UNIVERSITY OF BELGRADE

FACULTY OF MECHANICAL ENGINEERING

Abdulhakim Essari

ESTIMATION OF COMPONENT DESIGN WEIGHTS IN CONCEPTUAL DESIGN PHASE FOR TACTICAL UAVs

Doctoral Dissertation

Belgrade, 2015

УНИВЕРЗИТЕТ У БЕОГРАДУ
МАШИНСКИ ФАКУЛТЕТ

Abdulhakim Essari

ПРОЦЕНА ТЕЖИНА КОМПОНЕНТИ ТАКТИЧКИХ БЕСПИЛОТНИХ ЛЕТЕЛИЦА У ФАЗИ КОНЦЕПТА

Докторска дисертација

Београд, 2015.

EXAMINATION COMMITTEE:

Supervisor:	Dr. Zlatko Petrovic Full Professor University of Belgrade Faculty of Mechanical Engineering
Members	Dr. Ivan Kostic Full Professor University of Belgrade Faculty of Mechanical Engineering
	Dr. Mirko Dinulovic Full Professor University of Belgrade Faculty of Mechanical Engineering
	Dr. Aleksandar Grbovic Assistant Research Professor University of Belgrade Faculty of Mechanical Engineering
	Dr. Zoran Stefanovic Full Professor, retired University of Belgrade Faculty of Mechanical Engineering

Комисија за оцену и одбрану дисертације:

Ментор: др Златко Петровић, редовни професор
Универзитет у Београду - Машински факултет

Чланови комисије: др Иван Костић, редовни професор
Универзитет у Београду - Машински факултет

Др. Мирко Динуловић, редовни професор
Универзитет у Београду
Машински факултет

др Александар Грбовић, доцент
Универзитет у Београду
Машински факултет

др Зоран Стефановић, редовни професор у пензији
Машинског Факултета у Београду

DEDICATION

To Spirit of my Son, Ali

My parents

My wife

Aseel, Farouk, Kareem

Acknowledgment

I would like to express my deep appreciation and gratitude to my supervisor Professor Dr. Zlatko Petrović for his guidance and advises that made this work possible. It has been great pleasure to work under his supervision.

My sincere appreciation thanks and gratitude to the academic staff members of my faculty.

I would like to express my special thanks to all of my family members whose are always next to me support me and give me hope and courage.

My sincere appreciation, thanks and gratitude to the Libyan General People's Committee of higher education which gave me scholarship and have supported me to my further education.

Abstract

New formulas for component weight estimation and for takeoff weight estimation, in conceptual design phase are derived for a tactical unmanned aerial vehicle (TUAV). Formulas are derived by analyzing existing UAVs of the weighs from 100 to 500 kg, and which have similar characteristics. The materials suggested for UAV weight design are aluminum alloys and aluminum based composite materials.

Based on statistical trends, obtained from analyzed existing UAVs, takeoff weight is estimated from mission specification, and given payload weight. Software tools are developed in Matlab to facilitate takeoff and component weight calculations. The least square method is applied to analyze statistical data in order to develop trend functions which correlate TUAVs empty weight and takeoff weight. Existing formulas, developed for general aviation, for component and takeoff weight estimations are applied to TUAV and promising one are selected and adjusted to TUAV conceptual design phase.

Component weights are related to geometrical parameters, maximum speed, and takeoff weight of the TUAVs.

All existing and newly developed formulas are applied to typical TUAV example to validate it aplicability.

Абстракт

У тези су развијене нове формуле за процену полетне тежине и тежине компоненти тактичких беспилотних летелица (ТБЛ) у фази конципирања летелице. Формуле су изведене на основу анализе сличних изведених беспилотних летелице сличне намене. Тежине тактичких беспилотних летелица су у распону од 100 кг до 500 кг. Предложени материјали за израду ових летелица су алуминијумске легуре и композитни материјали, као и метални композитни материјали.

На основу статистичких трендова добијених анализом постојећих ТБЛ полетна маса се одређује на основу спецификације мисије летелице и тежине товара који летелица носи. За једноставнију процену тежина летелице се користе софтверски алати засновани на језику МАТЛАБ. Трендови су дефинисани применом методе најмањих квадрата они су засновани на анализираним статистичким подацима који добијени анализом постојећих ТБЛ.

У тези је урађен типичан пример, на којем су примењене постојеће формуле развијене за лаке авионе и опште ваздухопловство. Селекциониране су формуле које уз малу дораду могу бити примењене за прорачун тежина компоненти и полетне тежине ТБЛ у фази конципирања летелице.

Тежине компоненти су повезане са геометријским параметрима компоненте, брзином лета ТБЛ и полетном тежином ТБЛ.

Коначно су све постојеће и ново добијене формуле примењене на типичну ТБЛ у циљу валидације и применљивости развијених формула.

TABLE OF CONTENTS

LIST OF FIGURES

LIST OF TABLES

1. Introduction:

The dependence on Unmanned Aerial Vehicles (UAV’s) in last decade grow significantly especially for combat missions, and the demand for UAV’s is greatly increased. UAV’s play an important role in fields like, information superiority, collateral damage, urban area fighting and precision strikes against high payoff targets. UAV’s evolved to include size growth of strategic UAV’s for carrying more payload weight, and longtime endurance, and minimize tactical UAV’s size.

UAV’s are defined as aircraft which flies without pilot inside it. It can be released into the air in several ways such as ordinary runway, by hand, by launcher, by rocket, from bigger airplane, or any other convenient mean. UAV sizes start from insect size to thousands of kilograms according to its mission, altitude, endurance, range, and payload carried. Tracking and controlling of the UAV's are done automatically or by wireless connections. It also can fly fully autonomously by computer programmed mission, starting from the takeoff moment until it returns controlled all time by computers inside it. (2)

2. Distinction between manned and unmanned aircraft:

The distinction between manned and unmanned aircraft for performance the same mission is that the manned aircraft needs, additionally to two pilots’ crew, two seats with mechanical adjustment features, safety harnesses and parachutes. The cabin should be wider to accommodate the crew. Bigger fuselage size results in higher surface area, which generates more drag. (2)

Supplements such as windows for visibility and doors are required, also cabin environmental controls and instrumentation and avionics are part from manned aircraft, etc.

Everything that we talked about previously means hundreds of kilograms of extra weight, which doubles the cost of the aircraft production several times, and increases the operating cost such as fuel consumption which is much lower as the weight of the aircraft is less. On the other hand, it is clear that unmanned aircraft save the human life. And significantly decreases the cost of aircraft manufacture. (2)

Because UAVs have high-aspect-ratio wings and fly in low-density conditions, often at low speeds; airflow is characterized by low Reynolds numbers. Aerodynamic properties of such flying object are significantly different from properties of general aviation aircrafts or ultra-light aircrafts. New category of flaying vehicles need extensive research in low Reynolds number flight regime, searching for shapes of sufficiently high lift coefficients and reasonably low sensitivity to flow separation.

In conceptual design phase it is not required to specify exactly shape of the airfoil, but, it is required to estimate its aerodynamic characteristics since all geometry parameters depend on this estimation. In this thesis characteristics of the propeller airfoils will be used as a basis for aerodynamic calculations. Having the small cord length and airfoil close to the spinning axis have similar Reynolds numbers as small UAV's. (2)

The most important parameter which dictates all other design parameter is estimation of the UAV's weight. Since there are no enough reliable sources for such estimation, the main goal of this thesis is to establish empirical relationships which will lead to reliable takeoff weight estimation of the UAV's, with emphasis on tactical UAV's.

Since UAV's widely vary in size, and flaying range it is necessary to introduce some kind of classification among them. It will be adopted classification presented in (2), and this thesis is concentrated around, so called, tactical UAV's. Classification in more detail will be presented in this thesis.

Design of any flaying vehicle is performed in three steps:

- Conceptual design phase
- Preliminary design phase
- Detail Design phase

Only conceptual design phase is considered to be effort of relatively small group of engineers and specialists. This phase is also the cheapest it should provide the answer if the vehicle is possible to design and what characteristics will it have. It is also only paper phase requiring no special equipment and research. Outer geometry of the vehicle is also defined in this phase.

In preliminary design phase all technological aspects have to be clarified, necessary wind tunnel tests have to be performed, and production methods have to be selected and tested. This phase could require more engineering hour engagement even different research institution involvement. Cost of this phase is considerably higher than the cost of conceptual design phase.

Detail design phase starts when all components of the vehicle are conceptually designed. In this phase real objects are created, documentation produced both for parts, assemblies, tools and test equipment. This phase requires the longest period to finish, engage lot of engineers and specialist and has the highest cost. (2)

3. Motivation of the work:

Since conceptual design phase cost least, it is wise to perform it thoroughly and to postpone crucial decisions as late as possible since all subsequent phases are continuation of this phase. This will increase the cost of this phase but since its share in total cost is usually less than 5% increase is not significant, but can reduce significantly the cost of subsequent phases. It cannot be stressed enough that outcome of the conceptual design phase greatly depend on initial weight estimation of the UAV. This thesis will contribute to this problem by deriving equations for initial and component weight estimation.

2. UAV's Specifications for Research data:

2.1 Introduction:

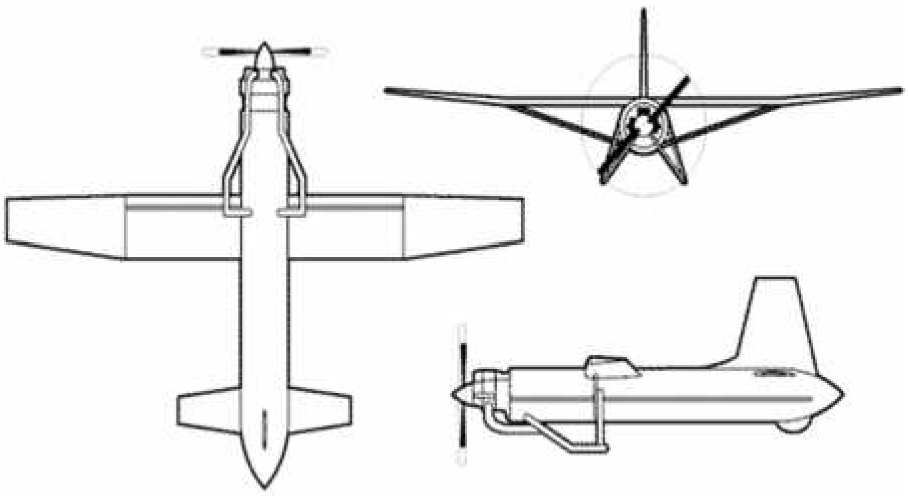
Studding weight estimation of tactical UAVs requires collecting data for the existing UAVs which have the similar characteristics, mission, and takeoff weight. The aircrafts presented in this work are tactical UAVs having takeoff gross weight ranges between 100 – 500 Kg.

All of the UAV's data provided in this chapter is from (1) & (20).

2.2 D-4 NPU (XIAN):

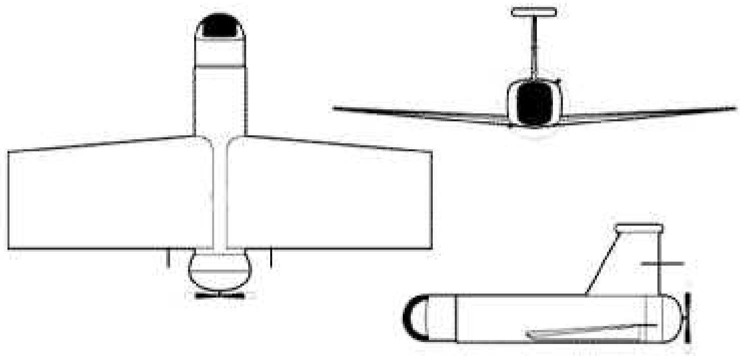
Country of origin, China

Wing span	4.3 m
Fuselage length	3.3 m
Payload weight	28 kg
Take off weight	140 kg



2.3 Brevel:

Country of origin, France & Germany

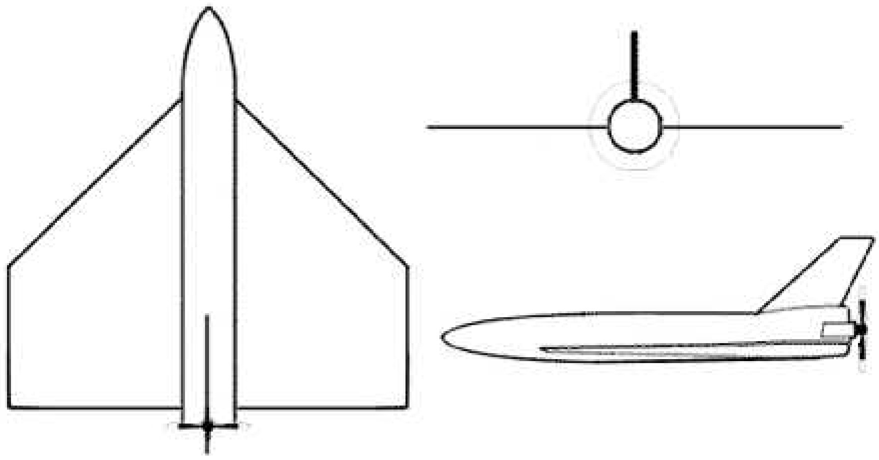


Wing span	11 ft, 1 in (3.4 m)
Unmanned aerial vehicle	low wing monoplane
Length	2.3 metres
Height	0.9 metres
Operational altitude	2,000 metres
Operating speed	150 km/hour
Launch	rocket launch from flatbed vehicle parachute and airbag recovery
Maximum take-off weight	150 kg

2.4 Crecerelle:

Country of Origin, UK & France

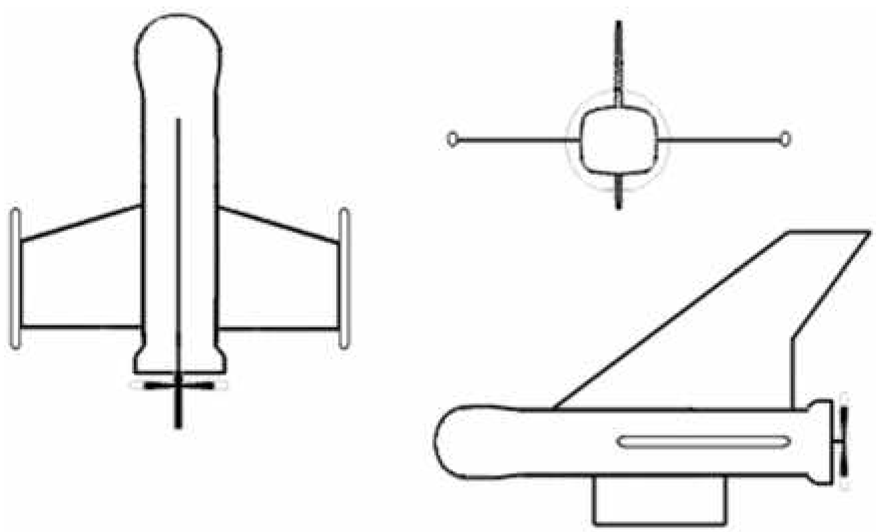
Length	2.40 m (7 ft 11 in)
Wingspan	3.30 m (10 ft 10 in)
Height	0.70 m (2 ft 4 in)
Gross weight	120 kg (250 lb)
Length	2.40 m (7 ft 11 in)
Maximum speed	240 km/h (150 mph)
Endurance	5 hours
Service ceiling	3,100 m (10,000 ft)



2.5 TAIFUN:

Country of Origin, Germany

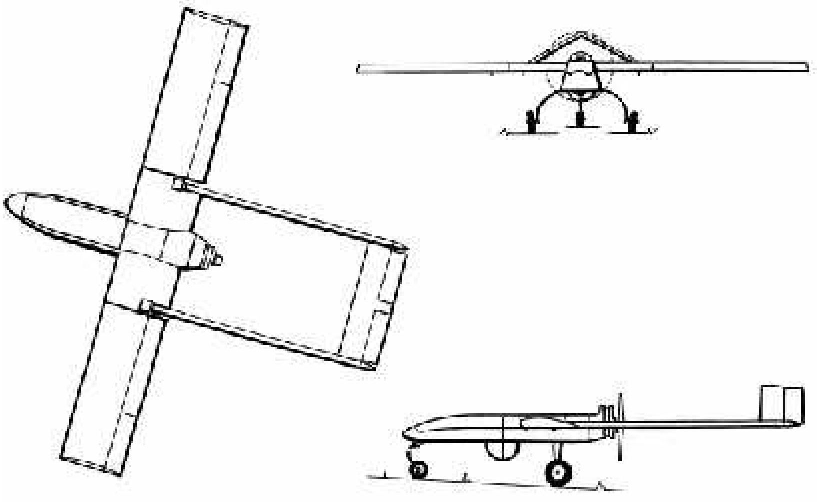
Length	2.08 m (6.82 ft)
Wingspan	2.26 m (7.41 ft)
Ceiling	4,000 m 13,123 ft
Max Range	500 kilometer
Length	2.08 m (6.82 ft)
Speed at High Altitude	202 kph



2.6 Shadow 200 AAI RQ7:

Country of origin: USA

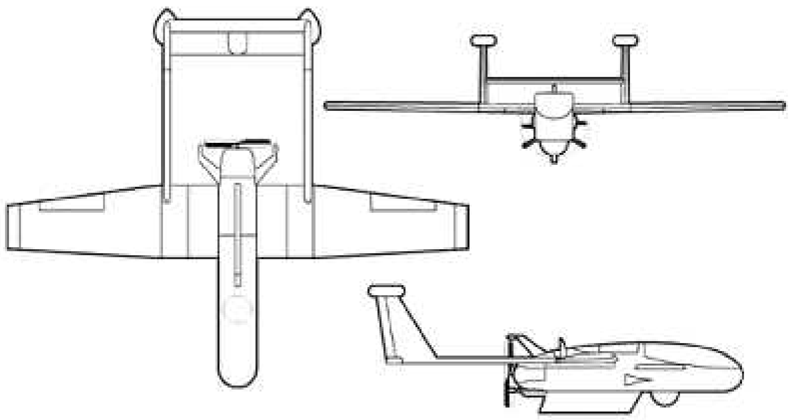
Length	11.2 ft (3.4 m)
Wingspan	14 ft (4.3 m)
Height	3.3 ft (1.0 m)
Empty weight	186 lb (84 kg)
Gross weight	375 lb (170 kg)
Maximum speed	127 mph; 204 km/h (110 kn)
Cruising speed	81 mph; 130 km/h (70 kn)
Range	68 mi (59 nmi; 109 km)
Endurance	6 h/ 9 h Increased Endurance
Service ceiling	15,000 ft (4,572 m)



2.7 Mirach 26:

Country of origin, Italy

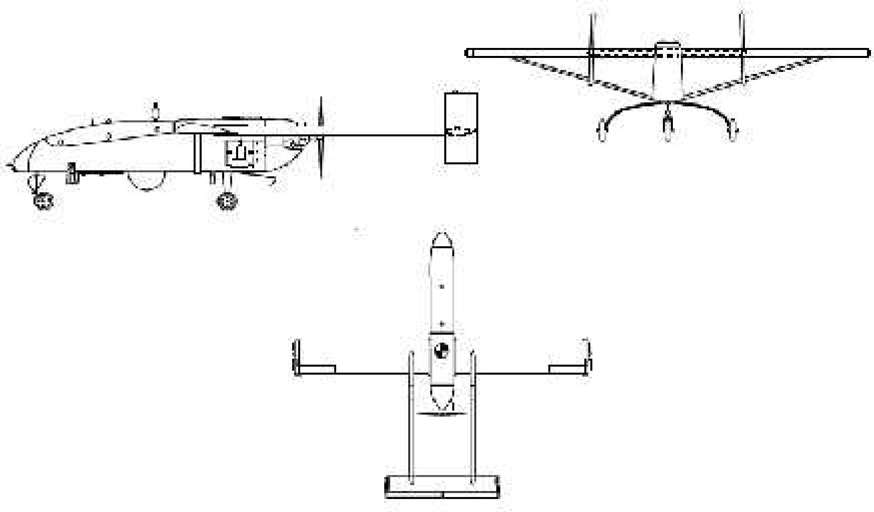
Length	3.85 m (12 ft 8 in)
Wingspan	4.73 m (15 ft 6 in)
Height	1.27 m (4 ft 2 in)
Gross weight	200 kg (440 lb)
Empty weight	110 kg
Capacity	50 kg (110 lb) payload
Maximum speed	220 km/h (138 mph)
Endurance	6 hours
Service ceiling	3,500 m (11,500 ft)



2.8 Pioneer AAI RQ2:

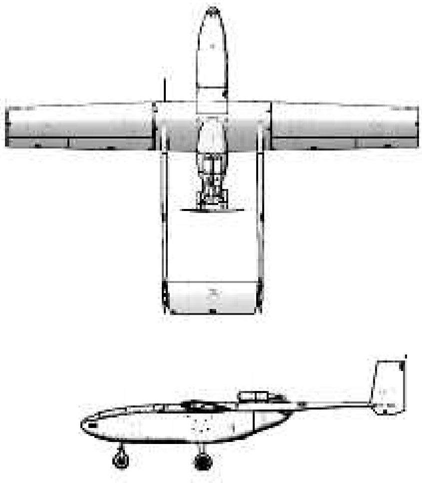
Country of origin, USA Israel

Length	14 feet (4 m)
Height	3.3 feet (1.0 m)
Weight	205 kilograms (452 pounds)
Empty Weight	131 kg (392 lb)
Wingspan	16.9 feet (5.2 m)
Speed	110 knots (200 km/h)
Range	five hours at 185 kilometers
Ceiling	4600 meters (15,000 ft)
Fuel Capacity	44-47 liters



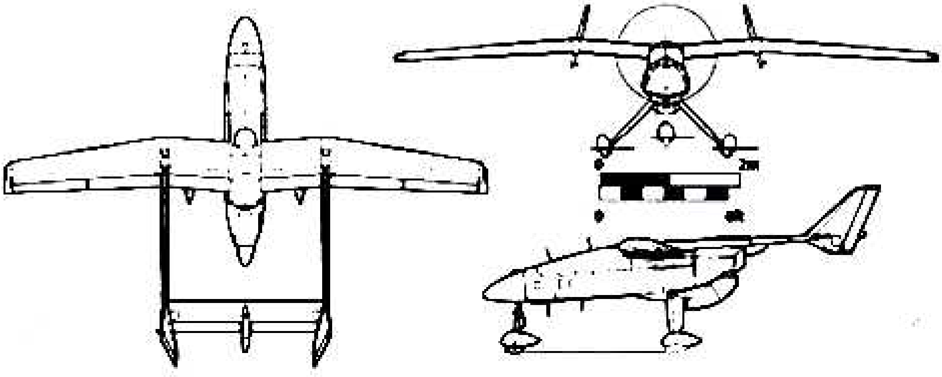
2.9 Viking 400:

Wing Span	6 m (20.0 ft)
payload carrying	From 33-45 kg (75-100 lbs)
Empty weight	145 kg (320 lb)
Max Gross Takeoff Weight	239 kg (540 lbs)
Length	4.4 m (14.7 ft)
Endurance	From 8.2 -11.4 hours
Cruise Speeds	60 kts / 90 kts



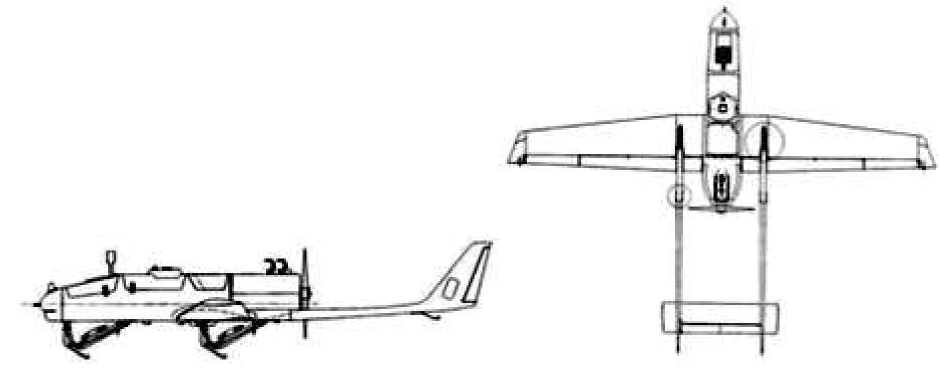
2.10 Falco:

Air vehicle length	5,25 m
Wing span	7,20 m
Height	1,80 m
MTOW	420 Kg
Endurance	8-14 hours
Max payload weight	70Kg
Ceiling	6500 m
Max airspeed	60 m/s
Maximum speed	216 km/h (134 mph)



2.11 RUAG ADS 95 Ranger:

Wing span	5.708 m (18 ft 8.7 in)
Wing area	3.41 m2 (36.70 sq ft)
Length overall	4.611 m (15 ft 1.5 in)
Fuselage: Max width	0.42 m (1 ft 4.5 in)
Max depth	0.47 m (1 ft 6.5 in)
Height overall	1.125 m (3 ft 8.3 in)
Tail plane span	1.553 m (5 ft 1.1 in)
Max payload	45 kg (99.2 lb)
Max launching weight	275 kg (606 lb)
Max level speed	130 kt (240 km/h; 149 mph)
Cruising speed	97 kt (180 km/h; 112 mph)
Loiter speed	70 kt (130 km/h; 81 mph)
Stalling speed	49 kt (90 km/h; 56 mph)
Ceiling	5,480 m (18,000 ft)
Command link range	81 n miles (150 km; 93 miles)
Endurance:	standard fuel 5 h

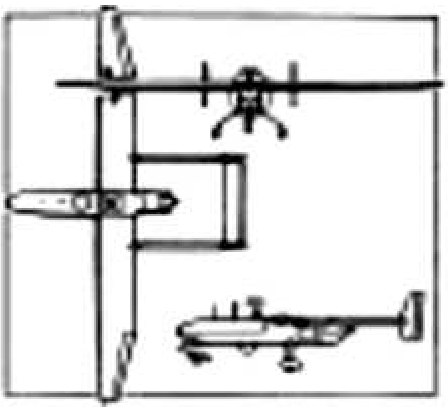


2.12 INTA/Ceselsa/EADS Dornier SIVA:

Wing span	5.81 m (19 ft 0.7 in)
Length overall	4.025 m (13 ft 2.5 in)
Width, wings folded	2.20 m (7 ft 2.6 in)
Fuselage: Max width	0.42 m (1 ft 4.5 in)
Max depth	0.55 m (1 ft 9.7 in)
Max launching weight	300 kg (661 lb)
Height overall	1.15 m (3 ft 9.3 in)
Propeller diameter	0.71 m (2 ft 4.0 in)
Payload bay volume	15.0 dm3 (0.53 cu ft)
Fuel weight: standard	40 kg (88.2 lb)
max	60 kg (132.3 lb)
Max payload	40 kg (88.2 lb)
Max level speed	92 kt (170 km/h; 105 mph)
Normal cruising speed	76 kt (140 km/h; 87 mph)
Max rate of climb at S/L	300 m (984 ft)/min
Ceiling	6,000 m (19,680 ft)
Mission radius	81 n miles (150 km; 93 miles)
Endurance: with max payload	6 h
with max fuel	10 h

