing the probability of successful fire suppression.

As a result of the success of the design, it has been determined that the fire-fighting aircraft can meet aerial fire-fighting requirements, and would be an attractive aircraft for the target audience.

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