2.13 Kentron Seeker:

Wing span	7.00 m (22 ft 11.6 in)
Wing area	4.427 m^2 (47.65 sq ft)
Length overall	4.438 m (14 ft 6.7 in)
Fuselage length	3.09 m (10 ft 1.6 in)
Height overall	1.30 m (4 ft 3.2 in)
Tail unit span	1.60 m (5 ft 3.0 in)
Wheel track	1.20 m (3 ft 11.2 in)
Payload bay volume	120.0 dm^3 (4.23 cu ft)
Weight empty	151 kg (323 lb)
Fuel weight: standard	49 kg (108 lb)
max	61 kg (134 lb)
Payload: standard	40 kg (88.2 lb)
max	50 kg (110 lb)
Max T-O/launching weight	240 kg (529 lb)
Max level speed	95 kt (176 km/h; 109 mph)
Cruising speed	65 kt (120 km/h; 75 mph)
Ceiling	6,100 m (20,000 ft)
Endurance: standard payload	15 h
max payload	12 h

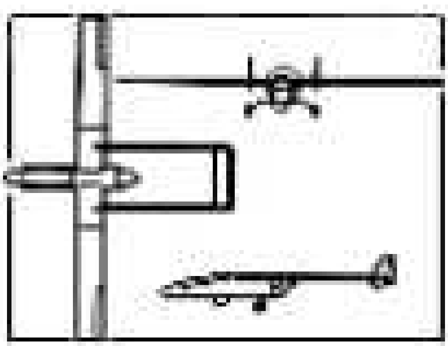


2.14 Meggitte Sentry:

Wing span: A	3.35 m (11 ft 0.0 in)
B	3.90 m (12 ft 9.5 in)
Wing area: A	2.84 m2 (30.62 sq ft)
B	2.91 m² (31.37 sq ft)
Length overall: A	2.44 m (8 ft 0.0 in)
B	2.57 m (8 ft 5.0 in)
Fuselage: Length: A	1.65 m (5 ft 5.0 in)
Max width: A	0.76 m (2 ft 5.9 in)
Weight empty: A	59.0 kg (130 lb)
B	81.6 kg (180 lb)
Fuel weight: A	27.2 kg (60 lb)
B	32.7 kg (72 lb)
Max payload with full fuel: A	27.2 kg (60 lb)
B	34.0 kg (75 lb)
Max launching weight: A	113.4 kg (250 lb)
B	147.4 kg (325 lb)
Max level speed: A	95 kt (176 km/h; 109 mph)
B	110 kt (203 km/h; 126 mph)
Cruising speed: A	80-85 kt (148-157 km)
B	80 kt (148 km/h; 92 mph)
Endurance A,B	8 hr
Ceiling: A	4,575 m (15,000 ft)
B	4,875 m (16,000 ft)

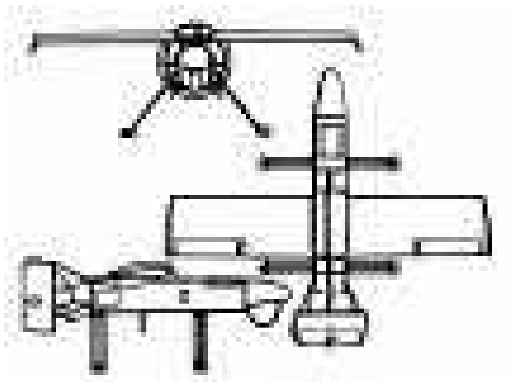
2.15 AAI Shadow 600:

Wing span	6.83 m (22 ft 4.8 in)
Wing area	3.754 m^2 (40.41 sq ft)
Length overall	4.77 m (15 ft 7.8 in)
Height overall	1.24 m (4 ft 0.8 in)
Weight empty	148.4 kg (327 lb)
Max payload	41.0 kg (90.4 lb)
Max T-O weight	265 kg (584 lb)
Max level speed	104 kt (193 km/h; 120 mph)
Cruising speed	75 kt (139 km/h; 86 mph)
Endurance	12-14 hours
Maximum altitude	16,000 ft or 4,877 m
Range	200 km



2.16 Yakovlev Pchela/Shmel:

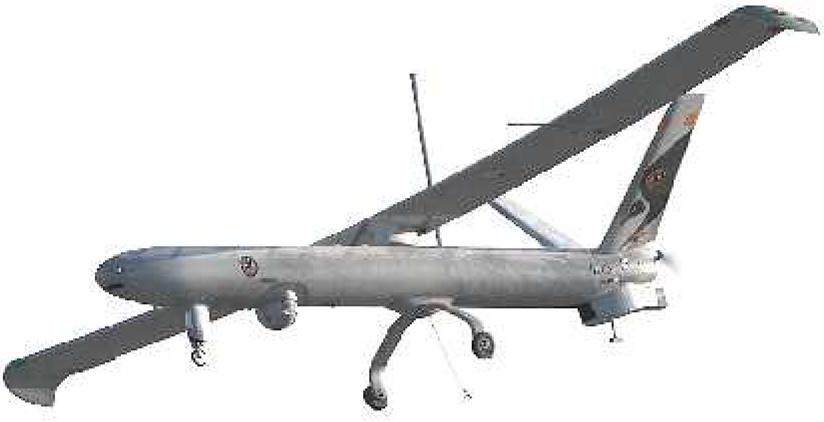
Wing span	3.25 m (10 ft 7.9 in)
Wing area	1.83 m³ (19.7 sq ft)
Length overall	2.78 m (9 ft 1.4 in)
Height overall	1.11 m (3 ft 7.7 in)
Max launching weight	138 kg (304 lb)
Max level speed	97 kt (180 km/h; 112 mph)
Cruising speed	65 kt (120 km/h; 75 mph)
Operating height range	min 100 m (330 ft), Max 2,500 m (8,200 ft)
Endurance	2 h



2.17 Silver Arrow Hermes 450:

Wing span	10.51 m (34 ft 6.0 in)
Wing chord, constant	0.69 m (2 ft 3.2 in)
Wing area	6.90 m2 (74.27 sq ft)
Length overall	6.10 m (20 ft 0.0 in)
Body diameter (max)	0.52 m (1 ft 8.4 in)
Height overall	2.37 m (7 ft 9.3 in)
Tail plane span	2.95 m (9 ft 8.1 in)
Wheel track	1.45 m (4 ft 9.1 in)
Payload bay volume	300.0 dm3 (8.5 cu ft)
Weight empty	200 kg (441 lb)

Max usable fuel	105 kg (231.5 lb)
Max payload*	150 kg (331) lb
Max T-O weight	450 kg (992 lb)
Max level speed	95 kt (176 km/h; 109 mph)
Cruising speed	70 kt (130 km/h; 80 mph)
Stalling speed	42 kt (78 km/h; 49 mph)
Max rate of climb at S/L	274 m (900 ft)/min
Max operating altitude	5,480 m (18,000 ft)
Endurance	20 hr



2.18 GSE Vindicator:

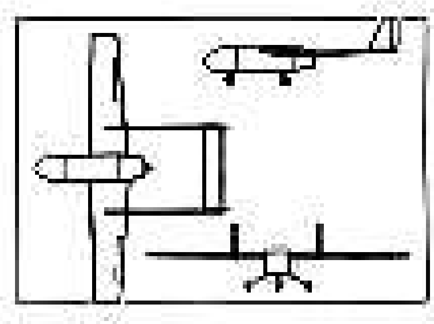
Wing span	7.04 m (23 ft 1.2 in)
Wing area	4.83 m2 (52.0 sq ft)
Length overall	4.79 m (15 ft 8.4 in)
Body diameter (max)	0.46 m (1 ft 6.0 in)
Wetted fuselage area	3 m^2 (ft^2)
Weight empty	268 kg (590 lb)
Max payload	90.7 kg (200 lb)
Max T-O weight	476 kg (1,050 lb)
Max level speed	200 kt (370 km/h; 230 mph)
Cruise/loiter speed	80-90 kt (148-167) km/h
Stalling speed	56 kt (104 km/h; 65 mph)

Ceiling	9,145 m (30,000 ft)
Endurance	20 hr

2.19 KAI Doyosae:

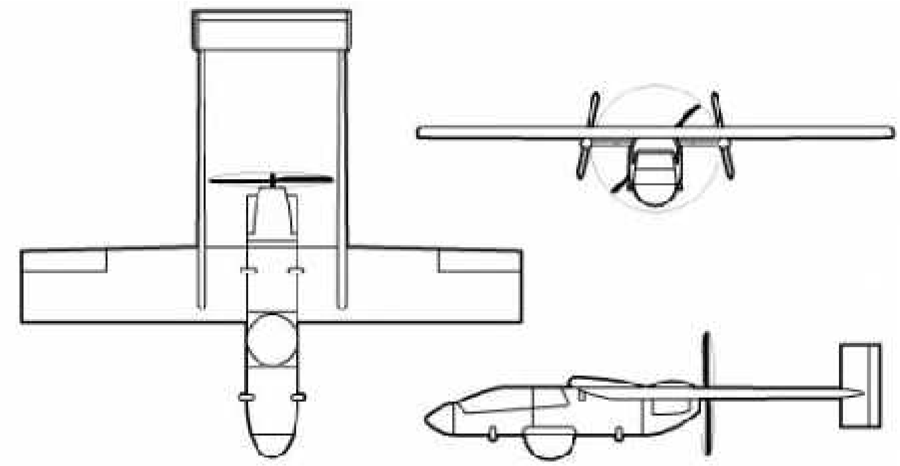
Country of origin: South Korea

Wing span	4.80 m (15 ft 9.0 in)
Wing area	2.84 m2 (30.57 sq ft)
Length overall	3.52 m (11 ft 6.6 in)
Fuselage: Length	2.00 m (6 ft 6.75 in)
Max width	0.45 m (1 ft 5.7 in)
Max depth	0.45 m (1 ft 5.7 in)
Height overall	1.342 m (4 ft 4.8 in)
Tail plane span	1.462 m (4 ft 9.6 in)
Wheel track	1.00 m (3 ft 3.4 in)
Wheelbase	1.17 m (3 ft 10.1 in)
Propeller diameter	0.66 m (2 ft 2.0 in)
Weight empty	113 kg (249 lb)
Max fuel weight	15 kg (33.1 lb)
Sensor payload	0.5 kg (1.1 lb)
Max T-O weight	130 kg (286.5 lb)



2.20 IAI Scout

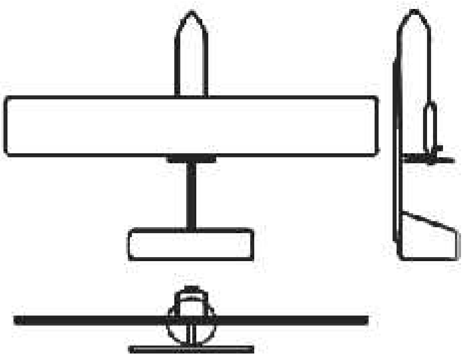
Country of origin: Israel



Capacity: payload	38 kg (84 lb)
Length	3.68 m (12 ft 1 in)
Wingspan	4.96 m (16 ft 3 in)
Height	0.94 m (3 ft 1 in)
Empty weight	96 kg (211 lb)
Gross weight	159 kg (350 lb)
Power plant	piston engine, 16 kW (22 hp)
Maximum speed	176 km/h (109 mph)
Endurance	7 hours 30 min
Service ceiling	4,600 m (15,000 ft)

2.21 S-TEC Sentry Country of origin

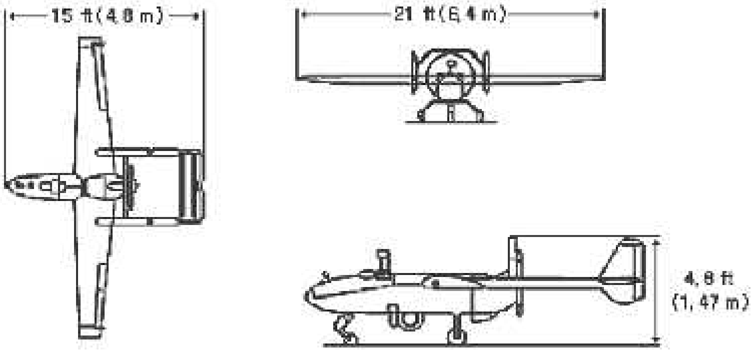
payload	27 kg (60 lb)
Length	8 ft 0 in (2.24 m)
Wingspan	11 ft 0 in (3.35) m
Empty weight	59 kg (130 lb)
Gross weight	109 kg (240 lb)
Power plant	1 × 26 hp (20 kW)
Maximum speed	175 km/h (110 mph)
Endurance	8 hours
Service ceiling	16,000 ft (4,900 m)



2.22 KAI 300 (Bizo, Songgolmae, and RQ-101)

Country of origin: Korea

Wing Span	6.4 m (21 ft)
Maximum Level Speed	185 km/h (100 kt)
Length Overall	4.7 m (15 ft)
Cruising Speed	120–150 km/h (65–81 kt)
Maximum Takeoff Weight	290 kg (640 lb)
Maximum Payload	45 kg (99 lb)
Ceiling	4.57 km (15,000 ft)
Power Plant	1 X 50 hp rotary over 120 km
Operational Radius	over 200 km (108 nm)
Maximum Endurance	6.0 h



2.23 IAI Searcher

Country of origin: Israel

length	5.85 m, 19.19 ft
width	8.54 m, 28.02 ft
height	1.25 m, 4.1 ft
Empty weight	350 kg, 772 lb
Takeoff weight	500 kg, 1102 lb
Max speed	200 kmh, 108 kn
Ceiling	6100 m, 20000 ft

2.24 Tadiran Mastiff:

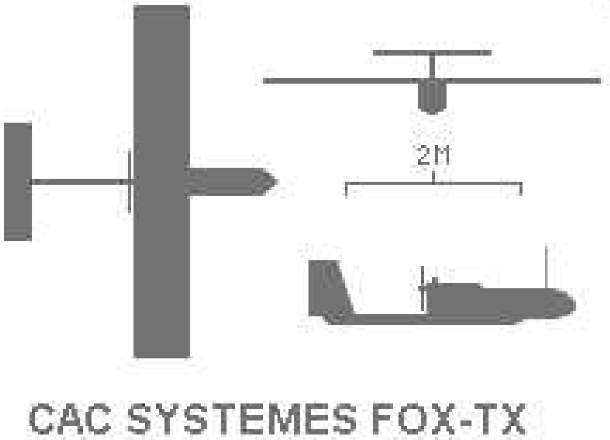
Country of origin: Israel

Capacity payload	37 kg (81 lb)
Length	3.3 m (10 ft 10 in)
Wingspan	4.25 m (13 ft 11 in)
Height	0.89 m (2 ft 11 in)
Empty weight	72 kg (170 lb)
Gross weight	138 kg (304 lb)
Maximum speed	185 km/h (115 mph)
Endurance	7 hours 30 min
Service ceiling	4,480 m (14,700 ft)

2.25 Fox-TX:

Country of origin: France

wingspan	4 meters 13 feet 1 inch
length	2.75 meters 9 feet
payload weight	30 kilograms 66 pounds
empty weight	73 kilograms 161 pounds
launch weight	135 kilograms 298 pounds
endurance	5 hours
maximum speed	180 Km/h 112 MPH / 97 KT
service ceiling	3,000 meters 9,800 feet



2.26 Phoenix:

Country of origin: UK

Length	3.8m
Wingspan	5.5m
Maximum launch weight	175kg
Weight mission pod	50kg
Motor	WAE 342, two stroke, flat twin fuel injection,19kW (25hp)
Propeller	Two blade fixed pitch wooden propeller, 780 mm
Generator	Plessey 900 watts
Maximum speed	85 knots, 155km/h
Flight endurance	5 hours

Radius of operation	more than 50 km
Maximum altitude	2,750m - 9,000ft
Launch & Recovery	Truck-mounted hydraulic catapult, parachute - airbag

3. Literature Review:

3.1 Brief History:

The initial notion about unmanned aerial vehicles came from Austria, in the middle of nineteenth century, precisely on August 1849, when Austrian tried to bomb Italian city of Venice trying to control it using balloons carried with explosives in the form of bombs. These balloons failed to perform their task as planned because wind changed the direction of the balloons and send them away of their targets. The concept strongly urged and encouraged to think about winged aircrafts or as known later as unmanned aerial vehicles, UAVs. Then more efforts and researches were extensively done to ensure success the invention to fly UAVs autonomously for military purposes. (11)

First use of UAVs was by United States Navy in the First world war as aerial torpedoes. After the First World War, a significant progress was happened in the production of UAVs, especially in United States, Germany, and Great Britain. Then Germany succeeded to produce a simple unmanned aircraft that was used to attack Great Britain during the Second World War. (11)

The first UAV that had the ability to land was The Queen Bee, making them reusable for future missions. The Queen Bees were used by the Royal Navy between 1935 and 1947 as targets for anti-aircraft gunners. The remotely controlled Queen Bee had a biplane configuration and the ability to reach speeds of 100 mph, or 160 kmh, travel up to 300 miles (480 km) and a service ceiling of 17000 feet (more than 5000 m). (12)

After the Second World War, the demand for UAVs has increased especially in military applications. Consequently the research on UAVs gained major importance in order to include the performance of more complicated missions. (13)

In 1964, D-21 UAV had been tested for the first time. The D-21 still remains the fastest UAV, as a result of its ability to reach speeds of Mach 4. It had a range of 3000 miles (5000 km), with a service ceiling of 90000 feet (27000 m). It had advantage of an anti-radar coating. The only D-21 built flew 4 missions, crashing during the last one. (14)

Early in 1970's the AQM-34 Firebee, were used by the United State Air-Force in executing different surveillance missions. And later in 2002 was modified to carry and deliver different payloads.

In late of 1970’s, Israeli Air force succeeded to manufacture and produce Pioneer UAV, which was considered in that time as the best UAV in the world. Then it was modified to be used as source of intelligence gathering.

The advantages, capabilities, and flexibility of UAVs make them able to do a lot of tasks and missions which are carried out by manned aircrafts. (15)

The surveillance UAVs of tomorrow may be developed into MAVs, or micro aerial vehicles, minute spies so tiny they can take off and land in the hands palm of their operators'. The U.S., Great Britain, Korea, and Israel are developing MAVs for surveillance use in the future. (14)

Civilian applications of UAVs started in limited areas but in increased. They become preferable for some missions which were considered as dangerous environments for manned aircraft. (16)

However, UAVs cannot replace manned aircrafts in specific missions that require the physical presence of the pilot to control and make the correct decision in the correct time. (17)

3.2 Background:

3.2.1 Tactical UAV’s:

Tactical UAV has been a main focus of the system development and operational employment since the 1970s. (2) In last decades, Tactical UAV has developed significantly into multi-role, multi-mission platforms. Tactical UAV mission set could be expanded to include target designation, in addition to the current applications in Reconnaissance, Surveillance, information superiority, precision strikes against high payoff targets and Target Acquisition. (18)

3.2.2 UAV Tactics:

UAV tactics are defined as the general strategies used by cooperative members in a UAV team to execute a certain required task. Each member in the UAV team is responsible for doing its role to ensure the execution of the whole team operation.

Types of tasks that may be performed by UAVs team: (19)

- Swarming.
- Formation Reconfiguration.
- Task/Target Assignment.
- Dynamic Encirclement.

3.3 UAV Categories:

All figures and images of UAV's mentioned in this part are from references (1),(2) & (20).

3.3.1 According to application:

The advantages of using UAV's support and encourage the wide increase in their applications.

UAVs are classified according to their applications into two main categories as follows:

3.3.1.1 Military uses:

- reconnaissance
- surveillance immediate battlefield
- acts of electronic warfare

- drop some equipment and materials for some of the most important tasks
- station rebroadcast wireless

Remote Sensing -Pipeline Spotting and Powerline Monitoring. -Volcanic Sampling. -Mapping and Agriculture -Geology and Meteorology	**Surveillance** - Law Enforcement - Traffic Monitoring - Coastal/Maritime Patrol - Border Patrol	**Disaster Response** - Chemical Sensing - Flood Monitoring - Wild fire Management
Delivery - Fire fighting - Crop Dusting - Package Delivery	**Entertainment** - Cinematography - Advertising	**Communication Relay** - Internet - Cellular Phone
Transport - Cargo Transport	**Broadcast** - Television/ Radio	**Search and Rescue** - Spotting - Circling

3.3.1.2 Civilian uses:

- Scientific research in various fields.
- Monitor the state of the weather.
- Pollution of territorial waters.
- Participating in search and rescue operations and missing like, humans, cars and planes lost in Sahara or boats lost in the sea.
- Traffic control.
- Control of oil and gas pipelines and identify leaks.

- Control of Strategic centers to monitor variables.
- Reconnaissance and surveillance naturally infested areas such as fires, floods, earthquakes, raging volcanoes.
- Reconnaissance and surveillance borders against infiltrators and smugglers and monitoring and follow them day and night.
- Television camera and direct transport on different occasions

3.3.2 According to Shape:

UAV is classified into four main categories according to the shape and structure as follow:

1. Fixed-wing UAVs

2. Rotor-craft UAVs

3. Flapping-wing UAVs

4. Unconventional UAVs

Fixed-wing and rotor-craft UAVs, have been the dominating shape for various applications and scientific research. However, the flapping-wing UAV has started to attract recent attention.

3.3.3 According to type:

The UAVs are classified according to their types as follows:

3.3.3.1 Micro Air Vehicles: (MAV)

- MAVs generally weigh less than 0.5 lb (0.25 kg).
- MAV chord Reynolds numbers generally less than 10^5.
- Chemical battery electric propulsion is the most prevalent type used.
- MAVs can provide short-range tactical imagery. Because of their maneuverability and small size, these unmanned aircraft might be capable of flying in urban canyons or even inside buildings.

- Tactical reconnaissance for house-to-house fighting.
- Possible equipment: mini-microphones, detectors for radiation or toxic gas.
- Size 10 centimeter, carries a camera with range about 50 meters (red circle).

Sanders Micro STAR, Surveillance MAV (Wo =85 g)

3.3.3.2 Small (Mini)Unmanned Aircraft

- These are larger than MAVs and smaller than small tactical UASs.
- 1–55 lb (0.5- 25 kg) gross weight.
- Sometimes this general category is also called Mini UAS.
- Generally electric powered.
- Endurance ranges between 0.5–2 hr.

Tasuma MSV-10, surveillance UAV (Wo= 4.2 kg)

3.3.3.3 Small Tactical Unmanned Aircraft Systems

- Larger than SUASs and smaller than tactical UASs.
- Weights typically range from 55–200 lb (25-89 kg).

Small Tactical UAV, LUNA X-2000 30 kg (66 lb)

3.3.3.4 Tactical Unmanned Aircraft Systems

The Tactical Unmanned Aircraft Systems should be characterized by:

- Takeoff gross weight ranges between 200 to 1320 lb (90-600 kg).
- Endurance of most tactical UAVs ranges 5–12 hrs
- Maximum altitudes for tactical UAVs normally are 20,000 ft (6000 m) or less.

Tactical UAV AII RQ7 Shadow

3.3.3.5 Uninhabited Combat Aerial Vehicles

That is designed from inception as a strike platform with internal bomb bays or external weapon pylons,

- A high level of survivability, and a takeoff gross weight of greater than 1,320 lb. This class of UA has been called uninhabited combat aerial
- (UCAV) and later an uninhabited combat aircraft system (UCAS).

MQ 9 reaper

3.3.3.6 Medium-Altitude Long Endurance: (MALE)

The Medium-Altitude Long Endurance UAV should be characterized by:

- Gross weights may vary from 1,000–10,000 lb.
- Payload capacities of 200–1,000 lb.
- Endurance range of 12–40 hrs.

Heron UAV

3.3.3.7 High-Altitude Long Endurance

The High-Altitude Long Endurance UAV should be characterized by:

- Endurance greater than 24 hrs.
- The system weight is above 5,000 lb in most cases.

Hale UAV

3.3.4 According to the Launch:

Launchers are devices provide the aircraft with a stabilizing track and launch energy to take the aircraft from stillness to the flight condition. These launch technologies predate conventional landing gear on flying aircraft.

3.3.4.1 Launch systems:

The goal of the use of equipment simply quit is to provide acceleration to the speed of the plane which is enough to raise them in the air, with the provision that provides safety precautions for the plane and its cargo.

3.3.4.1.1 Wheels assembly:

Wheeled' landing is appropriate for large and long range UAVs that use runways. Vehicles are usually equipped with main landing gear and nose landing gear. The purpose of landing gear is to allow acceleration over the runway up to takeoff, and deceleration after touchdown until stop of the engines. Control during takeoff and landing is usually done manually by ground station stick control.

General Atomics MQ-9 Reaper

3.3.4.1.2 Rail Launchers

Rail launchers are common for unmanned aircraft weighing less than 250 kg. This technique enables runway-independent launch. (2)

The advantages of this way it is easy to change the launcher places and facilitates takeoff during night , does not happen any glare, or smoke reveals a of launcher , in addition to lower costs at all.

Rail launch: Raven UAV

3.3.4.1.3 Rocket Launch

In the case of using the method with help of rocket-propelled releasing after the installation of a missile push the plane, after placing the plane above the simple holder with work of the engine, and then running the booster rocket, until the plane reach the required speed; and then the rocket fall to the ground while the plane continued on its flight depending on the engine, and notes that the only exit flame from the rocket boosters cause security problems, as well as detect the launch site, in addition to higher expenses for all other media.

Rocket launch

3.3.4.1.4 Hand Launch:

Most hand-launched unmanned aircraft weigh less than 10 kg with wing spans less than 3 m.

Hand launch

3.3.5 According to recovery:

Its primary mission is to restore the aircraft to the ground safely, or with the least damage possible, after the end of the task of the plane, or in the case of emergency, or in the case of running out of fuel; thus vehicle needs to be summoned and retrieved immediately.

In other way it is defined as transitioning the UAV from a flying state to a non-flying state. The definitions of these states are identical to those described in the preceding launch discussion. In the case of a conventional runway recovery, this phase can be considered landing. Alternatively, the recovery of expendable assets might result in the destruction of the UAV. Regardless of recovery technique, all UAVs must eventually come to rest on the ground, water, or moving platform.

In the restoration process, they used the wheels, skis, umbrellas, using nets, gas baegs or a combination of more than one way. (2)

3.3.5.1. Wheels:

In the case of the use of the wheels, as in conventional aircraft, the router reduces the speed to become a speed of the plane as little as possible, and you need an abundance of caution so as not to hit the plane land at the moment of contact, and preferred to use the corridor paved, or land with grass, and the plane cannot be used wheels in the land of sand, or uneven, or upon a hitch and grit.

3.3.5.2. Parachutes

Umbrellas are used as the key, or backup emergency, and you need to stop the engine before opening the canopy, and suitable for use over all types of land, and do not need to prepare a landing place, and do not need the skill factor guidance, but they are affected by the winds.

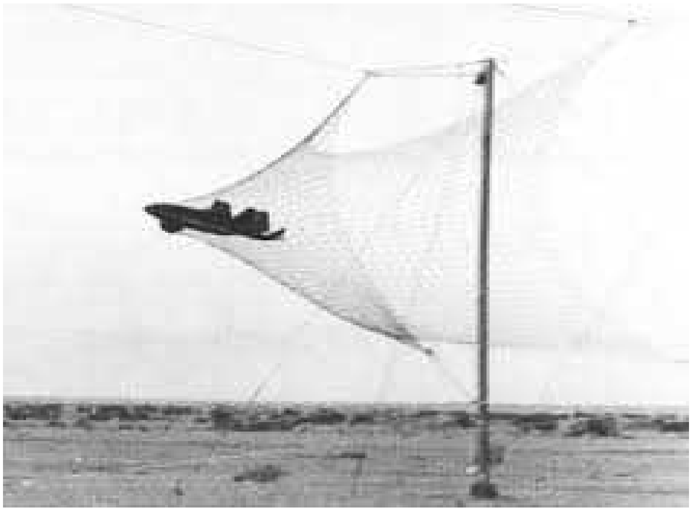
Parachutes recovery

3.3.5.3. Skis recovery:

The use of skis way Faramlah, and characterized by the use of different types of ground; whether cobble, or sand, or even ice, or rock, and check balanced, and greater stability as suitable for landing at night. And rely on our system to eject the assistant skis.

3.3.5.4. Net Recovery

The nets can be used based on my head in the restoration process, and in this case need to be manually steer the plane in the landing phase; since moving toward the network at the lowest speed, and stop the engine before the plane crashing into the network, which absorb the energy of motion. This method is valid and above almost all types of land, as well as over the ships; however, they need to be accurate, and the skill of workers directive.



Net recovery

3.3.5.5. Cable-Assisted Recovery:

A. Arresting Cables on Rigid Horizontal Platforms

Cable-Assisted Recovery

B. Suspended Cables Recovery:

Suspended cable recovery

4. Conceptual Design:

4.1 Introduction:

Designing of UAV passes through three main phases, conceptual design phase, preliminary design phase, and detail design phase.

Conceptual design of the aircraft basically requires answering the next three questions, configuration arrangement, size and weight, and performance of the aircraft.

Design is iterative process as shown in the design wheel in Figure (4-1). In the conceptual design phase, after each new design is analyzed and sized, redrawn is needed to conform the new weights and sizes such as, gross weight, fuel weight, wing size, fuselage size, and other changes. (3)

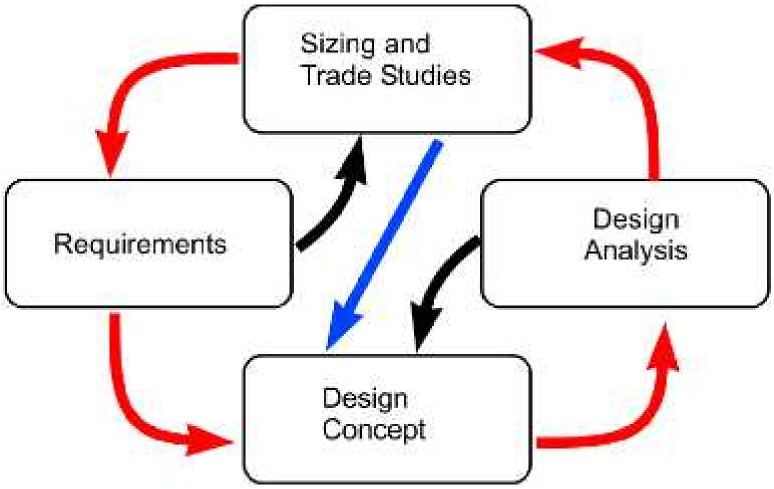


Figure (4-1) design wheel

The following elements of the airplane are defined in Conceptual Design phase:

1. Takeoff mass

2. Wing loading and specific thrust.

3. Dimensions and shape of the wing.

4. Dimensions and shape of the fuselage.

5. Dimensions and shape of command and stabilizing surfaces.

6. Dimensions, kind and shape of high lift devices.

7. Selection of propulsion.

8. Take off and landing distance calculation.

9. Static stability calculations and determination of required deflection of command surfaces.

10. Concept of the UAV.

11. Price of the UAV.

4.2 UAV mission profile:

For performing the UAV mission, the quantity of fuel supplied, which represents the total fuel weight, is divided to three sections. First section of fuel amount is consumed through the mission. The second of fuel amount is reserved as trapped fuel, which is not allowed to be used through the UAV mission. The third section is the quantity required by design specifications. (2)

Fuel weight is proportional related to takeoff weight (W_f/W_o), In addition to the empty weight and the payload weight, and it is independent of the aircraft weight design calculations. Estimation of fuel quantity is dependent on the mission performance using fuel consumption and aerodynamics criteria.

The conventional mission profile of UAV's is shown in figure (4-2) below, and it is divided to segments as follows:

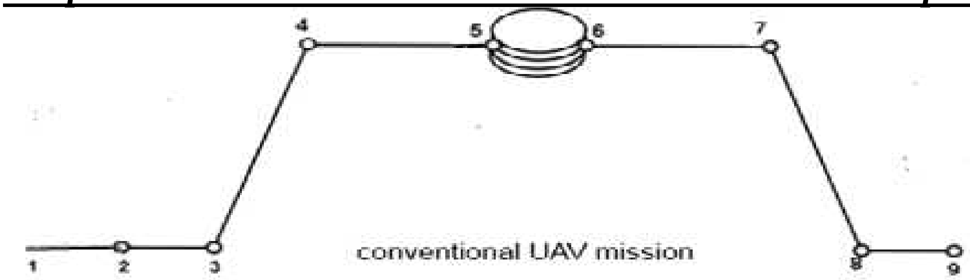


Figure (4-2) mission profile for conventional UAV

- 1-2. Starting and taxing.
- 2-3. Take off.
- 3-4. Climbing.
- 4-5. Cruising.
- 5-6. Loitering.
- 6-7. Cruising.
- 7-8. Descent.
- 8-9. Taxi and cut off.

Each of the mission segments consumes fuel. It is required to estimate amount of fuel necessary to accomplish the mission profile. Some of the fuel consumption during mission segment can be calculated, for other mission segments some statistical estimation is used.

Fuel in the tanks cannot be consumed totally always some amount is trapped in the corners of the tank and in supply lines. Also some additional fuel is needed to be kept as reserve in the case that runway is busy when vehicle need to circle around waiting for landing. This amount is estimated to be 30 minutes cruising or 10% of total fuel amount w_f as considered in this work. (3)

4.3 Fuel fraction phase Relationships:

The fuel fraction for each phase is defined as; Fuel fraction phase = (W_i/W_{i-1}) where W_i is the weight of the vehicle at the end of mission segment and W_{i-1} is the weight of the vehicle at the beginning of the mission segment. According to the table (4-1). (4), each vehicle category has its weight fraction for each phase. The warm up, taxing, take off, climb, descent, and landing weight fractions are specified in the table below, cruising and loitering weight fractions can be calculated using some formulas such as Breguet range equation and endurance equation.

Table (4-1) Fuel mission segments for different types of aircrafts.

Aircraft engine type	**weight fraction (W_i/W_{i-1})**					
	Warm up	taxi	Take off	climb	descent	landing
Homebuilt	0.998	0.998	0.998	0.995	0.995	0.995
Single engine	0.995	0.997	0.998	0.992	0.993	0.993
Twin engine	0.992	0.996	0.996	0.990	0.992	0.992
Agricultural	0.996	0.995	0.996	0.998	0.999	0.998
Business jet	0.990	0.995	0.995	0.980	0.990	0.992
Military	0.990	0.990	0.990	0.980	0.990	0.995

Conceptual design process is illustrated in figure (4-3) which represents refined sizing and performance optimization. In the optimization stage of the process, optimization techniques are conducted to get the lightest or lowest-cost aircraft that will satisfy all the requirements of performance and the design mission. The result of this optimization is better weight estimations of takeoff and fuel to meet the requirements of the design mission. Also revision of wing size and engine is required as a result of optimization. (3)

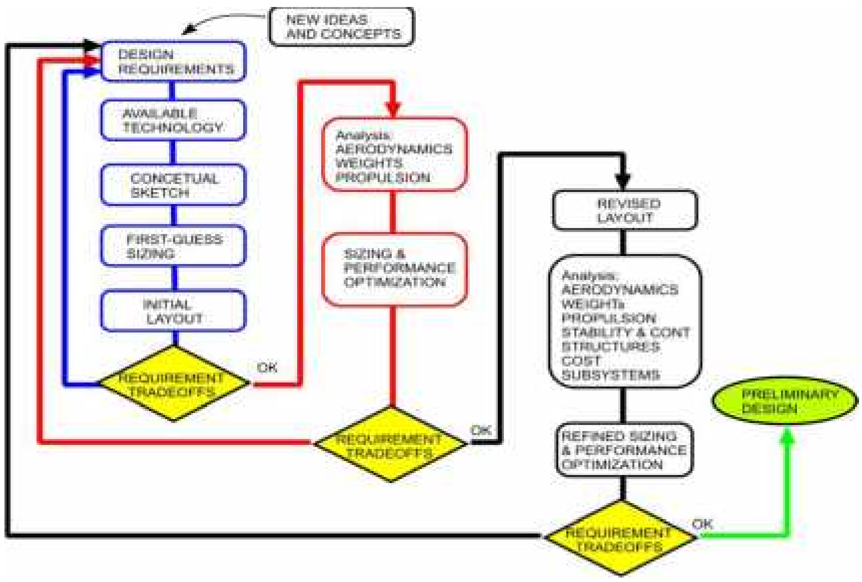


Figure (4-3) conceptual design phase

4.4 UAV sizing:

The main parameter of UAV weight estimation is takeoff gross weight Wo, this Parameter defines the most parameters of weight estimation relations. takeoff gross weight Wo has dramatically effective on cost, competitive, environment, and suitability for various operations. Therefore, the sizing discussion starts with evaluative methods for defining this important parameter. (2)

4.5 Weight estimation:

4.5.1 Introduction:

Lowest weight of an aircraft design is a subject of the most importance. Although minimization of weight acquired at some initial additional cost, the effect on whole operating costs are greater for most high performance designs. To obtain a weight minimization and the associated costs rely upon the design process phase.
Weight of the aircraft is strongly affected by choice of the aircraft layout, geometry and the formation details during the initial conceptual design phase.
The design layout should be attentional optimized and high accuracy of the initial weight prediction is a basic. The prediction of weight is necessary to set an objective for the structure and design systems, additionally to make an assessment of the design qualities. Estimation of weight minimizations are generally for constant design performance, unless engine performance is limited and doesn't allow this. Growth of takeoff weight is associated with growth in any aircraft component weight. Howe ever the component weight increase is caused by a design change to improve performance; the final result may be a takeoff weight reduction. (22)

The knowledge of the weight of the vehicle, including the components weight, empty weight, and the design takeoff weight is the key of successful design process of an aircraft. The preliminary stage of design is the next stage which follows the conceptual design phase, where the sizing and aircraft weight estimation were done. These weight values were required to determine the performance and dimensions of the aircraft, however these values are difficult to predict without some form of formal analysis. The methods suggested by Roskam, Raymer, Kundu, Gundlach, and others are developed from the statistical analysis. This analysis reveals a relationship between vehicle takeoff and empty weights, based on the statistics of similar aircraft that have already been designed and which may or may not be in active service. The foundation of such analysis is technology limitations, with the premise being that most designs are limited by current and existing technology, and thus must satisfy some form of relationship dictated by these limitations (Roskam, 2004b). This graph is known as a ’Technology Diagram’ and it was used to find the required design takeoff and empty weights based on specifications from the project specifications.

4.5.2 Materials:

Material selection for UAV components manufacture is driven by two main factors, namely specific strength and weight. The availability of materials was a factor considered for the construction of the UAV’s. Composite materials are commonly used where an attractive material to be used for most components; however the properties of these materials needed to be quantified in order to ensure structural integrity.

A variety of commonly available materials were identified to be used in the construction of the aircraft. The structural requirements, construction method and manufacturing methods were considered for each component individually to select the appropriate material. Properties of each material need to be carefully considered as each component has different requirements for strength and electrical transparency. Table (4-2) shows the materials that were identified as both available and suitable for the purposes of a construction process.

In recent years, composite materials have become more popular in the aviation industry. For example, more than 25% of Airbus A380 is made of composite materials (Airbus 2012). Making the UAV out of composite materials may not be the best choice for sunny atmosphere since ultra-violet radiation deteriorates properties of resin matrix. Using a combination of composite materials and aluminum would be the most attractive option, as the composites are needed to lower the overall weight of the aircraft while aluminum cover will protect resin from ultraviolet radiation. (21)

Table (4-2) Shows the materials for aircrafts manufacture and their properties

Material type	Material name	Density Kg/m^3	Yield stress (MPa)	Share stress (MPa)
Cloth	E grade fibreglass	1965	724	70.3
	S grade fibreglass	1439	1509	51
	Carbon fibre - Weave	1600	345	35
	Carbon fibre - Uni	1600	550	70
Resin	Epoxy resin	1550	96.5	N/A
	Polyester resin	1810	185	N/A
Sandwich Core	Isolite expanded PS foam	24	0.14 (comp)	0.26
	Extruded blue styrofoam	40	0.7	0.4
	Honeycomb core	30	1.4	1.1
	Closed cell foam core	41	0.5	0.6
Filler	Microsphere filler	250	N/A	N/A
	Milled glass	1360	130	N/A
Laminate	Ply wood		31	6.2 (edge)
	Balsa wood	160	73 (axial)	1.1
Metal	Aluminum alloy 2024-T3	2780	345	283
	Mild steel threaded rod	7870	260	

4.5.3 Takeoff weight estimation:

Takeoff weight is the total designed weight of the UAV as it begins its mission. Design takeoff gross weight can be broken into empty weight, payload weight, and fuel weight. Empty weight can be broken to structure weight, landing gear weight, engine weight, and anything else is not considered as payload, or fuel.

For UAVs:

$$Wo = W_e + W_{pl} + W_f \qquad (1)$$

Where, W_{pl} is the payload weight, and W_f is the weight of the stored fuel, W_e is the empty weight of the vehicle.

The payload weight is given and it is known as design requirements, but empty and fuel weights are unknown and they related to total takeoff weight.

Empty weight and fuel weight both are fractions of takeoff weight.

Takeoff weight of the vehicle given in (1) can be modified by introducing the following ratios:

$$W_f/W_o, \quad \text{and} \quad W_e/W_o$$

So

$$Wo = (W_e/W_o)*W_o + (W_f/W_o)*W_o + W_{pl}$$

Expressing the payload weight as a function of other weights we get:

$$Wo - (W_e/W_o)*W_o - (W_f/W_o)*W_o = W_{pl}$$

Finally takeoff weight is expressed from this equation as:

$$W = \frac{Wp}{1-(We/Wo)-(Wf/Wo)} \qquad (2)$$

Wpl is payload weight and usually is specified in design requirements; *Wf/Wo* is fuel weight fraction and is calculated from UAV's mission segments; *We/Wo* is empty weight fraction which has to be estimated somehow. Usual approach is to study historical empty weight fractions of already existing vehicles.

4.5.4 Empty weight estimation:

The empty weight is a function of aircraft load, both on the ground and during the flight, which depends on the maximum takeoff weight. The load in the air is a result of aircraft speed–altitude abilities and wind. A higher speed capacity would increase the operational empty weight fraction to retain structure integration. (5)

Empty weight fraction can be estimated statistically as shown in figure (4-5):

Empty fraction weights for UAVs weights between 100-500 kg range between 0.49 up to 0.71 and tend to rise with increasing of takeoff weight.

Empty weights of selected existing vehicles are represented by circles in the diagram in Figure (4-5). It could be observed weak dependency between takeoff weight W_o, and empty weight fraction W_e/W_o. Functional relationship could be obtained by passing some curve in the least square sense through these points. Various such functions are used in available literature, but most frequently exponential form is used, the same form used here.

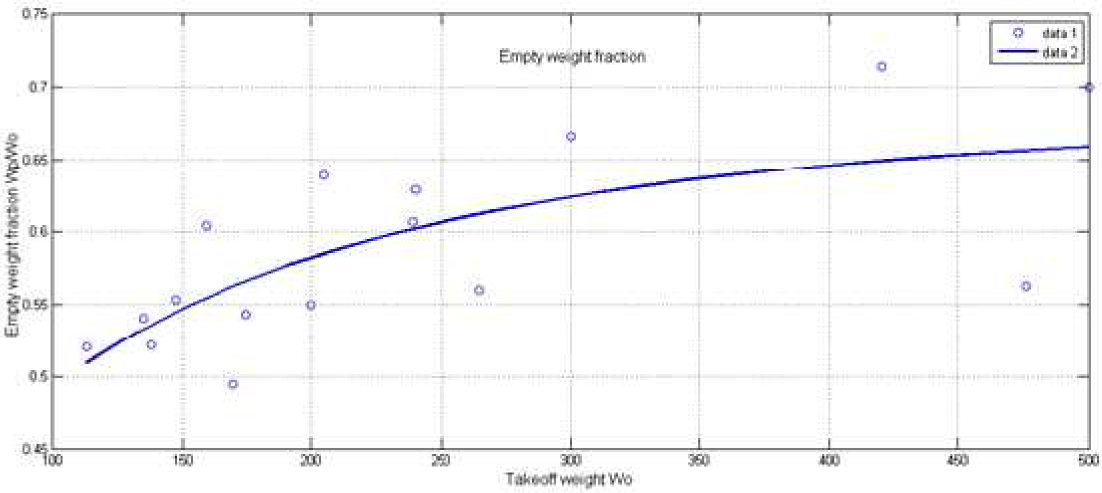


Figure (4-5) Empty weight fraction versus takeoff weight

Equation of the empty weight fraction curve obtained by least square method is:

$$We/W_o = \exp(-0.0679 \times \log W_o^2 + 0.9174 \times \log w_o - 3.4951)$$

For example when takeoff weight w_o equals 240 kg, empty weight fraction is $w_e/w_o = 0.6$.

Equation (3), from Raymer, has suggested the following relationship for determination of empty weight fraction:

$$\mathbf{W_e/W_o = A * W_o^c * k} \quad (3)$$

k=1 for fixed sweep, coefficients **A** and **k** are selected from Table (4-3) which is copied from Raymer's book (3). If we select coefficients **A** and **k** for homebuilt aircrafts and apply this formula the following is obtained:

A = - 0.09, c = 0.99, K=1 for fixed sweep, and $W_o=240\ kg$

$$W_e/W_o = 0.60$$

Thus empty weight is $W_e = 240 \times 0.6 = 144\ kg$,

Table (4-3) Coefficients A, k, and c for some aircrafts types

$W_e/W_0 = A W_0^C K_{vs}$	A	C
Sailplane—unpowered	0.86	−0.05
Sailplane—powered	0.91	−0.05
Homebuilt—metal/wood	1.19	−0.09
Homebuilt—composite	0.99	−0.09
General aviation—single engine	2.36	−0.18
General aviation—twin engine	1.51	−0.10
Agricultural aircraft	0.74	−0.03
Twin turboprop	0.96	−0.05
Flying boat	1.09	−0.05
Jet trainer	1.59	−0.10
Jet fighter	2.34	−0.13
Military cargo/bomber	0.93	−0.07
Jet transport	1.02	−0.06

K_{vs} = variable sweep constant = 1.04 if variable sweep
= 1.00 if fixed sweep

4.5.5 Payload weight estimation:

Payload weight is known as it is specified in the requirements of the design process. In our case Payload fraction can be estimated statistically Figure (4-6), by taking the payload average weight of several UAVs. The payload fraction weights vary from 0.16 to 0.28, and decreases slightly with increase of takeoff weight. Least square fit for payload weight is obtained from Figure (4-6), and is given as:

$$W_p / W_e = 0.2708 \times \log wo^2 - 3.2833 \times \log wo + 8.1398$$

For example At Wo = 250 kg, pay load weight fraction weight is 0.18, thus W_{pl} = 45 kg

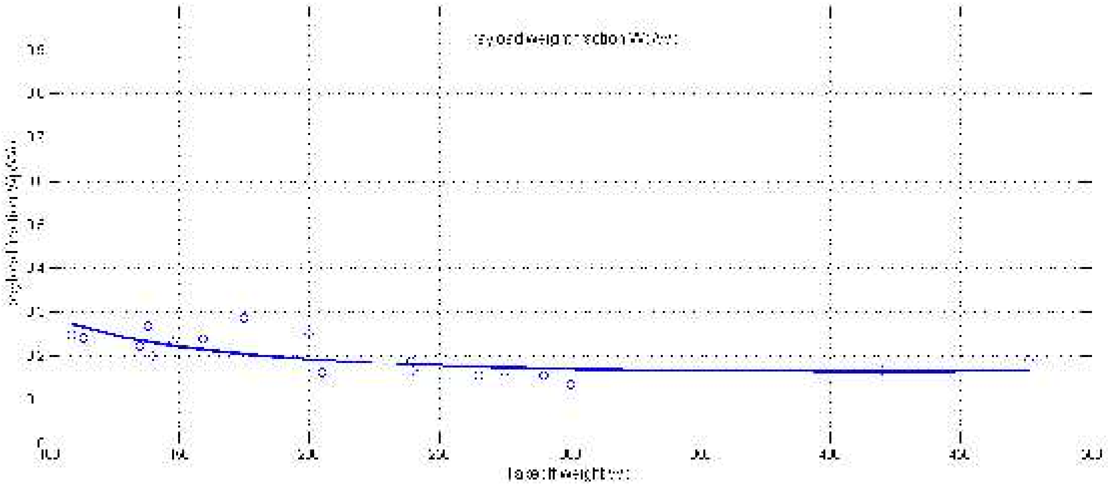


Figure (4-6) Payload fraction weight versus takeoff weight

4.6 Specific fuel consumption:

Specific fuel consumption (c), Table (4-4), is defined as the rate of consumed fuel divided by trust. And it is measured in (kg of fuel per hour) per kg, which it is 1/hr.

Table (4-4) Specific fuel consumption

Propeller: $C = C_{bhp} V/(550\eta_p)$ Typical C_{bhp} and η_p	Cruise	Loiter
Piston-prop (fixed pitch)	0.4/0.8	0.5/0.7
Piston-prop (variable pitch)	0.4/0.8	0.5/0.8
Turboprop	0.5/0.8	0.6/0.8

4.7 Maximum takeoff weight and wing area relationship:

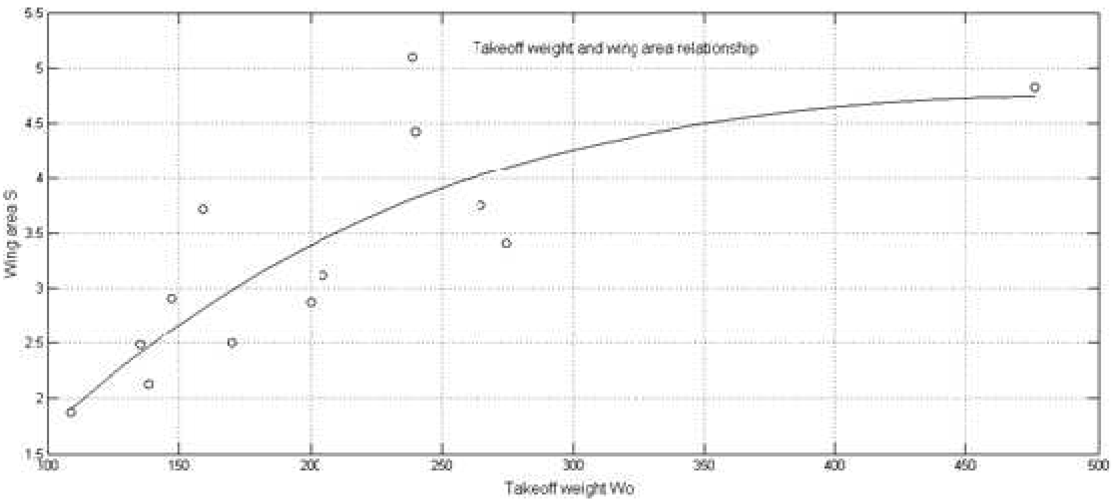
Size of the vehicle is determined in such way that vehicle satisfies all flight mechanic requirements and constraints. By the word size it is assumed geometric lengths, areas, or mutual distances between aerodynamic surfaces.

Initial guess of aerodynamic surface sizes can be obtained by statistical trend analysis, but later must be confirmed by analyzing flight mechanical constraints.

Main wing reference area of the flaying vehicle S_W, as a function of W_o is shown in Figure (4-7).

This figure is helpful for getting a starting value of wing area in a preliminary sizing for a new aircraft design that would be refined through the sizing analysis. (5)

Required flight characteristics determine the size of reference wing area. Smaller takeoff and landing speeds for the same airfoil cross-section of the wing require larger wing reference area. Larger wing also means heavier wing but not linearly, because wing aspect ratio dictates the size of the airfoil, and moments of inertia of the wing cross section are quadratically proportional to the size of the airfoil. The larger wing area with increased aircraft takeoff weight is necessary to afford the flight operating conditions. A large wing reference area is required for performance at low speed of the aircraft, which precedes the performance requirements during cruise. Therefore, wing-sizing provides the minimum wing planform area to satisfy both the takeoff and the cruise requirements. (5)



Figure(4-7) Wing area versus takeoff weight

Equation of the wing area curve obtained by least square method is:

$$S/W_o = \exp(-0.3817 \times \log W_o^2 + 4.7632 \times \log W_o - 13.3012)$$

4.8 Wing span of the aircraft:

4.8.1 The Square-Cube Law:

From the following example, it can be seen that the increase in weight is faster than the increase in area (the subscript 1 represents the small cube and the subscript 2 represents the larger cube):

A= Area, V= volume, L= length

$A2 = A1 \times (L2/L1)^2$, a 4-fold increases from 6 to 24 square units

$V2 = V1 \times (L2/L1)^3$, an 8-fold increase from 1 to 8 cube units

Applying this concept to a wing, increasing its span would increase its volume faster than the increase in surface area, although not at the same rate as for a cube. Volume increase is associated with weight increase, which in turn would require stiffening of the structure, thereby further increasing the weight in a cyclical manner. This is known as the *square-cube law* in aircraft design terminology. This logic was presented a half century ago by those who could not envisage very large aircraft.

$$\text{weight} \propto \text{span}^3 \quad \text{wing planform area, } S_w \propto \text{span}^2$$

then,

Wing-loading, $W/S_w \propto b$

Increasing aircraft wing size would increase its volume faster than the increase in surface area.

With steady improvements in new-material properties, miniaturization of equipment, and better fuel economy, wing span is increasing with the introduction of bigger aircraft.

Wing span is a function of aspect ratio which is proportional to the square of the wingspan, and the wing area is directly related with the square of the wing span. Figure (4-8) shows the relationship between takeoff weight and the wing span.

$$Sw \propto \text{span}^2$$

Equation of the empty weight fraction curve obtained by least square method is:

$$b / W_o = 0.1596 \times \log wo^2 - 1.3169 \times \log wo + 6.1444$$

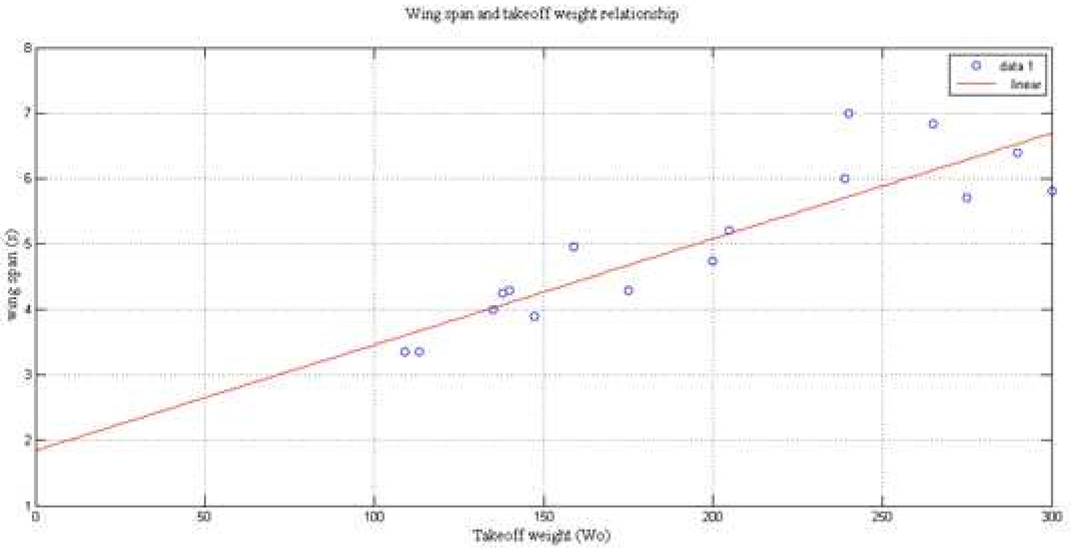


Figure (4-8) Wing span and takeoff weight relationship

4.9 Aspect ratio of the wing:

Aspect ratio of the wing (A) is used to indicate to the wing efficiency, and it is mathematically expressed as the square of the wing span divided by the referenced wing area, or wing span divided by the cord in case of rectangular wing shape.

Aspect ratio is an important parameter in determining the performance. Figure (4-9).

$$A = b^2/s$$

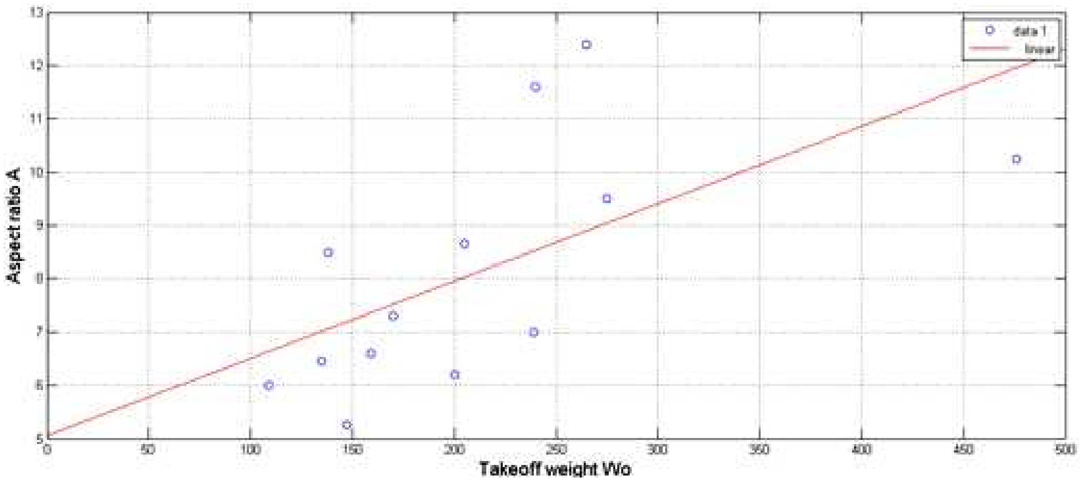


Figure (4-9) represents aspect ratio and takeoff weight relationship.

All these trends should not be used as primary design tool since driving criterion is flight mechanics characteristics of the UAV. These trends can be used as first guess of the geometric parameters of the wing.

5. Study case:

5. 1. Initial estimation of takeoff weight:

In this thesis, application of the developed charts is illustrated on the same example. Let us suppose that we wish to design UAV of the following characteristics:

- Range R = 400 km
- Max speed = 180 km/h
- Cruising speed V_{cruise} = 150 km/h
- Payload mass W_{pl} = 40 kg
- Takeoff from runway (runway length not shorter than 300 m)
- Landing on runway (runway length not shorter than 300 m)
- Landing speed 40 km/h
- Loitering time 2 h
- Maneuvering load factor n=4
- Landing load factor n=20
- Minimum climb angle 5 deg (G~1/12).
- Minimum climbing speed V=2 m/s
- Mission segments are given in the figure (5-1). Cruise distances between 2 and 3, and between 4 and 5 are equal to 200km. Loitering time between points 3 and 4 is 2h.

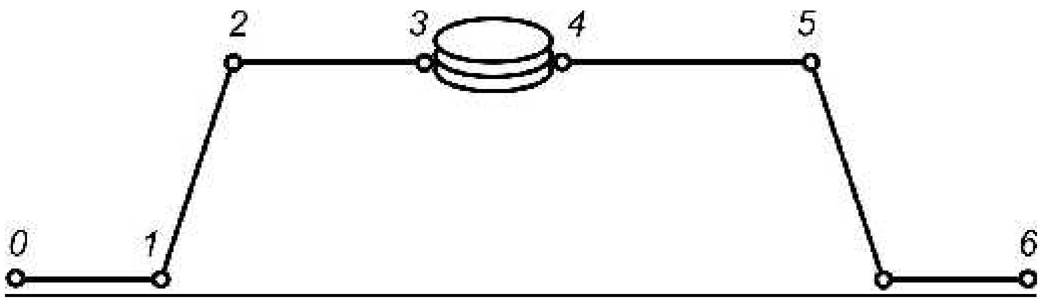


Figure (5-1) Fuel mission

5.2 Takeoff weight estimation:

Specific fuel consumption (C) from Raymer in table (5-1)

Table (5-1)Specific fuel consumption

Consumption Cpower lb=hp _ h (N=kW h)	**Cruising**	**Loitering**
Fixed pitch propeller	0.4 (2.38)	0.5 (2.98)
Adjustable pitch propeller	0.4 (2.38)	0.5 (2.98)
Turboprop	0.5 (2.98)	0.6 (3.58)

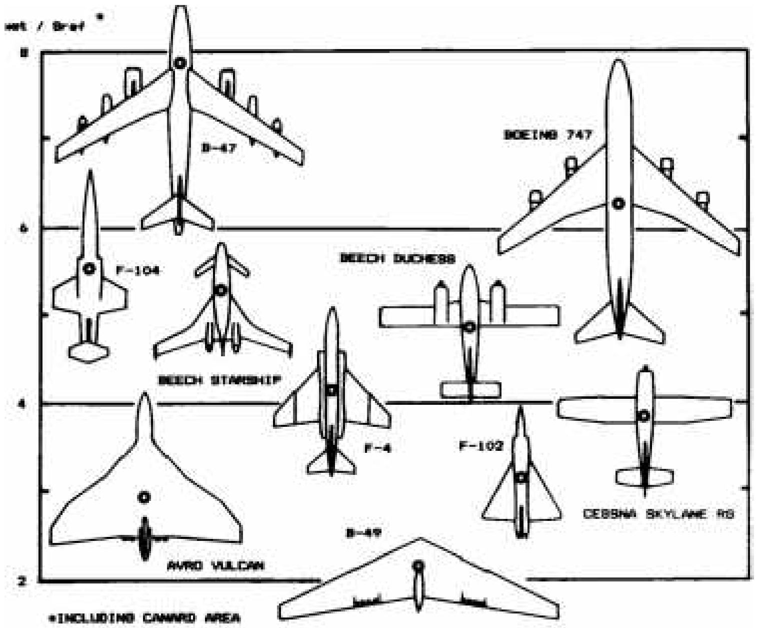


Figure (5-2) Wetted area ratio

5.3 Wing Aspect Ratio:

Aircraft generally are considered to have a high aspect ratio or low aspect ratio. High aspect ratio wings have lower induced drag meaning a higher lift-to-drag ratio, as well as a higher lift-curve slope. In comparison, a lower aspect ratio wing has reduced weight, better aeroelastic properties and improved lateral stability (3).

Aspect ratio of UAVs is in range (6 – 9). (2)

AR is selected to be 8.

From figure (5-3), wetted area ratio is:

$$S_w/S_{ref} = 4$$

5.4 L/D Estimation:

Lift to drag ratio or L/D, is a measure of the design aerodynamic efficincy of the UAV, and at subsonic speeds, L/D is strongly affected by two parameters of the design; wing span and wetted area.

During the horizntal flight, left is equal to the aircraft weight. So L/D ratio depends only on the value of drag D.

Dominant component of drag in subsonic speeds for UAV's are induced drag, friction drag and pressure drag.

Drag generated by the lift, it is called induced drag, this kind of drag is dominantly a function of the wing aspect ratio.

The other type of drag is caused by contact of air with the UAV skin, this kind of drag is called parasite drag or friction-skin, this kind of drag is directly related to total exposed wing area (wetted area) of the UAV.

Pressure drag is generated by unbalanced pressure over vehicle surfaces.

Cruise weight fraction and loitering weight fraction depend on some other requirements such as, range, velocity, and loitering time. (3)

From Figure (5-3), for wetted aspect ratio = $A/ (S_w/S_{ref}) = 2$ and from the curve which corresponds to fixed landing gear prop aircraft.

$(L/D)_{max} = 13$

Since chart is given for higher Reynolds number flows, in absence of any better source we will assume that maximum finesse ratio is somewhat reduced (say by factor 0.85), thus:

$(L/D)_{max}=13 \times 0.85=11$

(L/D) at cruise = $(L/D)_{max}=11$, (L/D) at loitering = $0.866\ (L/D)_{max}=9.5$

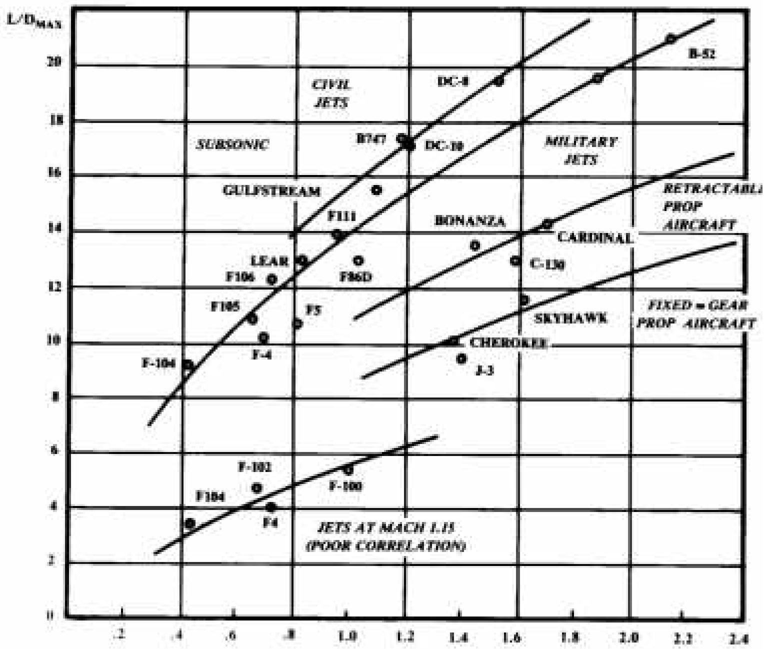


Figure (5-3) Maximum lift to drag ratio trends

Table (5-2) Lift to drag ratio at cruise and loiter mitions

Aircraft engine	Cruise	Loiter
Jet	$0.866\ (L/D)_{max}$	$(L/D)_{max}$
Prop	$(L/D)_{max}$	$0.866\ (L/D)_{max}$

5.5 Fuel mission:

The conventional mission profile of UAV's is shown in figure (5-1), and it is divided to segments according to Raymer as follows:

- Warm up and takeoff,

$$W_1 / W_0 = 0.97$$

- Climb,

$$W_2 / W_1 = 0.985$$

- Cruise for first segment,

$$R = \frac{V}{C} \times \frac{L}{D}\ ln \left(\frac{Wi}{Wi-1}\right) \tag{5-1}$$

$$W_i/W_{i-1} = \exp . \frac{-R}{V(\frac{L}{D})}$$

$$W_i/W_{i-1} = \exp . \frac{-2 \quad \times 0.4}{1 \quad (1 \ )}$$

W$_3$ / W$_2$ = 0.953

- Loiter,

$$\frac{L}{D} = 0.866\ (L/D)_m$$

$$E = \frac{L/D}{C} \ln \left(\frac{Wi}{Wi-1}\right) \tag{5-2}$$

$$W_i/W_{i-1} = \exp . \frac{-E}{(\frac{L}{D})}$$

$$W_i/W_{i-1} = \exp . \frac{-2 \times 0.5}{(0.8 \quad \times 1 \ )} = 0.90$$

W$_4$ / W$_3$ = 0.913

- Cruise for second segment,

$$R = \frac{V}{C} \times \frac{L}{D} \ln \left(\frac{Wi}{Wi-1}\right)$$

$$W_i/W_{i-1} = \exp . \frac{-2 \quad \times 0.4}{1 \quad (1 \ )}$$

W$_5$ / W$_4$ = 0.953

- Landing,

W$_5$ / W$_4$ = 0.995

$W_6/W_0 = 0.97 \times 0.985 \times 0.953 \times 0.9 \times 0.953 \times 0.995 = 0.78$

$$(Wf/\,Wo) = 1.06\times (1\text{- }0.78) = 0.233$$

From equation (5-3) *Wo* can be calculated if we know fuel fraction and empty weight fraction. Table (5-3)

$$W = \frac{Wp}{1-(We/Wo) - (Wf/\,Wo)} \tag{5-3}$$

$$W = \frac{40}{1-(We/Wo) - 0.233}$$

Table (5-3) Takeoff weight values for different value of empty weight fraction

Wo (guess)	We/Wo	Wo (calculated)
250	0.6	239.5
240	0.595	232.5
230	0.59	226
220	0.585	220

First estimation of takeoff weight W_o for the previous characteristics is: W_o = 220 kg, and empty weight fraction is We/Wo = 0.585. Figure (5-4)

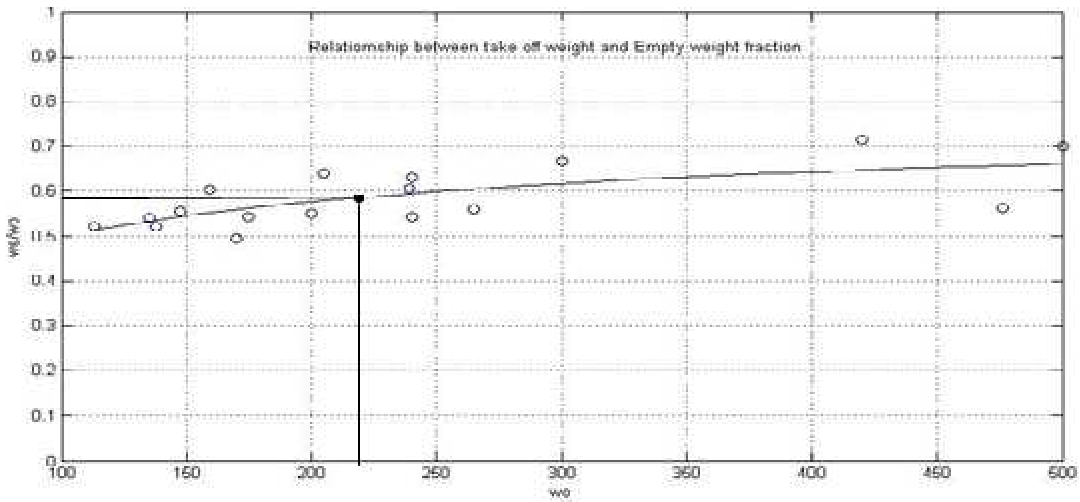


Figure (5-4) Empty weight fraction and takeoff weight relationship

So our initial estimate of takeoff weight is W_O*=220kg*. The next step is to estimate UAV's geometric parameters based on this estimated initial weight.

5.6 Wing loading:

Wing-loading, W_o/S_w, is defined as the ratio of the takeoff weight to the wing reference area. This is an important parameter in the aircraft sizing and it has a significant role in the design process.

The most important parameters in airplane performance are the calculation of T/W and W/S. So it is important to optimize these parameters.

T/W and W/S are connected in performance calculation in number of ways. For example for takeoff:

- Small wing load (lower takeoff speed, requires smaller T=W, but smaller T=W reduces acceleration (longer runway).
- Bigger W=S smaller wing (lighter vehicle), higher takeoff speed, more powerful engine is needed (heavier vehicle. Greater power increases acceleration faster takeoff speed is achieved what shortens takeoff runway.
- Stalling speed is usually critical demand.
- This speed does not depend on engine size.
- Based on this speed we can determine initial wing loading.
- Knowing W/s we can calculate necessary T/W for various performance demands.
- However, T/W better fit to statistical trends for certain airplane categories, thus we have to determine W=S.

5.6.1 Wing loading and Takeoff weight relationship:

Wing-loading, W_o/Sw, is defined as the ratio of the takeoff weight to the wing reference area. This is an important parameter in the aircraft sizing and it has a significant role in the design process.

The influence of wing-loading is shown in figure (5-5). The tendency is to have lower wing-loading for smaller aircraft and higher wing-loading for larger aircraft operating at high-subsonic speed. High wing-loading requires the assistance of better high-lift devices to operate at low speed. (5)

Wing loading has an effective influance upon sized UAV takeoff weight. The wing loading has an oppossite relationship with the wing size, if the wing loading is reduced, then the wing size is larger. Reduced wing loading may improve performance of the vhicle, but this will increase the drag and the empty weight of the aircraft as a result of the larg wing which will increases takeoff weight to perform the mission.

Wing loading influences landing and takeoff distance, climb rate, stall speed, and turn performance, and it determines the design lift coefficient and impacts drag through its effect upon wetted area and wing span.

To insure that the wing provides enough lift in all circumstances, the designer should select the lowest of the estimated wing loading. (3)

Wing loading equation is given as follows:

$$\frac{W}{S} = \frac{1}{2}\rho V_s^2 \quad C_{L_m} \tag{5-4}$$

The stall speed has an important contribution to flying safety, and it is determined by the wing loading and the maximum lift coefficient, and landing speed which has an important influence factor in defining the landing distance, is defined by the stall speed.

Landing speed must be bigger than stall speed,

$V_{land} = k\ .V_{stall}$

Where, k = 1.3 for civil airplanes, and k = 1.2 for military airplanes. (3)

For UAV’s we will consider k = 1.37

From the data, landing speed = 40 km/h

Vs = 40/1.37 = 29.2 km/h = 8.11 m/s

Air density ρ at h=0 is, $\rho = 1.225\ k\ \ /m^3$

Maximum lift coefficient C_{L_m} is defined from Figure (5-6) which is given for manned vehicles and big airplanes, $C_{L_m} = 1.5\ a\ \ s \quad a \quad \Lambda = 0$. Since tactical UAVs have smaller chord lengths and thus smaller flight Reynolds number we have to reduce this lift coefficient by some amount. Let us take initially 85% of this value so $C_{L_m} = 1.5x0.85 = 1.28$

$$\frac{W}{S} = \frac{1}{2}x1.225x8.11^2x1.28 = 51.4 \text{ Kg/m2}$$

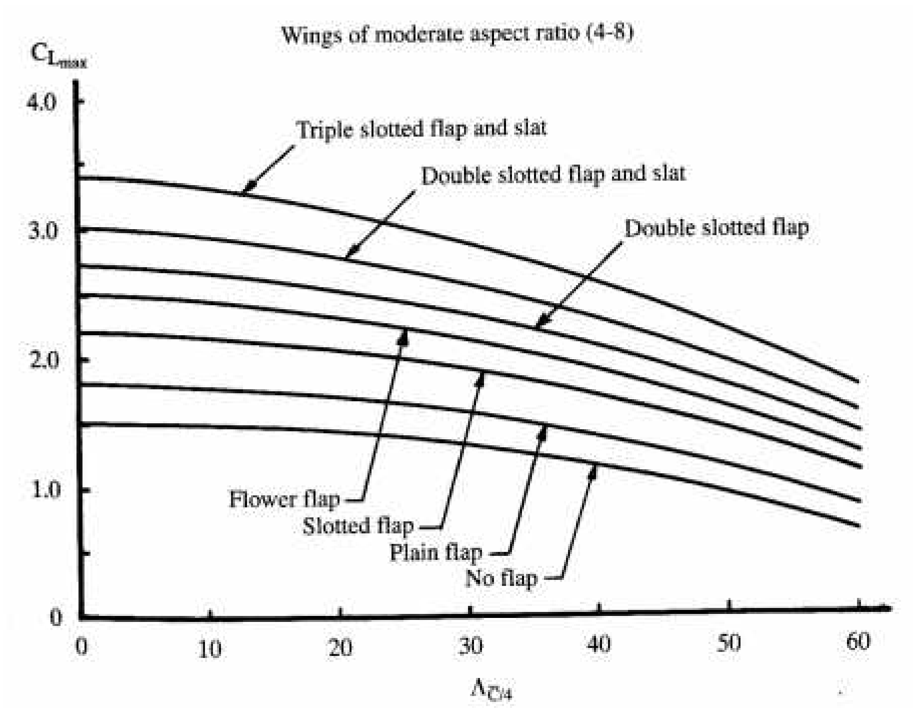


Figure (5-6) Maximum lift coefficient C_{L_m} value for different sweep angle

Equation of curve of wing loading and takeoff weight is:

(Sw/Wo) / Wo = exp (0.3817 x log wo^2- 3.7632 x log (Wo) + 13.3012);

The relationship between wing loading and takeoff weight is illustrated on Figure (5-5).

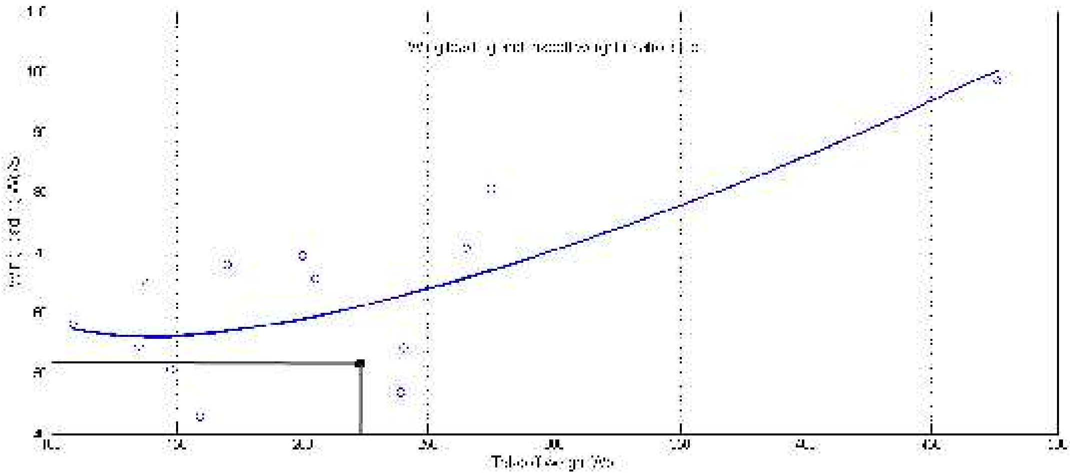


Figure (5-5) wing loading and takeoff distance relationship

5.6.2 Wing loading landing distance:

Landing distance (S_{land}) is the horizontal distance the airplane covers from the aircraft first touch till it comes to a complete stop. The approach to landing begins at the height of 50 ft (15.2 m). The flight speed at this point is called ‘Approach speed’ or landing speed.

Wing loading is largely determines the landing distance. Wing loading affects the approach speed, which must be a certain multiple of stall speed (1.3 for civil aircrafts, 1.2 for military aircrafts).

First-guess estimation of the total landing distance in feet, including obstacle clearance is approximately: (3)

$$S_{land} = 0.3\times (\text{approach speed in knots})^2 \ \text{ft}$$

$$S_{l} = 80 \times \left(\frac{W}{S}\right) \times \left(\frac{1}{\sigma \times C_{L_m}}\right) + S_a \qquad (5\text{-}5)$$

ρ_o = atmospheric density at sea level and

$$\sigma = \frac{\rho}{\rho_o}$$

Obstacle clearance distance, $S_a = 450$.

From classifications minimum landing runway distance S_{ll} = 300 m = 984 ft

$$984 = 80 \times \left(\frac{W}{S}\right) \times \left(\frac{1}{1x1.28}\right) + 450$$

$$\frac{W}{S} = 8.544 \frac{lb}{ft^2} = 41.71 \frac{K}{m^2}$$

5.6.3 Wing loading for cruise:

Wing loading should be selected to get higher L/D at cruising, which maximizes range during cruising. A propeller aircrafts get maximum range as flying at the speed for best L/D. so to maximize range should fly such that:

Wing loading for maximum range:

$$\frac{W}{S} = q \times \sqrt{\pi \times A \times e \times C_{D_0}} \qquad (5\text{-}6)$$

Oswald efficiency or Drag to left efficiency can be found from equation (5-6-1). (3)

$$e = 1.78(1 - 0.045A0.68) - 0.64 \qquad (5\text{-}6\text{-}1)$$

e = 0.8

Lift drag coefficient for prop, $C_{D_0} = 0.03$

$$q = 0.5 \times \rho \times V^2 = 790\ p$$

$$\frac{W}{S} = 790 \times \sqrt{\pi \times 8 \times 0.8 \times 0.03}$$

$$\frac{W}{S} = 62.5 \frac{K}{m^2}$$

5.6.4 Wing loading for loiter endurance:

For the best performance of the propeller aircraft at loiter, the induced drag is three times the parasite drag. This also provides the wing loading to get minimum power done.

The designer should determine the loitering velocity which is usually ranges between (80-120 knots) or (150 – 220 km/h) and then select the wing loading.

$$\frac{W}{S} = q \times \sqrt{3 \times \pi \times A \times e \times C_{D_0}} \tag{5-7}$$

$$\left(\frac{W}{S}\right)_{l} = \sqrt{3} \times \left(\frac{W}{S}\right)_{c} = 108.25\,\frac{k}{m^2}$$

The result of the equation above for the wing loading is the average during loiter. This must be converted to takeoff conditions by dividing loiter wing loading by the ratio of the average loiter weight to the takeoff weight. This ratio can be assumed to be about 0.85.

5.6.5 Wing loading for Sustain turns:

Sustained turn rate is defined as the maximum load factor at some flight condition that the aircraft can sustain without slowing down or losing altitude.

During a turn, the lift equals the weight times the load factor *n* divided by the dynamic pressure.

$$\frac{W}{S} = \frac{q}{n} \times \sqrt{\pi \times A \times e \times C_{D_0}} \tag{5-8}$$

The equation below gives the wing loading that exactly attains a required sustained load factor *n*

$$\frac{W}{S} = \frac{\frac{T}{W} \pm \sqrt{\left(\frac{T}{W}\right)^2 - \frac{4 \times n^2 \times C_{D_0}}{A \times e \times \pi}}}{\frac{2 \times n^2}{q \times A \times e \times \pi}} \tag{5-9}$$

Piston-prop propulsion use propeller with efficiency(η). Specific thrust for piston-prop vehicles is calculated as:

$$\frac{T}{W} = \frac{\eta \times \frac{V \times T}{\eta}}{V \times W} = \left(\frac{\eta}{V}\right) \times \left(\frac{P}{W}\right)$$

$$\frac{w}{s} = \frac{\frac{\eta}{V} \times \frac{p}{w} \pm \sqrt{\left(\frac{\eta}{V}\right)^2 \times \left(\frac{p}{w}\right)^2 - \frac{4 \times n^2 \times C_{D_0}}{A \times e \times \pi}}}{\frac{2 \times n^2}{q \times A \times e \times \pi}} \tag{5-10}$$

$$\frac{w}{S} = \frac{\frac{0.8}{41.66} \times \frac{p}{w} \pm \sqrt{\left(\frac{0.8}{41.66}\right)^2 \times \left(\frac{p}{w}\right)^2 - \frac{4 \times 4^2 \times 0.03}{8 \times 0.8 \times \pi}}}{\frac{2 \times 4^2}{80.53 \times 8 \times 0.8 \times \pi}}$$

5.6.6 Wing loading for Climb and glide:

Climb is defined as a vertical velocity and the climb gradient *G* is the ratio between the vertical and horizontal distance traveled. Solving for wing loading yields:

$$\frac{w}{s} = \frac{\left(\left(\frac{\eta}{V} \times \frac{p}{w}\right) - G\right) \pm \sqrt{\left(\frac{\eta}{V} \times \frac{p}{w} - G\right)^2 - \frac{4 \times C_{D_0}}{A \times e \times \pi}}}{\frac{2}{q \times A \times e \times \pi}} \tag{5-11}$$

$$\frac{w}{S} = \frac{\left(\left(\frac{0.8}{41.66} \times \frac{p}{w}\right) - 1/12\right) \pm \sqrt{\left(\frac{0.8}{41.66} \times \frac{p}{w} - 1/12\right)^2 - \frac{4 \times 0.03}{8 \times 0.8 \times \pi}}}{\frac{2}{80.53 \times 8 \times 0.8 \times \pi}}$$

5.6.7 Wing loading for Climb limit:

The power-to-weight needed for climb is given by equation:

$$\frac{P}{W} = g \times V \times \left(G + 2 \times \sqrt{\frac{C_D}{\pi \times A \times e}}\right) \tag{5-12}$$

$$\frac{P}{W} = 9.81 \times 41.66 \times \left(\frac{1}{12} + 2 \times \sqrt{\frac{0.03}{\pi \times 8 \times 0.8}}\right)$$

= 65.6 w/kg

5.6.8 Wing loading and Takeoff distance:

Takeoff distance for ground roll is definsed as the travelled distance from the point where the aircraft starts moving till the point where wheels leave the ground. The liftoff speed for a normal takeoff is for the safety reasons 1.1 times stall speed. (3). Figure (5-8).

There is another critiria for take off distance which is obstacle clearance distance. It is defined as the distance required from the point where the aircraft starts moving till the aircraft has reached some specified altitude. This is usually 15 m.

The takeoff distance is affected by wing loading and power to weight ratio. The equation below can be used to determine the required wing loading to attain some required takeoff distance.

$$\frac{W}{S} = (T \quad) \times \sigma \times C_{L_T} \times \frac{hp}{W} \tag{5-13}$$

C_{L_T} - lift coefficient at takeoff selected here to be = 1.2, $\sigma = 1$ is density height ratio.

TOP – takeoff parameter. This can be choosen from the figure (5-9) below according to takeoff distance.

Takeoff distance = Ground takeoff length × 1.66 (3)

Takeoff distance = 984 × 1.66 = 1633 ft

$$\text{TOP} = 165 \, \frac{\text{ll}}{\text{ft .h}}$$

By substitution in equation (5-13), we will find different values of $\frac{p}{W}$ for different values of wing loading $\frac{W}{S}$.

$$\frac{W}{S} = 165 \times 1 \times 1.2 \times \frac{hp}{W}$$

Figure (5-7) Illustrates the relationship between Wing loading and takeoff weight.

For example we have wing area $S_w = 5.27\ m^2$; Table (5-4):

Table (5-4) Power loading values for different values of wing loading for takeoff distance

W/s(kg/m^2)	W/s (lb/ft^2)	W (lb)	p/w (hp/lb)	P (hp)	P (kw)	P/W (w/kg)
20	4.096	232.36	0.0207	4.81	3.588	34.317
40	8.19	464.72	0.041	19	14.2	67.9
45	9.216	522.82	0.0465	24.33	18.15	77.145
50	10.24	580.9	0.0517	30	22.38	85.14
60	12.288	697.1	0.062	43.22	32.24	101.96
70	14.337	813.27	0.0724	58.88	43.9	119
100	20.49	1161.8	0.1035	120.23	89.7	170.2

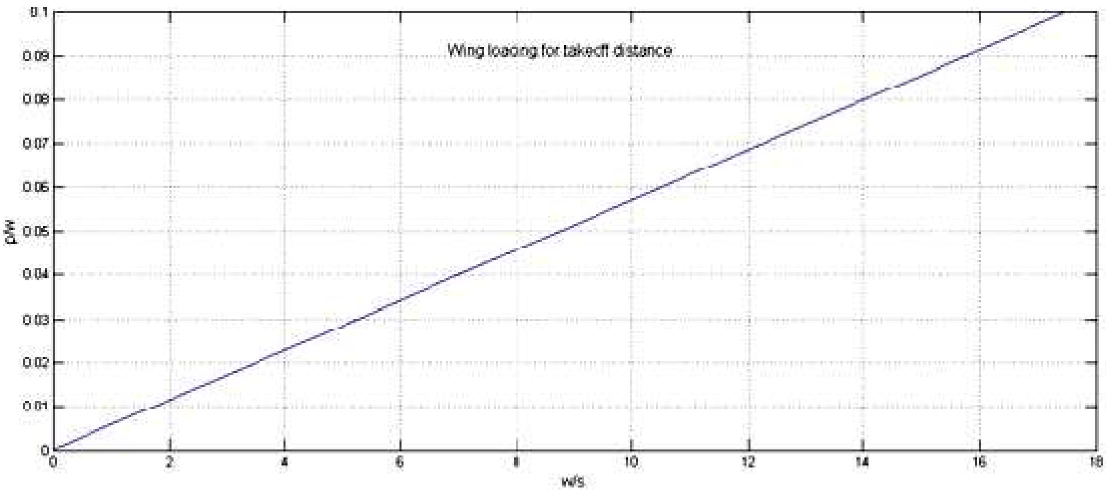


Figure (5-7) Wing loading and takeoff weight relationship

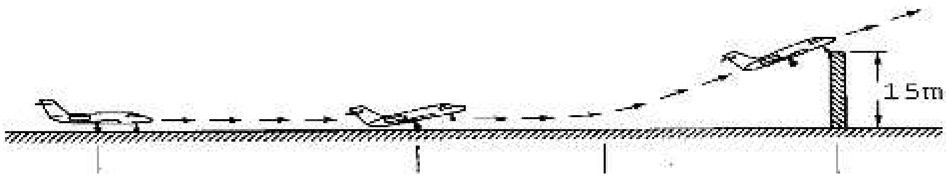


Figure (5-8)The definition of take-off distance.

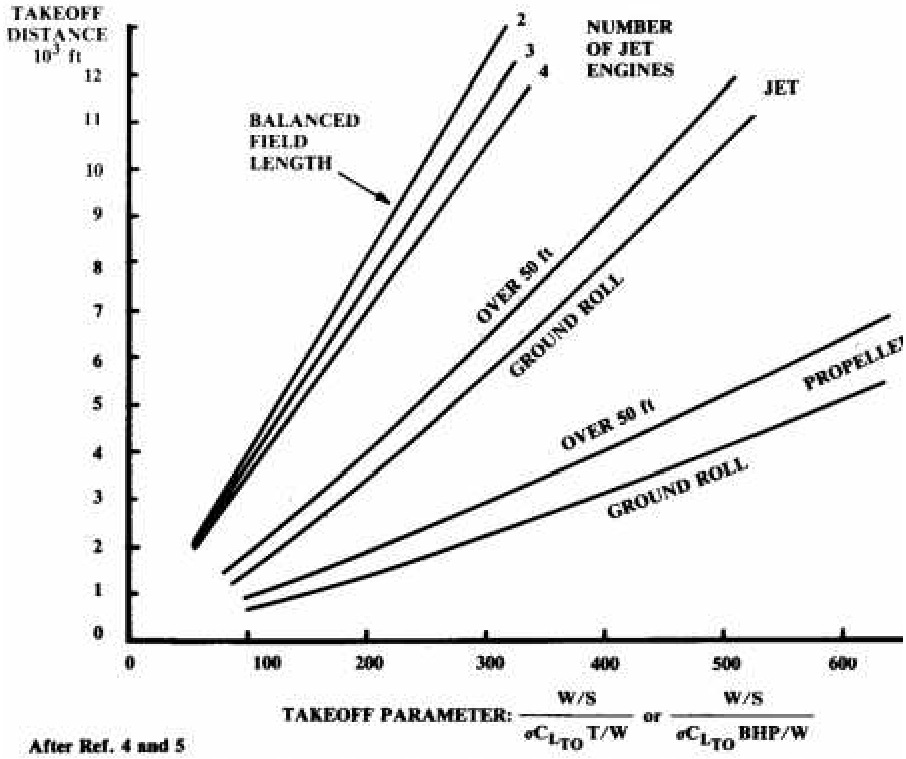


Figure (5-9) The relationship between takeoff parameter and takeoff distance

5.6.9 Wing loading for Cuise speed:

The wing loading for cruising speed is given by equation:

$$\frac{W}{S} = \frac{\sigma}{0.65 \times Ah \times \frac{p}{w}} \times \left(\frac{V_C}{3.16 \times k}\right)^3 \qquad (5\text{-}14)$$

K= 1:00 biplanes with fixed landing

=1:15 monoplane without strutsfixed landing gears

=1:30 monoplanes withretractable landing gears

Ah = $\frac{\rho}{\rho_o}$ without turbocharger

= 1 with turbocharger

5.6.10 Wing loading for Maximum speed:

The wing loading for maximum speed is given by equation:

$$\frac{W}{S} = \frac{\sigma}{Ah \times \frac{P}{W}} \times \left(\frac{V_m}{3.16 \times k}\right)^3 \qquad (5\text{-}15)$$

5.6.11 Wing loading for Maximum ceiling:

The wing loading for maximum ceiling is given by equation:

$$\frac{W}{S} = q\sqrt{A \times e \times \pi \times C_{D_0}} \qquad (5\text{-}16)$$

$$= 790 \times \sqrt{8 \times 0.8 \times \pi \times 0.03} = 62.54 \text{ kg/m}^2$$

5.6.12 Matching for Sizing Requirements:

It is now possible to determine the combination of quantities which define the design point; base on the methods above. This process is also known as the matching process. Figure (5-10).

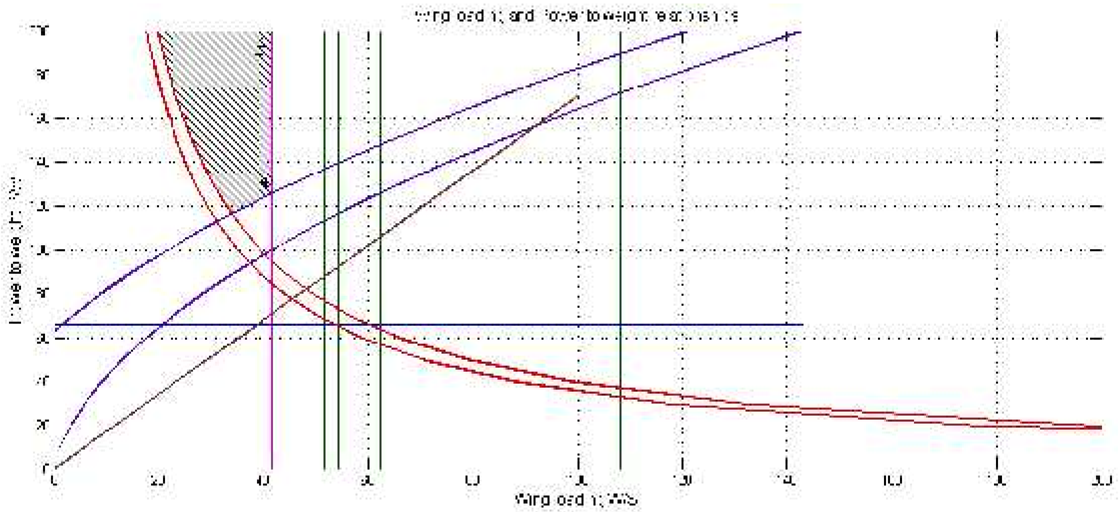


Figure (5-10) Wing loading and power loading relationship for aircraft design parameters

5.7 UAV sizing:

5.7.1 Wing sizing:

Unmanned aircraft conceptual design is often only concerned with the structural weight rather than details of the structure itself. Initial weight allocations are established from the conceptual unmanned aircraft sizing analysis, and then detailed structural analyses are performed in preliminary and detail design phases. (2)

P/W = 130, p = 28.6 kw.

W/S = 41.715, S = 5.27 m^2

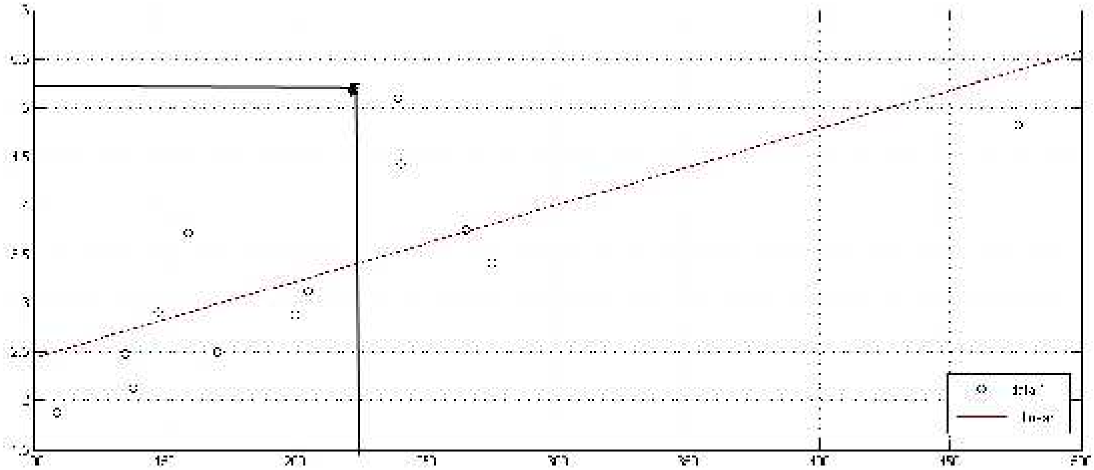


Figure (5-11) Wing area and Takeoff weight relationship

Here we got high wing area because we selected small landing speed much smaller than it is selected for existing UAVs.

AR = b^2/S,

Wing span b = 6.5 m,

Wing elevator length = b/4 = 1.625 m for each side of the wing.

Wing cord C = 0.81 m

Wing span & wing area versus takeoff weight are shown on Figures (5-11) & (5-12).

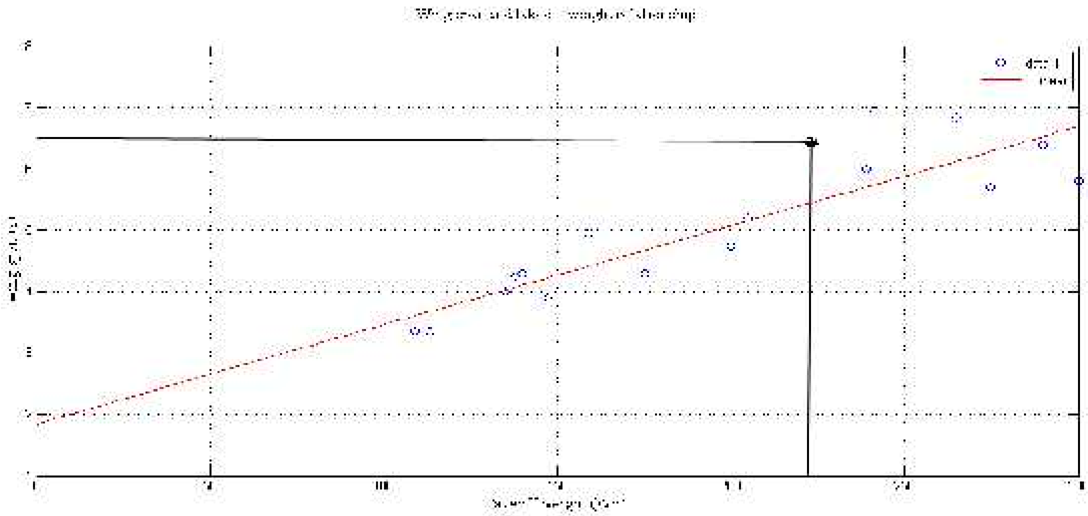


Figure (5-12) Wing span and Takeoff weight relationship

5.7.2 Empennage Sizing:

Volumetric coefficient of horizontal and vertical stabilizer:

$$V_H = \frac{S_H \times L_H}{S \times C} \tag{5-17}$$

$$V_V = \frac{S_V \times L_V}{S \times b} \tag{5-18}$$

Where, S_H and S_V surface areas of horizontal and vertical stabilizer, S { wing reference area, L_H and L_V horizontal and vertical stabilizer arm (distance between center of gravity and aerodynamic center) Figures (5-13) and (5-14), b is the wing span, and C is mean aerodynamic chord, $V_{H,}$ Tail volume coefficient, V_V Vertical Tail volume coefficient.

5.7.2.1 Horizontal Tail Sizing:

Typically the horizontal tail surface area ranges from one fifth to one fourth of the wing surface area. The horizontal tail can have a sweep and a dihedral. Sweeping of the horizontal tail would effectively increase the tail arm L_{HT}, which is an important consideration when sizing the horizontal tail. For a T-tail configuration, the tail arm further increases. (5)

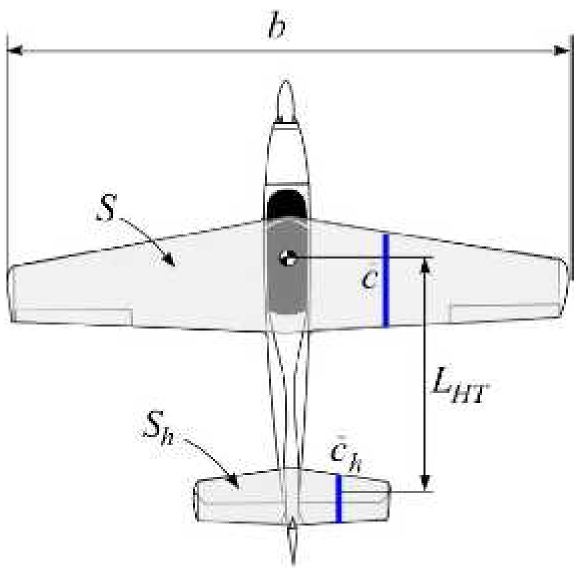


Figure (5-13) Horizontal stabilizer arm

We calculate the wing area before (S_w) = 5.27 m^2

$$\frac{S_H}{S} = 0.213\ (\pm\ 0.031)$$

We will consider $\frac{S_H}{S} = 0.23$

$$S_H = 5.27 \times 0.23 = 1.21\ m^2$$

Aspect ratio of the horizontal tail $AR_h = \frac{2}{3} AR_{W.}$ (6)

We will consider Aspect ratio AR = 5

Horizontal tail Span

$$b_H^2 = 5 \times 1.21 = 6.05\ m^2$$

$$b_H = 2.46\ m$$

Horizontal tail cord

$$C_H = \frac{S_H}{b_H} = 0.49\ m$$

Root chord = tip chord=mean aerodynamic chord = 0.49 m

$$\frac{L_H}{C} = 3.09\ (\pm 0.565)$$

From equation (5-17)

$$V_H = \frac{S_H}{S} \times \frac{L_H}{C}$$

$$V_H = 0.23\ \times 3.32$$

$$V_H = 0.763$$

$V_H = 0.658\ (\pm\ 0.135)$, $V_H = 0.52: 0.79$

$$L_H\ = 0.763 \times \frac{1}{0.2} \times 0.81 = 3.09\ m$$

Taper ratio λ = 1

Sweep angle $\Lambda = 0^{o}$

$$\frac{t}{c} = 0.12$$

5.7.2.2 Vertical Tail Sizing:

The vertical tail surface area is about 12 to 20% of the wing reference area. The vertical tail design is critical to takeoff, especially in tackling yawed ground speed resulting from a crosswind and/or asymmetric power of a huge aircraft. A large vertical tail can cause snaking of the flight path at low speed, which can be resolved easily by introducing a “yaw-damper” (a matter of aircraft control analysis). At cruise, a relatively large vertical tail is not a major concern. Sweeping of the vertical tail would effectively increase the tail arm L_V, an important dimension in sizing the vertical tail. (5)

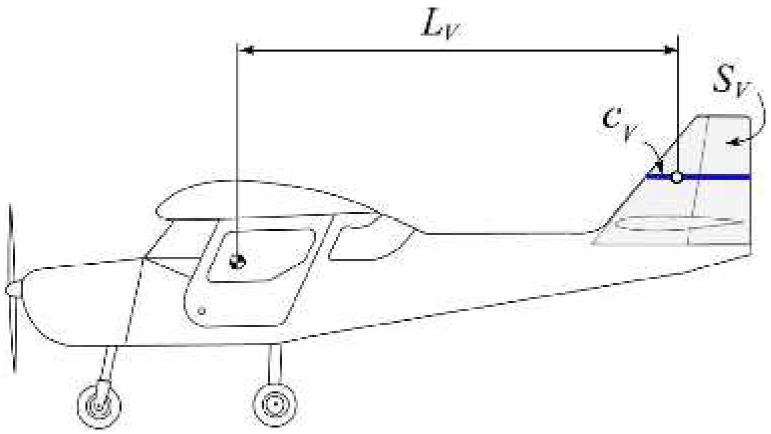


Figure (5-14) vertical stabilizer arm

$$\frac{S_V}{S} = 0.107\ (\pm\ 0.034)$$

We will consider $\frac{S_V}{S}$= 0.1

The vertical tail area $S_V = 0.53\ m^2$

$$\frac{L_V}{b} = 0.418\ (\pm 0.081)$$

From equation (5-18)

$$V_V = \frac{S_V}{S} \times \frac{L_V}{b} = 0.1 \times 0.4$$

$$V_V = 0.04$$

$$L_V = 0.46 \text{ x } 6.5 = 2.99 \text{ m}$$

Assume Aspect ratio of vertical tail $\quad AR_v = 1.4$

Vertical tail span $b_v = \sqrt{(S \quad \times A \quad)} =$

$$b_v = 0.86 \text{ m}$$

Sweep angle $\beta = 20^{o}$

Vertical tail Mean aerodynamic cord = 0.615 m

Tip cord = 0.55 m

Root cord = 0.68

Taper ratio λ = 0.8

5.7.3 Landing Gear Sizing

The main landing gear height is determined by the clearance between the fuselage, wing tips, propeller blade tips, vertical tail and ground during landing and take-off. These components must not be in contact with the runway during the landing and takeoff processes.

5.7.4 Fuselage Sizing:

The fuselage must perform many functions and house many systems. Yet all the fuselage can do is decrease from the performance due to weight and drag effects. Therefore, the negative impacts must be minimized by making the fuselage compact and given low-drag shaping.

Components installed within the fuselage can include avionics, subsystems, propulsion system, landing gear, payloads, and in some cases fuel. The placement of these components must support aircraft balance.

Payload bays that can accommodate variable weights and fuel tanks for longitudinal locations near the center of gravity during conceptual design because the center-of-gravity travel must be kept within a narrow range. (2)

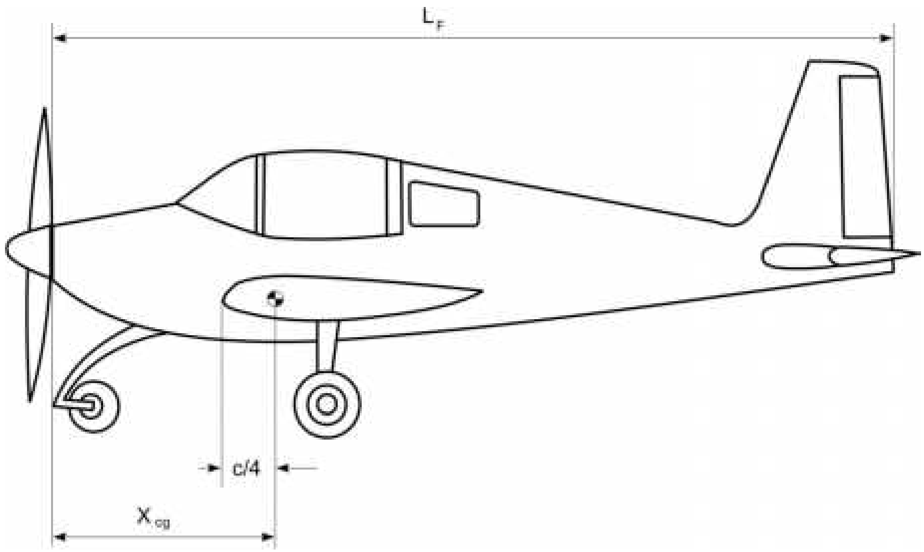


Figure (5-15) Aircraft sketch illustrates the fuselage length and the center of gravity

Fuselage length can be calculated from equation (5-19):

$$L_F = 0.538 \times b + 1.66 \tag{5-19}$$

$$L_F = 5.16 \text{ m}$$

Where, b represents the wing span. The relative distance between airplane nose (without propeller) to airplane's center of gravity $Xcg_1 = Xcg \times L_F$

The fuselage shape supposed to be cylindrical, and has a diameter $D_f = 0.4$ m.

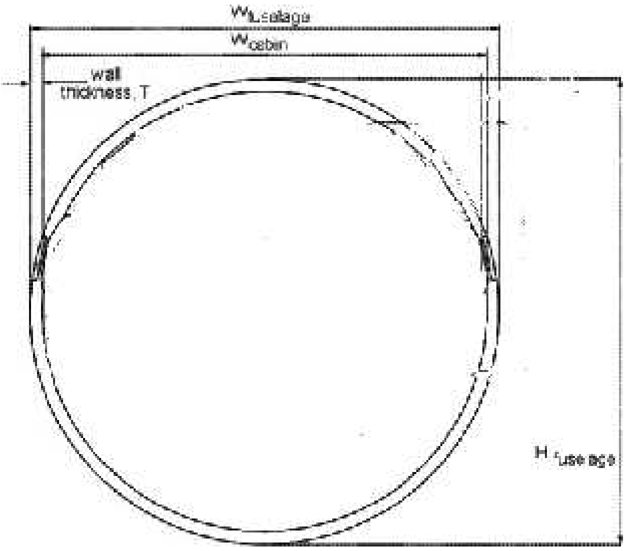


Figure (5-16) Cross section area of the UAV fuselage

5.8 Center of gravity:

To calculate the aircraft center of gravity, the center of gravity of each component is assumed to be its geometrical center. It is also assumed that the geometry of components can be adequately represented with rectangular blocks and cylinders. The span wise distribution of weight is assumed to be even on both sides of the fuselage so that the analysis of center of gravity in z direction is unnecessary. The distance in the (y) direction from the axis of rotation of the propeller to the (y) component of center of gravity is assumed to be sufficiently small that the moment created by the engine thrust about the center of gravity can be neglected.

Center of gravity position with respect to fuselage nose section:

$$\mathrm{Xcg} = X\,cg_1 \times \mathrm{L_F} \tag{5-20}$$

$$\mathrm{Xcg_1} = 0.31(\pm\ 0.022)$$

$$\mathrm{Xcg} = 0.33 \times 5.16 = 1.7 \text{ m}$$

6. Wing weight estimation

6.1 Structure components weight estimation:

6.1.1 Composite materials and their effect on the aircrafts components weight:

In the initial stages of aircraft design, the combination of engines, materials, and aerodynamic technology enabled aircraft speeds of slightly more than 300 Km/h; altitude was limited by human physiology. The prevalent material used for the aircraft manufacture was wood. First aircraft with higher strength-to-weight ratios of isotropic material properties was from duralumin, earliest types of age- hardened aluminum alloys, which is introduced In the 1930s by Durener Metallwerke of Germany with higher, and considerable growth in speed and altitude resulted. The development of aircraft manufacture materials to include metals opens new horizons of the manufacturing technology. Structure, aerodynamics, and engine development paved the way for substantial gains in speed, altitude, and maneuvering capabilities. (5)

In 1970s, aircraft industry has evolved considerably by the use of composites materials. Traditional materials for aircraft construction include aluminum, steel and titanium. The first advantage introduced by composites, that composite materials can offer reduced weight and assembly simplification.(23) Cast aluminum alloys are used extensively for making various engineering products due to their light weight and high strength to weight ratio. (27) The performance advantages associated with reducing the weight of aircraft structural elements. Composites are also being used increasingly as replacements for metal parts due to their advanced properties. (23) Al-Fe-Si-V alloys have attracted considerable interest since they maintain strength at temperatures higher than other Al-Fe-X system alloys. Excellent high temperature mechanical properties of Al-Fe-Si-V alloys are attributed to the high volume fractions of ultrafine cubic silicon intermetallic phase, $Al_{12}(Fe,V)_3Si$, and its slow coarsening rate. As the percentage of ceramic increases, composite displays more and more properties of ceramic, like improved stiffness and decreased ductility.(27) Composite materials have major effect on the aircraft industry such as, durability, corrosion resistance, and fatigue resistance especially for aluminum alloys, thin material with high integrity, highly complex loading, and additionally higher strength-to-weight ratios. (23)

Appreciable improvement in hardness levels was achieved by age hardening in modified and reinforced matrix of Al-8Fe-8Si-1.4V/SiC_p composites. This improvement increased with addition of Mg and also with increased SiC content in the matrix. Bending strength improved significantly by applying heat treatment to the modified and reinforced matrix. Modulus of elasticity increases 9.5% when 4% SiC_p added to the Al-8Fe-8Si-1.4V matrix. (28)

The factors that affect the aircraft weight are:

- Weight depends on the choice of material and its properties such as, Material elasticity, density, and strength-to-weight ratio and so on.
- Weight is directly affected by size, indicated by geometry (i.e., length, area, and volume).
- Weight depends on internal structural-member density – that is, the denser, the heavier.
- Weight depends on a specified limit-load factor n for structural integrity.
- Fuselage weight related to pressurization, engine and undercarriage mounts, doors, and so forth.
- Lifting-surface weight associated with the loading, fuel amount, engine weight and undercarriage mounts, and so on. (5)

6.2 Structure weight estimation methods:

Most of the methods and the equations followed in this chapter to determine the structure components weight estimation represent the general aviation airplanes class or unmanned aerial vehicles class if possible.

The aircrafts which are classified as a general aviation aircrafts are: (4)

- Home built airplanes.
- Single engine propellers.
- Twin engine propellers.
- Agricultural airplanes.
- Regional turbopropellers weight less than 12500 lbs (5600 kg).
- Low speed military trainers.
- Small and low speed flying boats.

6.3 Approximate method of structure group weight:

Each part of structure group has a parameter which is more effect than others in terms of weight estimation. The wing and empennage weights are specified for the exposed planform area values. The fuselage weight is determined for its skin area, or as called wetted area. The landing gear weight estimation is directly concerned with the takeoff gross weight, and it is estimated as a fraction of it. The installed engine weight is a multiple of the uninstalled engine weight. The residual items of the empty weight are determined as a fraction of takeoff gross weight. (3)

6.4 Wing weight estimation methods:

The first sizing of the wing was done using the results of the parametric analysis and its design was a highly iterative process. The main design driver was to minimize the weight consideration, which led to some compromises.

The first step was to determine the shape of the plan form, tapered or un-tapered, it appeared that a tapered wing was lighter than an un-tapered wing having the same features (area, aspect ratio and thickness). The area and the aspect ratio are directly calculated based on the performance requirements and the parametric analysis.

Models of conventional tactical UAVs weights between 100 to 500 kg are chosen for wing weight estimation. The parameters values of these UAVs are input to Matlab program to get the result on chart; the results are evaluated and compared with the results of calculated equations for UAV weights 250 kg as takeoff weight.

Wing weight represents about 17-27% of the empty weight. (24)

6.4.1 Raymer method:

General aviation aircrafts equation for wing weight estimation from Raymer:

$$W_w = 0.036 \times S_w^{0.7} \times W_f^{0.0} \times \left(\frac{A}{\cos^2\Lambda}\right)^{0.6} \times q^{0.0} \times \lambda^{0.0} \times \left(\frac{100 \times t/c}{\cos\Lambda}\right)^{-0.3}$$
$$\times \left(N \times w_d\right)^{0.4}$$

W_w – wing weight

S_w – exposed wing area

W_{fw} – fuel weight

A – aspect ratio

q – dynamic pressure

Λ – sweep angle at 25% MAC

λ – taper ratio

t/c – wing thickness cord ratio

N – ultimate load factor

W_{dg} – design gross weight

All of the dimension parameters of the wing geometry are presented in Raymer wing weight fraction (Ww/Wo) equation, beside some other parameters such as; design gross weight, ultimate load factor, fuel weight, and dynamic pressure. Raymer equation is mostly affected by values of takeoff weight and planform wing area of the aircraft.

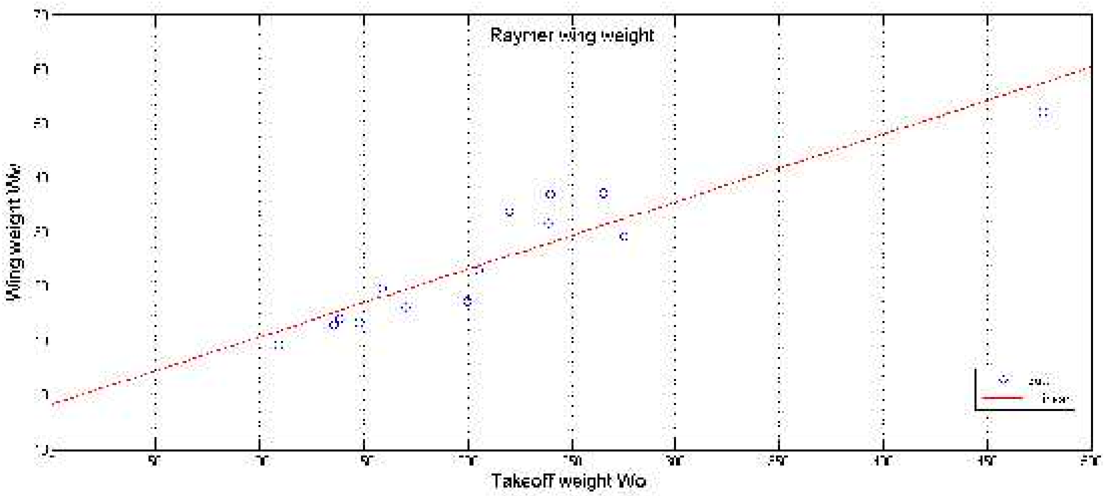


Figure (6-1) Raymer wing weight estimation

From figure (6-1), The results of wing weight fraction (Ww/Wo) of Raymer equation according to the fitting line range between 10% at 100 kg and 12.2 % at 500 kg takeoff weight. For 220 kg takeoff weight, wing weight is equal to 33.85 kg which represents 15.38 %

$$0.154 \times 220 = 33.85 \text{ kg}$$

By substitution in Raymer equation for parameters values that we got for aircraft which has takeoff value 220 kg, the result is as follow:

$$W_w = 0.036 \times 56.7^{0.7} \times 170^{0.0} \times \left(\frac{8}{c \quad ^2 0}\right)^{0.6} \times 16.4^{0.0} \times 1^{0.0} \times \left(\frac{1 \quad \times 0.1}{c \quad 0}\right)^{-0.3} \times (9 \times 485)^{0.4} =$$

$$W_w = 74.6\ ll = 33.85\ k$$

Comparing the outcome of the equation and the value of wing weight that we got from the figure, it is clear that there is approximately same.

6.4.2 Usaf method:

Usaf general aviation aircrafts equation for wing weight estimation:

$$W_w = 96.948 \times \left(\left(\frac{W_o \times N_z}{10^5} \right)^{0.6} \times \left(\frac{A}{\cos \Lambda} \right)^{0.5} \times \frac{S_w}{100}^{0.6} \times \left(\frac{1+\lambda}{2 \times t/c} \right)^{0.3} \times \left(1 + \frac{V}{500}\right)^{0.5} \right)^{0.9}$$

Wo – takeoff weight

N_z – ultimate load factor

A – aspect ratio

λ – taper ratio

S_w – Wing reference area

t/c – thickness to wing ratio

V – maximum level speed at sea level

In Usaf equation we can see that the parameters of the equation are moderately used, so that the equation outcome is not affected dramatically by changing any parameter individually. The equation basically depends upon the fixed number (96.948), we can see that clearly in figure (6-2), that most of results of the equation lay very close to the fitting line.

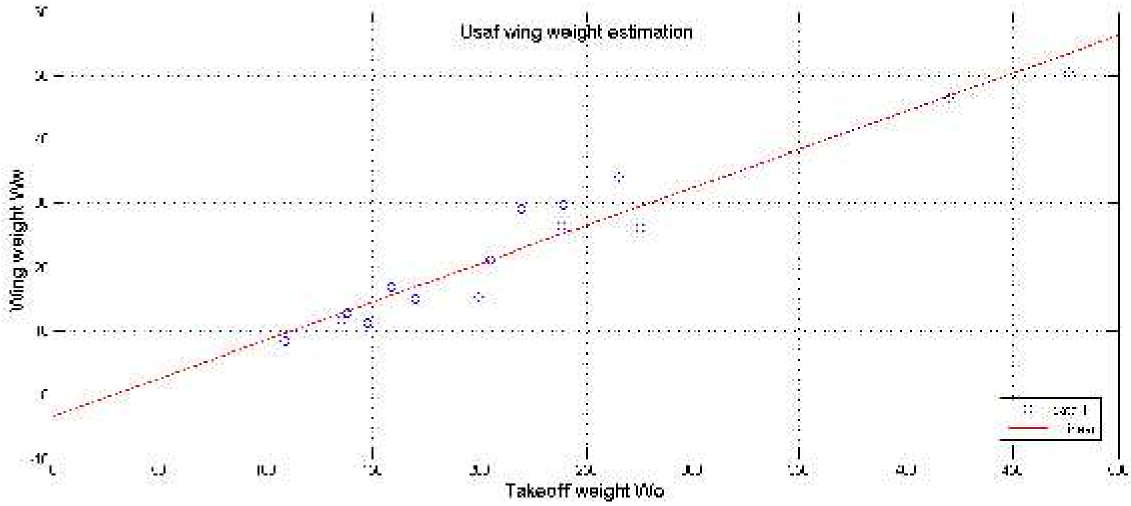


Figure (6-2) Usaf wing weight estimation

From Figure (6-2), the results of wing weight fraction (Ww/Wo) of Usaf equation according to the fitting line, range between 8% at 100 kg and 11.2 % at 500 kg takeoff weight. For 220 kg takeoff weight, wing weight is equal to 28.85 kg which represents 13.11 %

$$0.131 \times 220 = 28.85 \text{ kg}$$

By substitution in Usaf equation for values that we got for aircraft which has takeoff value 220 kg, the result is as follow:

$$W = 96.948 \times \left(\left(\frac{485 \times 9}{10^5}\right)^{0.6} \times \left(\frac{8}{c\ \ 0}\right)^{0.5} \times \frac{56.7^{0.6}}{100} \times \left(\frac{1+1}{2 \times 0.15}\right)^{0.3} \times \left(1 + \frac{108}{500}\right)^{0.5}\right)^{0.9} =$$

$$W = 63.6\ ll = 28.85\ k$$

6.4.3 Kroo method:

Kroo equation for wing weight estimation:

$$W = 4.22 \times S_w + 1.642 \times 10^{-6} \left(\frac{N_u \times b^3 \times (T(\quad \times Z \quad)^{0.5} \times (1 + 2 \times \lambda)}{(t/c) \times c\ \ ^2\Lambda \times S_W \times (1 + \lambda)}\right)$$

W_w – wing weight

S_{wg} – gross wing area

TOW- takeoff weight

ZFW – fuel weight

Λ_e – sweep of the structural axis

λ – taper ratio

t/c – average airfoil thickness

N_{ul} – ultimate load factor

b – wing span

The Wing weight estimation equation established by Kroo is fundamentally formed for big size aircrafts. This equation is dramatically controlled by gross wing area (S_w) parameter, beside some other parameters which has lesser extent effect such as ultimate load factor, wing span, takeoff weight, fuel weight, average airfoil thickness, taper ratio, and sweep of the structure axis.

By substitution in Kroo equation for parameters values that we got for aircraft weighs 220 kg as takeoff weight, the result is as follow:

$$W = 4.22 \times 56.7 + 1.642 \times 10^{-6} \left(\frac{9 \times 21.3^{3} \times (485 \times 170)^{0.5} \times (1 + 2 \times 1)}{(0.15) \times c\ {}^{2}0 \times 56.7 \times (1 + 1)} \right)$$

$$W = 246.5\ li = 111.91\ k$$

For 220 kg takeoff weight, wing weight is equal to 111.91 kg which represents 50.85 % of W_o.

The results of wing weight fraction (Ww/Wo) of the Kroo equation according to the fitting line, Figure (6-3), range between 52 % at 100 kg and 24 % at 500 kg takeoff weight.

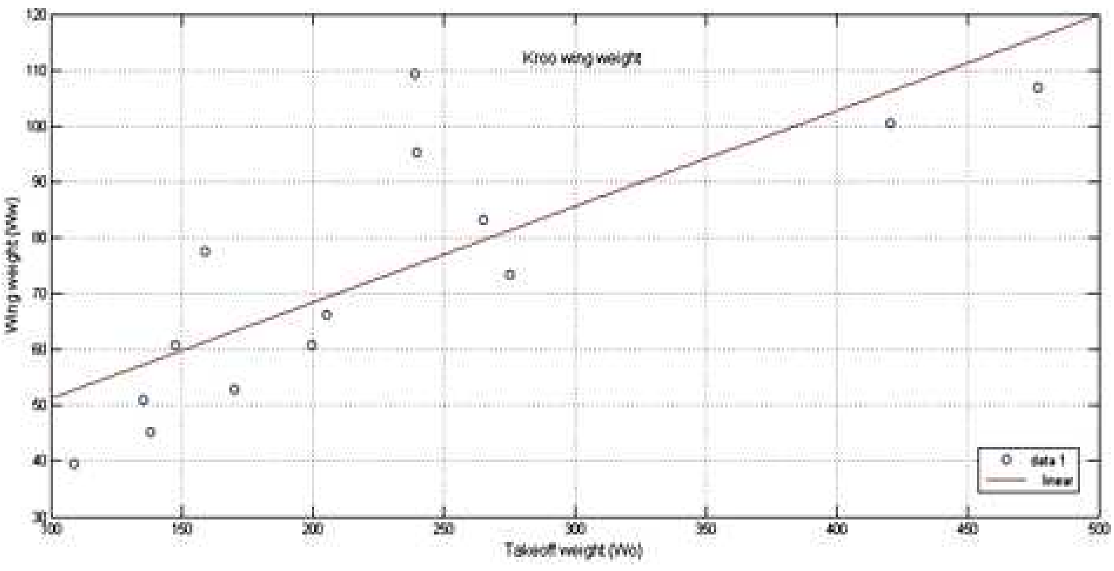


Figure (6-3) Kroo wing weight estimation

6.4.4 Torenbeek method:

This equation applies to light transport aircrafts with takeoff weights less than 5600 kg.

$$W = 0.00125 \times W_0 \times \left(\frac{b}{\cos \Lambda}\right)^{0.7} \times \left(1 + 6.3 \times \frac{\cos}{b}\right)^{0.5} \times N_z^{0.5} \times \left(\frac{b \times S_w}{t \times W_0 \times \cos}\right)^{0.3}$$

From Figure (6-4), fitting line indicates that the wing weight fraction for 100 kg is 8.5% and increased to reach 10.3% at 500 kg. For our aircraft which weighs 220 kg takeoff weight the wing weight fraction is 13.24 %.

$$0.1324 \times 220 = 29.13 \text{ kg}$$

By substitution in Torenbeek equation for parameters values that we got for aircraft which weighs 220 kg, the result is as follow:

$$W = 0.00125 \times W_0 \times \left(\frac{21.3}{\cos 0}\right)^{0.7} \times \left(1 + 6.3 \times \frac{\cos 0}{21.3}\right)^{0.5} \times 9^{0.5} \times \left(\frac{21.3 \times 56.7}{0.4 \times 485 \times \cos 0}\right)^{0.3} =$$

$$W = 64.17\ lb = 29.13\ kg$$

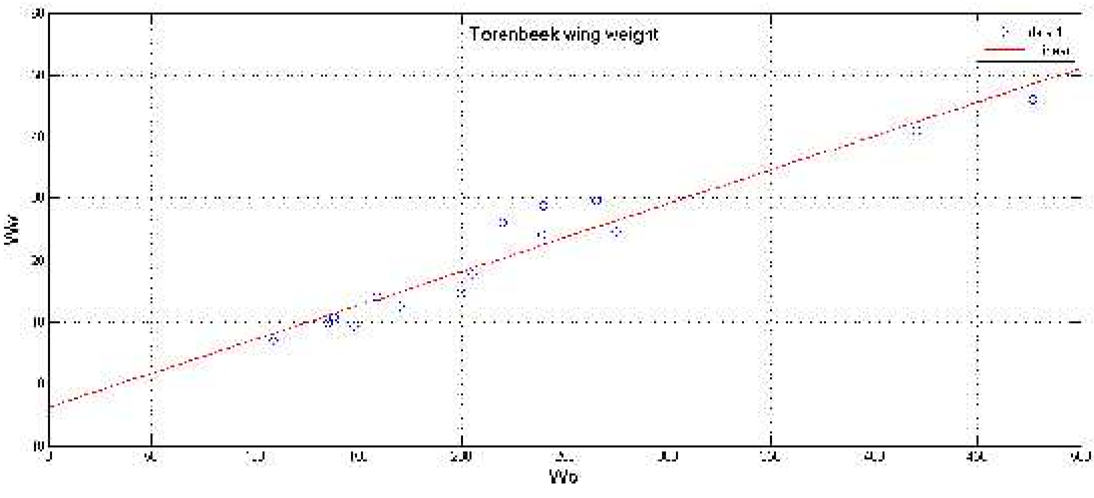


Figure (6-4) Torenbeek wing weight estimation

6.4.5 Jay Gundlach method:

The following wing weight equation is developed by Gerard for manned sailplanes: (25).

$$W = 0.0038 \times (N_z \times W_0)^{1.0} \times A^{0.3} \times S_w^{0.2} \times (1 + \lambda)^{0.2} \times (t/c)^{0.1}$$

N_Z - the ultimate load factor

W_o – takeoff weight

A – aspect ratio

S_w - wing planform area

λ – taper ratio

t/c - thickness-to-chord ratio

From figure (6-5), the results of wing weight fraction (Ww/Wo) of Gerard equation according to the fitting line range between 8.5% at 100 kg and 17.6 % at 500 kg takeoff weight. For our aircraft which weighs 220 kg, wing weight is equal to 35.14 kg which represents 15.97 %.

$$0.1597 \times 220 = 35.14 \text{ kg}$$

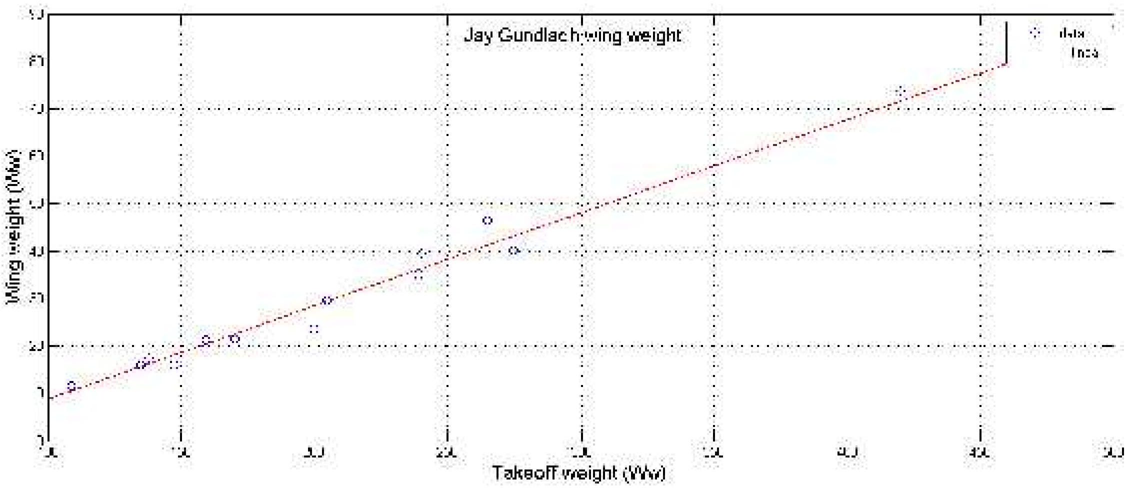


Figure (6-5) Jay Gunlach wing weight estimation

By substation in Gerard equation above for Wo = 220 kg and the other parameters we got before for wing area, aspect ratio, thickness to cord ratio, and taper ratio.

$$W = 0.0038 \times (9 \times 220)^{1.0} \times 8^{0.3} \times 5.27^{0.2} \times (1+1)^{0.2} \times (0.15)^{0.1}$$

$$= 35.135 \text{ kg}$$

6.4.6 Kundu method:

Kundu equation is derived and modified from general equation published by SAWE: (5).

$$M_W = K\ (M_{dg}N_Z)^{x1}\ SW^{x2}\ AR^{x3}\,(t/c)^{x4}\ (1+\lambda)^{x5}\ (\cos\Lambda_{1/4})^{x6}\ (B/C)^{\frac{x7}{t}}\ S_{CS}^{x8}$$

Kundu equation:

$$W \quad = 0.0215 \times (W_o \times N_z)^{0.4} \times S_W^{0.7} \times A \times (1+\lambda)^{0.4} \times \frac{\left(1-\frac{W_f}{W_o}\right)^{0.4}}{\cos\ \Lambda \times (t/c)^{0.4}}$$

The results of wing weight fraction (Ww/Wo) of Kundu equation according to the fitting line range between 18% at 100 kg and 22 % at 500 kg takeoff weight.

By substation in Kundu equation for takeoff weight and the other parameters we got before for our aircraft such as, wing area, aspect ratio, fuel fraction, thickness to cord ratio, taper ratio, and sweep angle, the result is as follow:

$$W \quad = 0.0215 \times (220 \times 9)^{0.4} \times 5.27^{0.7} \times 8 \times (1+1)^{0.4} \times \frac{(1-0.23)^{0.4}}{\cos\ 0 \times (0.15)^{0.4}}$$

$$= 60.9 \text{ kg}$$

Wing weight from Kundu is equal to 60.9 kg which represents 27.7 % of takeoff weight.

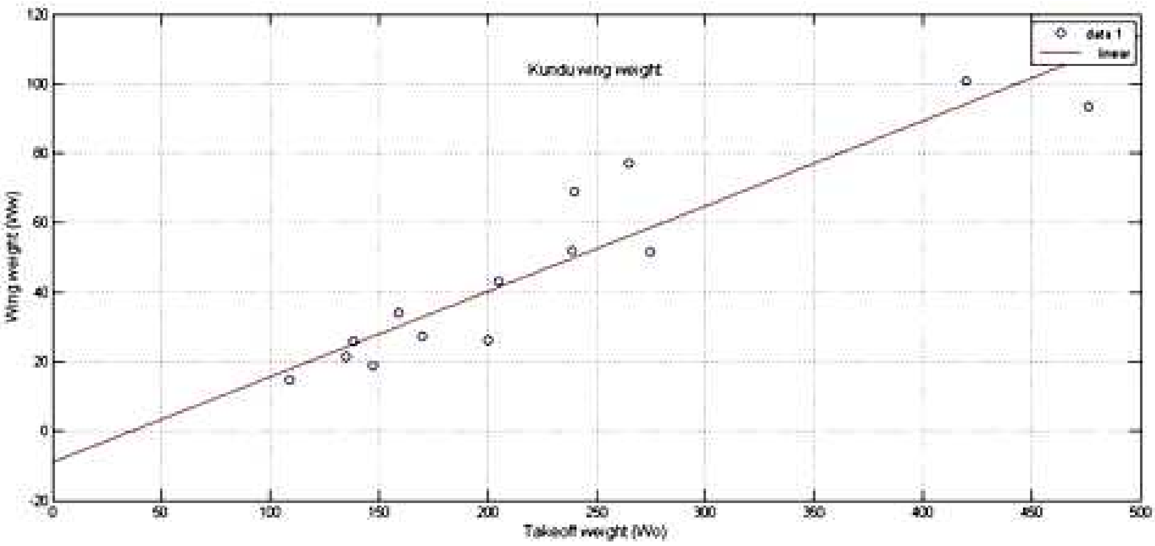


Figure (6-6) Kundu wing weight estimation

7. Empennage weight estimation:

Models of conventional tactical UAVs weights between 100 to 450 kgs are chosen for empennage weight estimation. The parameters values of these UAVs are input into Matlab program to get the results on charts, the results are evaluated for UAV weights 220 kg as takeoff weight to find out the suitable empennage weight.

7.1 Cessna Method:

The equations of Cessna for horizontal and vertical tails should be applied to small size and low performance aircrafts which has maximum speed less than 200 knots.

7.1.1 Horizontal tail:

Cessna introduced the following equation for general aviation aircrafts for horizontal tail weight estimation:

$$W_h = \frac{3.184 \times W^{0.8} \times Sh^{0.1} \times A_h^{0.1}}{57.5 \times t_r^{0.2}} \tag{7-1}$$

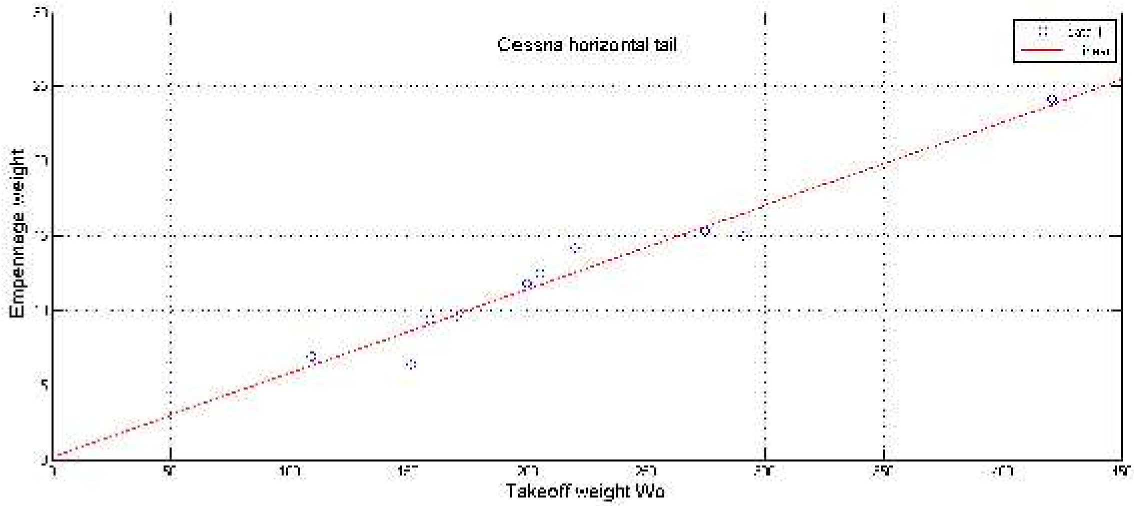


Figure (7-1) Cessna horizontal tail weight estimation

From figure (7-1) for horizontal tail, the fitting line indicates that the result is same for UAVs having takeoff weight ranges between 100 and 450 kgs. The result is about 5.6%, and it seems to be unsatisfied.

By substitution in equation (7-1) for H-tail weight estimation we get:

$$W_h = \frac{3.184 \times 485^{0.8} \times 13^{0.1} \times 5^{0.1}}{57.5 \times 0.19^{0.2}}$$

$$W_h = 31.24\ lb = 14.17\ kg$$

For UAV having 220 kg takeoff weight, horizontal tail is 14.17 kg.

7.1.2 Vertical tail:

The equation from Cessna for vertical tail weight estimation is as follow: (4)

$$W_v = \frac{1.68 \times W_0^{0.5} \times S_v^{1.2} \times A_v^{0.4}}{15.6 \times t_r^{0.7} \times \left(\cos \Lambda_{1/4}\right)^{0.8}} \qquad (7\text{-}2)$$

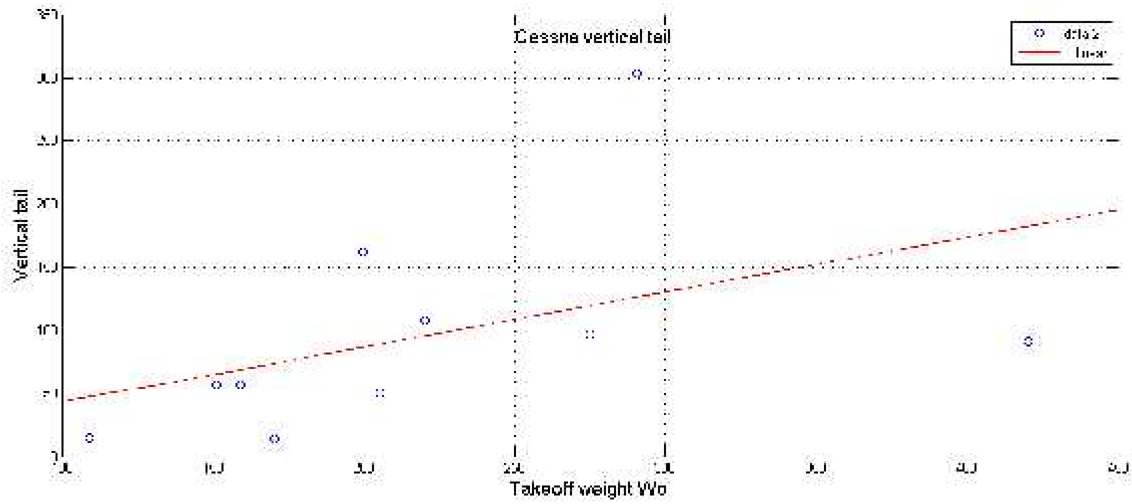


Figure (7-2) Cessna vertical tail weight estimation

By substitution in equation (7-2) for V-tail weight estimation we get:.

$$W_v = \frac{1.68 \times 485^{0.5} \times 5.7^{1.2} \times 1.4^{0.4}}{15.6 \times 0.24^{0.7} \times (\cos 20)^{0.8}}$$

$$W_v = 238.5\ lb = 108.2\ kg$$

The result for V-tail weight estimation as shown in Figure (7-2) and the outcome of the equation is unexpected and unacceptable.

From figure (7-2) for vertical tail, the fitting line indicates that the result is same for UAVs having takeoff weight ranges between 100 and 450 kgs. The result is about 43%.

For UAV having 220 kgs takeoff weight, vertical tail is:

$$0.43 \times 220 = 108.24 \text{ kg}$$

Empennage weight estimation for UAVs having takeoff weight ranges between 100 and 450 kg is same, and it is about 49%. Figure (7-3)

For 220 kg takeoff weight, the empennage estimated weight is:

$$0.556 \times 220 = 122.4 \text{ kg}$$

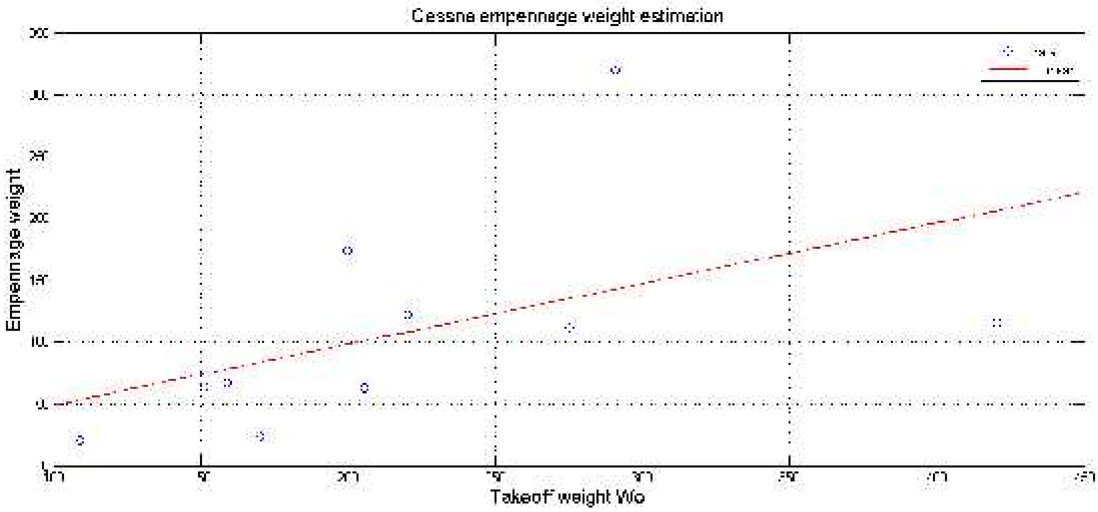


Figure (7-3) Cessna empennage weight estimation

7.2 Usaf Method:

The equations suggested by Usaf for finding empennage weight estimation should be applied to aircrafts with performance doesn't exceed 300 knots speed.

7.2.1 Horizontal tail:

$$W_h = 127 \times \left(\left(\frac{w_o N_z}{10^5} \right)^{0.8} \times \left(\frac{S_h}{100} \right)^{1.2} \times 0.289 \times \left(\frac{l_h}{10} \right)^{0.4} \times \left(\frac{b_h}{t_h} \right)^{0.5} \right)^{0.4} \qquad (7\text{-}3)$$

Horizontal tail weight for UAV having takeoff weight 220 kg is 7.217 kg.

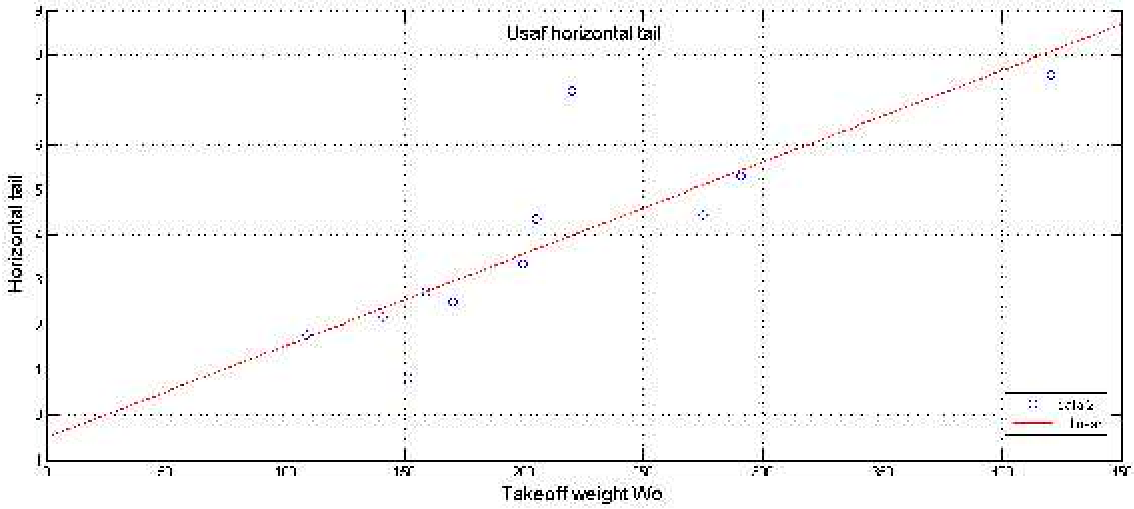


Figure (7-4) Usaf horizontal tail weight estimation

From figure (7-4), the horizontal tail weight for UAV having W_o 220 kg is 4.3 kg.

By substitution in equation () for V-tail weight estimation we get:

$$W_h = 127 \times \left(\left(\frac{485 \times 9}{10^5} \right)^{0.8} \times \left(\frac{13}{100} \right)^{1.2} \times 0.289 \times \left(\frac{10.14}{10} \right)^{0.4} \times \left(\frac{8}{0.192} \right)^{0.5} \right)^{0.4}$$

$$W_h = 15.87\ ll\ = 7.2\ k$$

7.2.2 Vertical Tail:

$$W_v = 98.5 \times \left(\left(\frac{w_o N_z}{10^5} \right)^{0.8} \times \left(\frac{S_v}{100} \right)^{1.2} \times 0.289 \times \left(\frac{b_v}{t_v} \right)^{0.5} \right)^{0.4} \qquad (7\text{-}4)$$

By substitution in equation (7-4) for V-tail weight estimation we get:.

$$W_v = 98.5 \times \left(\left(\frac{485 \times 9}{10^5} \right)^{0.8} \times \left(\frac{5.7}{100} \right)^{1.2} \times 0.289 \times \left(\frac{2.82}{0.24} \right)^{0.5} \right)^{0.4}$$

$$W_v = 5.56\ lb = 2.52\ kg$$

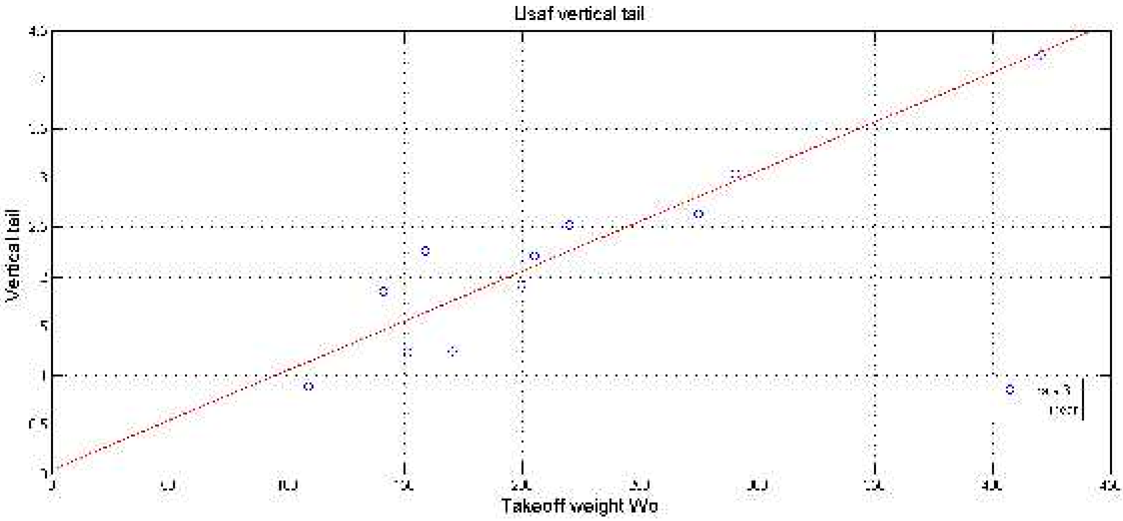


Figure (7-5) Usaf vertical tail weight estimation

The empennage weight of the UAV according to Usaf equations is 9.723 kg kg. Figure (7-6).

7.2 + 2.52 = 9.72 kg

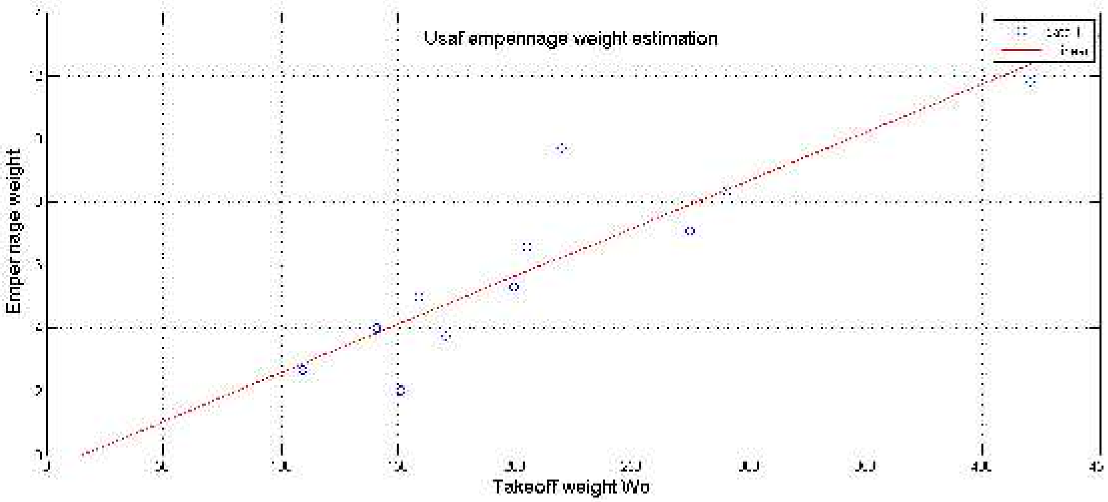


Figure (7-6) Usaf empennage weight estimation

7.3 Torenbeek Method:

The following equation is applied to light transport aircrafts which has dive speed less than 200 knots.

$$W_e = 0.04 \times (N_z \times (S_v + S_h)^2)^{0.7} \tag{7-5}$$

Torenbeek equation, (7-5), is a simple equation and it is completely depends upon the areas of both horizontal and vertical tails and ultimate load factor.

From figure (7-7) the outcome of the empennage weight for UAV weighs 220 kg, is 7.63 kg.

By substitution in Torenbeek equation we get:

$$W_e = 0.04 \times (9 \times (5.7 + 13)^2)^{0.7}$$

$$W_e = 16.8\ lb = 7.63\ k$$

Empennage weight for UAV weights 220 kg, is 7.63 kg.

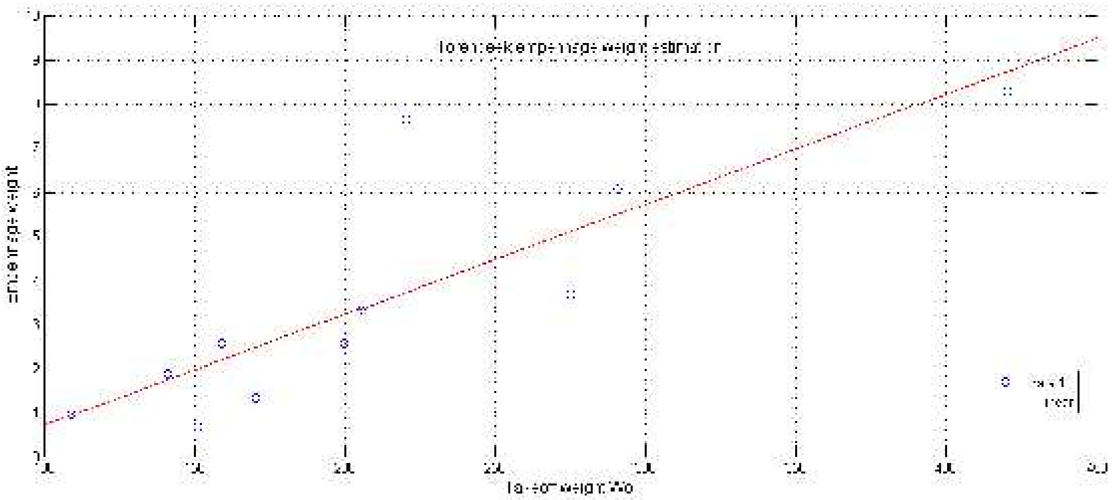


Figure (7-7) Torenbeek empennage weight estimation

7.4 Raymer Method:

The following equations from Raymer are established for general aviation aircrafts.

7.4.1 Horizontal tail:

Raymer equation for horizontal tail weight estimation is:

$$W_h = 0.016 \times \left(N_z \times W_d\right)^{0.4} \times q^{0.1} \times S_h^{0.8} \times \left(\frac{100 \times t/c}{\cos \Lambda}\right)^{-0.1} \times \left(\frac{A}{\cos^2 \Lambda_h}\right)^{0.0} \times \lambda_h^{-0.0} \quad (7\text{-}6)$$

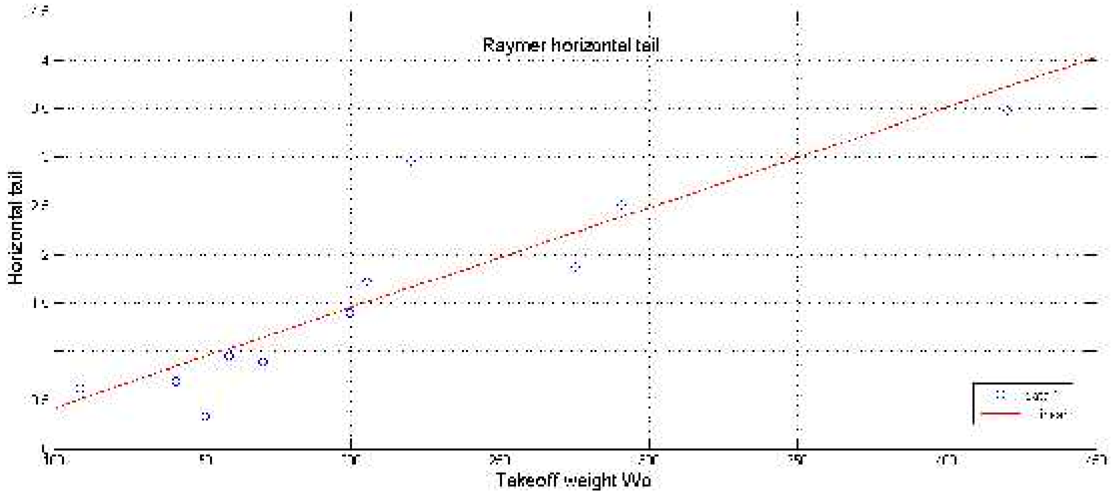


Figure (7-8) Raymer horizontal tail weight estimation

From figure (7-8) the result of the horizontal tail weight for UAV weights 220 kg is 2.96 kg.

7.4.2 Vertical tail:

$$W_h = 0.073 \times \left(1 + 0.2 \times \frac{H_t}{H_v}\right) \times \left(N_z \times W_d\right)^{0.3} \times q^{0.1} \times S_h^{0.8} \times \left(\frac{100 \times t/c}{\cos \Lambda}\right)^{-0.4} \times \left(\frac{A}{\cos^2 \Lambda_h}\right)^{0.3} \times \lambda_h^{0.09} \quad (7\text{-}7)$$

From figure (7-9) the result of the vertical tail weight for UAV weights 220 kg is 4.94 kg.

From figure (7-10) the empennage weight for UAV weighs 220 kg is 7.45 kg.

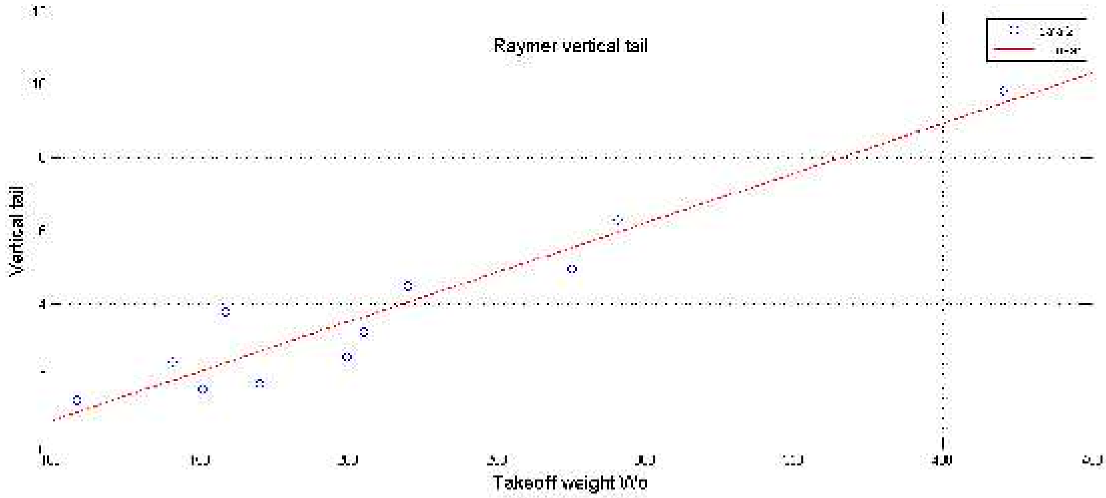


Figure (7-9) Raymer vertical tail weight estimation

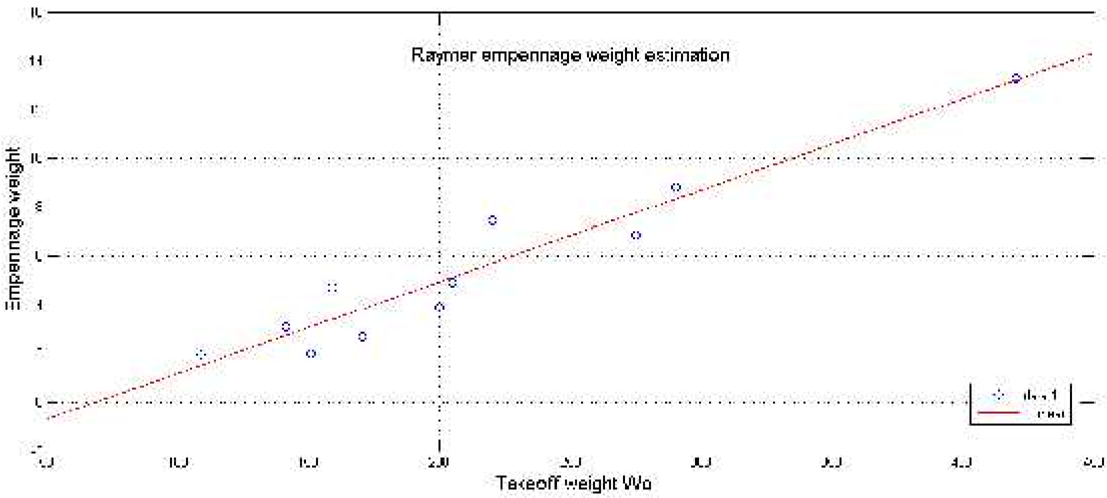


Figure (7-10) Raymer empennage weight estimation

7.5 Kundu Method:

Horizontal and vertical tails are lifting surfaces. The empennage does not have an engine or undercarriage installation.

Both the horizontal and vertical tails plane mass estimations have a similar form but they differ in the values of constants used.

The equation used here is established for Civil Aircraft.

$$M_e = 0.0213 \times (M_t \times N_z)^{0.4} \times S_w^{0.7} \times A \times (1+\lambda)^{0.4} / (\cos\Lambda \times t/c^{0.4}) \quad (7\text{-}8)$$

M_e - empennage mass

M_t - takeoff weight mass

For nonmetals are used, if there is reduction in mass due to lighter material, then the mass is reduced by that factor. If there is a 10% mass saving, then:

M_E *nonmetal* = 0.9 × *MEM*$_E$ *all metal*

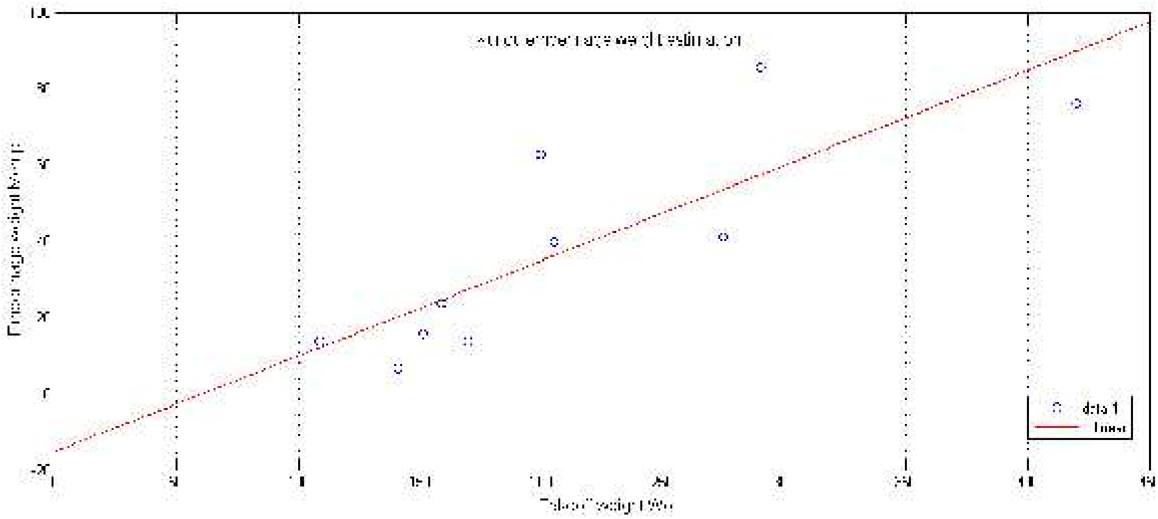


Figure (7-11) Kundu empennage weight estimation

7.5.1 Horizontal tail:

$$M_H = 0.02 \times k \quad \times (M_t \times N_z)^{0.4} \times S_w^{0.7} \times A \times (1+\lambda)^{0.4}/(c \quad \Lambda \times t/c^{0.4}) \qquad (7\text{-}9)$$

For all H-tail movement, use k_{conf} = 1.05; otherwise, 1.0.

$$M_H = 0.02 \times k \quad \times (485 \times 9)^{0.4} \times 13^{0.7} \times 5 \times (1+1)^{0.4}/(c \quad 0 \times 0.12^{0.4})$$

= 127.29 lb =57.8 kg

7.5.2 Vertical tail:

$$M_v = 0.0215 \times k \quad \times (M_t \times N_z)^{0.4} \times S_w^{0.7} \times A \times (1+\lambda)^{0.4}/(c \quad \Lambda \times t/c^{0.4}) \qquad (7\text{-}10)$$

For V-tail configurations, use k_{conf} = 1.1 for a T-tail, 1.05 for a midtail, and 1.0 for a low tail.

$$M_V = 0.0215 \times k \quad \times (485 \times 9)^{0.4} \times 5.7^{0.7} \times A \times \frac{(1+0.8)^{0.4}}{(c \quad 20 \times 0.12^{0.4})} =$$

$$= 45.85 \text{ lb} = 20.8 \text{ kg}$$

$$M_H + M_V = 57.8 + 20.8 = 78.6\ kg$$

For 220 kg takeoff weight aircraft, 78.6 kg for empennage is not satisfied.

7.6 Jay Gundlach method:

The formula used by Gundlach is established for both small and big aircrafts by changing w_a value according to the aircraft type. W_a ranges between 3.5–8 lb/ft2 for supersonic fighters and between 0.8 – 1.2 for small aircrafts.

$$W = w_a \times (S_h + S_v)\ , \qquad w_a = 0.8: 1.2 \text{ lb/ft2} \qquad (7\text{-}11)$$

For $w_a = 1.2$

$$W = 1.2 \times (13 + 5.7) = 22.44\ lb = 10.19\ kg$$

For $w_a = 1$

$$W = 1 \times (13 + 5.7) = 18.7\ lb = 8.48\ kg$$

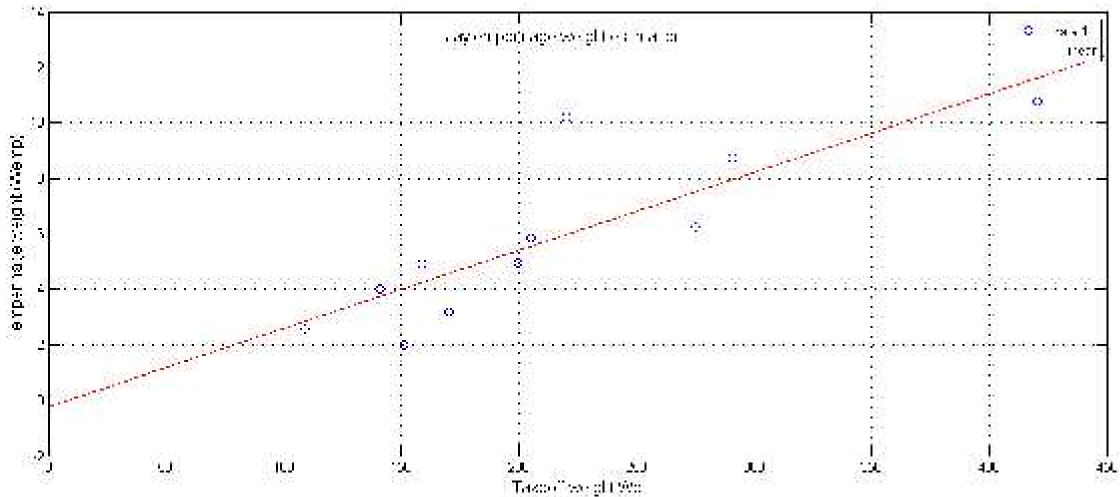


Figure (7-12) Jay Gundlach empennage weight estimation

7.7 Kroo method:

Kroo introduces two formulas for both horizontal and vertical tails.

7.7.1 Horizontal tail:

The horizontal tail weight, including elevator, is determined similarly, but the weight index introduces both exposed and gross horizontal tail areas as well as the tail length (distance from airplane c.g. to aerodynamic center of the horizontal tail). The method assumes that the elevator is about 25% of the horizontal tail area.

$$W_h = 5.25 \times S_h + 0.8 \times 10^{-6} \times \frac{\left(N_z \times {b_h}^3 \times W_o \times m \quad \times S_h^{0.5}\right)}{\left(\left(\frac{t_h}{c_h}\right) \times (\cos \ )^2 y_h \times l_h \times S_h^{1.5}\right)} \tag{7-12}$$

$$W_h = 31\ k$$

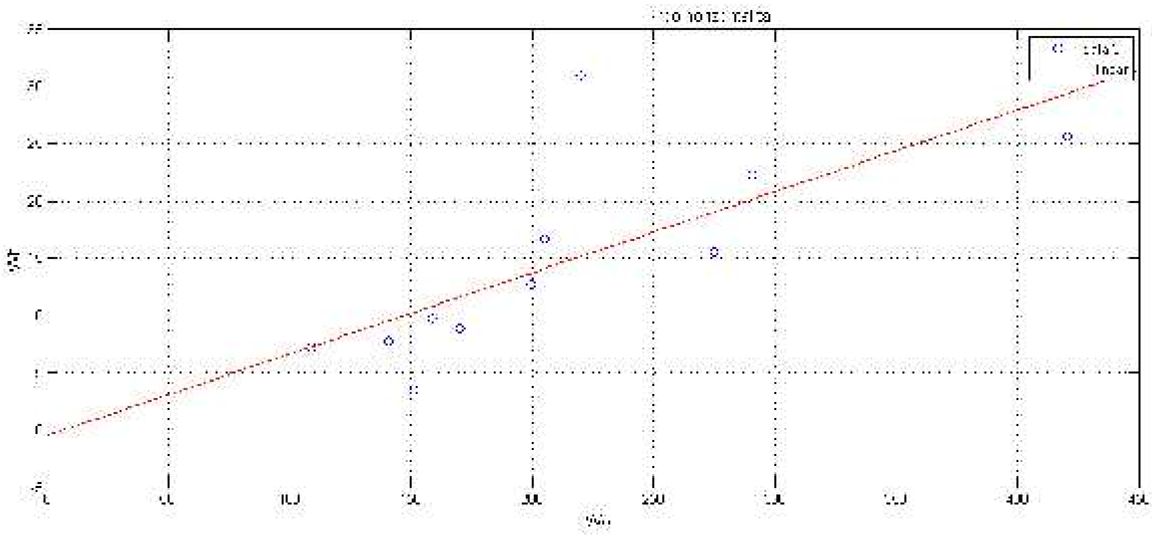


Figure (7-13) Kroo horizontal tail weight estimation

7.7.2 Vertical tail:

The rudder itself may be assumed to occupy about 25% of S_V and weighs 60% more per unit area. The weight of the vertical portion of a T-tail is about 25% greater than that of a conventional tail; a penalty of 5% to 35% is assessed for vertical tails with center engines.

$$W_V = 2.62 \times S_v + 15 \times 10^{-6} \times \frac{\left(N_z \times {b_v}^3 \left(0.8 + 0.44 \times \left(W_o / S_r \quad \right)\right)\right)}{\left(\left(\frac{t_h}{c_h}\right) \times (\cos y_h)^2\right)} \tag{7-13}$$

$$W_V = 17.8\ k$$

Total weight of the empennage group, Figure (7-15), for our aircraft using Kroo method is 47.8 kg for 220 kg takeoff weight.

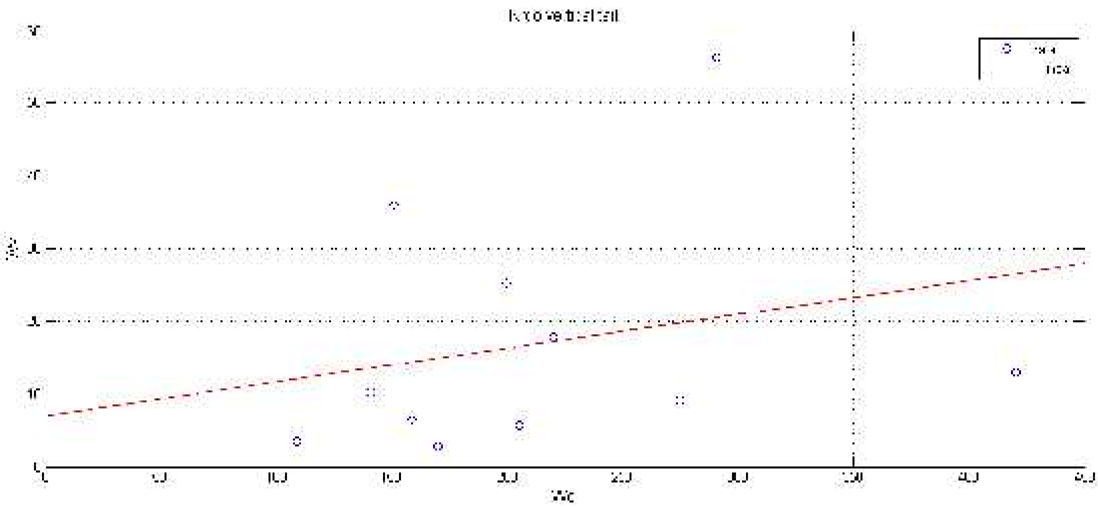


Figure (7-14) Kroo vertical tail weight estimation

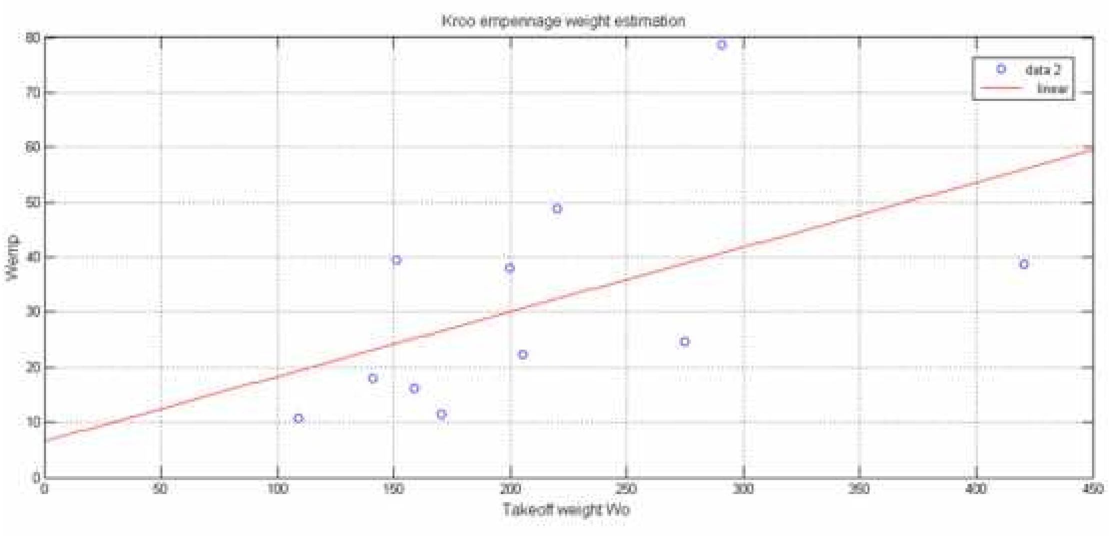


Figure (7-15) Kroo empennage weight estimation

7.8 Howe method:

7.8.1 Horizontal tail:

$$W_h = 0.047 \times V_d \times S_h^{1.2} \tag{7-14}$$

W_h for UAV having 220 kg takeoff weight is 2.98 kg. Figure (7-16).

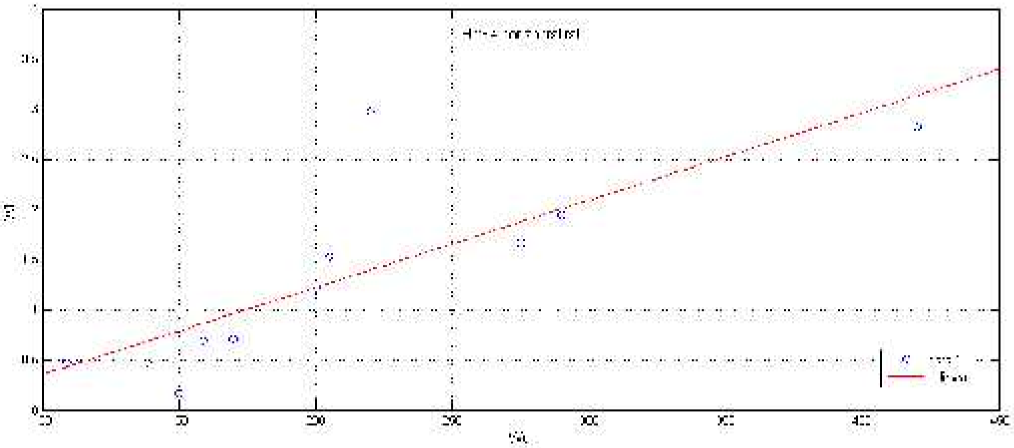


Figure (7-16) Howe horizontal tail weight estimation

7.8.2 Vertical tail:

$$W_v = 0.065 \times k \times V_d \times S_v^{1.1} \tag{7-15}$$

Wv for UAV having 220 kg takeoff weight is 1.566 kg. Figure (7-17).

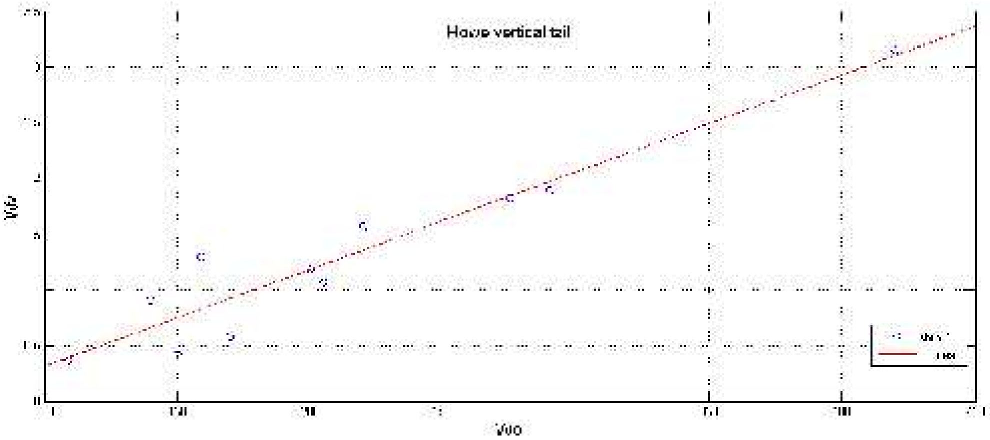


Figure (7-17) Howe vertical tail weight estimation

Total weight of empennage group, Figure (7-18), got by Howe equations for 220 kg takeoff weight aircraft is approximately:

2.98 + 1.556 = 4.54 kg

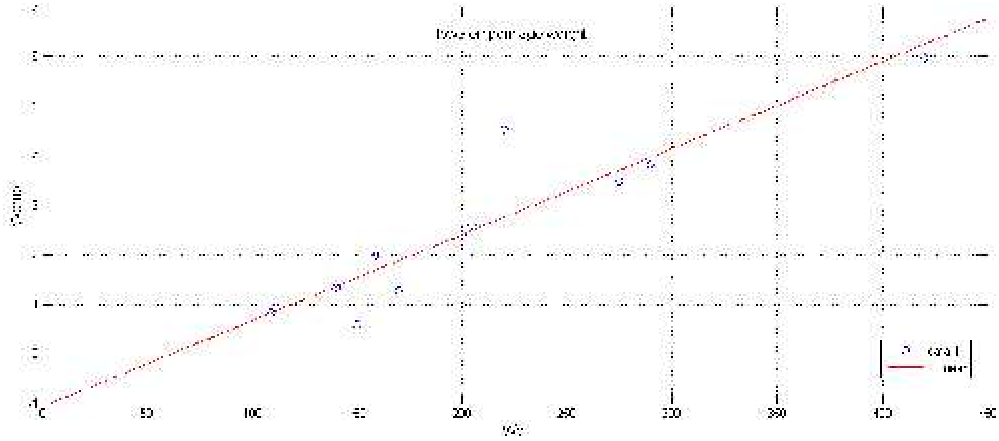


Figure (7-18) Howe empennage weight estimation

8. Fuselage weight estimation:

8.1 Introduction

A fuselage of aircraft is essentially a hollow shell designed to carry a payload, and the engine. Many parameters controlling the fuselage weight such as; its average diameter, length, skin area or wetted area, shell volume, pressurization, aircraft load factor, maximum possible speed which is dive speed, fuselage configuration, and construction material. High aircraft velocity and load factors require more material for structural integrity. Additional reinforcement weight is needed for the installation of engines and the landing gear on the fuselage. Pressurization of the cabin raises the fuselage-shell hoop stress level that requires more weight for reinforcement, and a rear-mounting cargo door is also a large increase in weight. (5)

8.2 Fuselage weight estimation methods:

A fuselage is defined as a hollow shell designed to accommodate a payload. The size of the fuselage is obviously controlling the mass of the body, The drivers for the fuselage group mass are its length, *L*; diameter, D_{ave}; shell area and volume, and additionally, the mass of the body depends on the aircraft layout (e.g. engine and undercarriage positions). (5)

8.2.1 Jenkinson method:

The following formula (8-1) which is suggested by Howe is recommended for civil aircrafts (50-300 seats).

$$W_b = 0.039 \times \left(2 \times L \times D \times \sqrt{V_d}\right)^{1.5} \tag{8-1}$$

W_b – Fuselage weight

D – Equivalent diameter

V_d – dive speed

L – Fuselage length

It is recommended that the above mass be amended as follows:

Increased 8% for pressurized cabin

Increased 4% for fuselage mounted engines

Increased 7% for fuselage mounted main undercarriage

Increased 10% for large cargo door

Reduced 4% if the fuselage is free from structural discontinuity

$$W_b = 0.039 \times \left(2 \times 5.16 \times 0.4 \times \sqrt{55.55}\right)^{1.5}$$

$$W_b = 7.49\ k$$

From Jenkinson equation shows the fuselage weight for UAV having 220 Kg takeoff weight is about 7.49 kg.

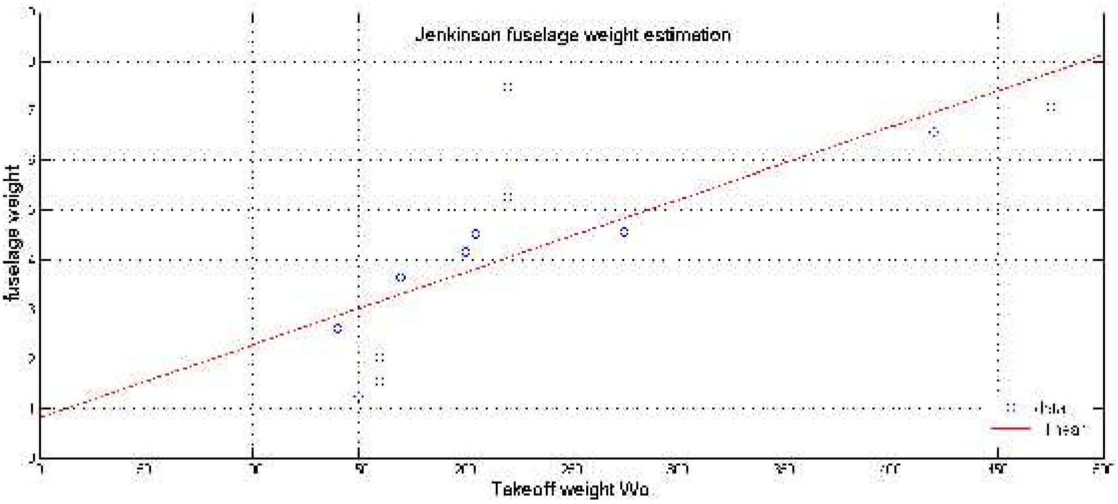


Figure (8-1) Jenkinson fuselage weight estimation

By Increased 4% for fuselage mounted engines and 7% for fuselage mounted main undercarriage, the total weight estimated of fuselage becomes 21 kg.

8.2.2 Howe method:

Howe has modified his equation for fuselage mass estimation to be:

$$W_b = 0.044 \times \left(L \times (B + H) \times \sqrt{V_d}\right)^{1.5} \qquad (8\text{-}2)$$

B – Fuselage maximum width

H – Fuselage maximum depth

V_d – dive speed

L – Fuselage length

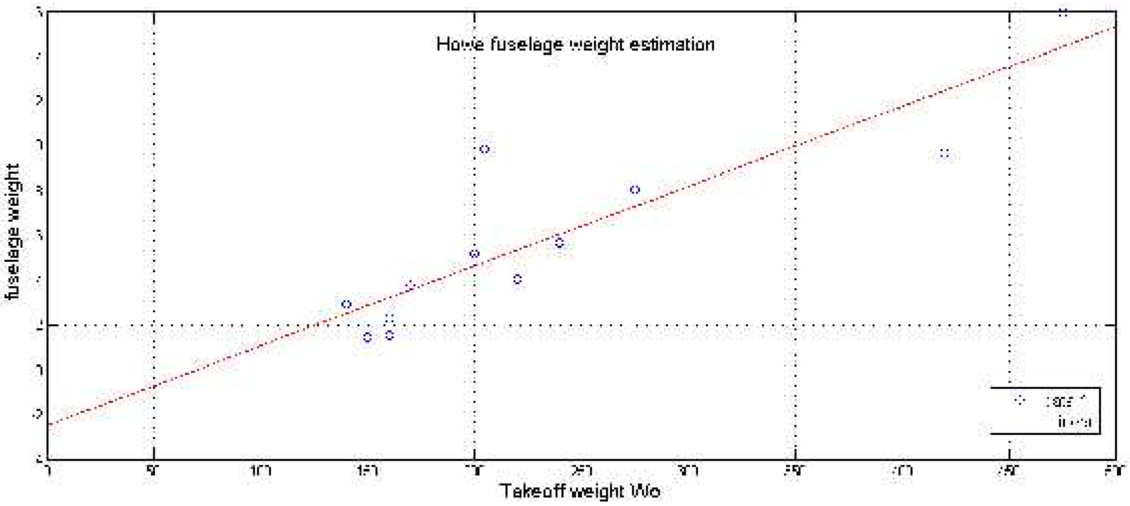


Figure (8-2) Howe fuselage weight estimation

$$W_b = 0.044 \times \left(5.16 \times (0.4 + 0.4) \times \sqrt{55.55}\right)^{1.5}$$

$$W_b = 9.82\ k$$

The equation result shows that the fuselage weight for UAV having 220 Kg takeoff weight is about 9.82 Kg.

8.2.3 Kundu method:

Kundu suggested the equation below for the fuselage weight with a fixed undercarriage can be written as:

$$W_b = 0.038 \times k_u \times k_e \times (W_o \times N_z)^x \left(2 \times L \times D \times \sqrt{V_d}\right)^{1.5} \qquad (8\text{-}3)$$

D – average diameter

V_d – dive speed

L – Fuselage length

ku - for fuselage undercarriage

ke - for fuselage-mounted engines = 1.05 to 1.07

Nz - Ultimate load factor

W_o - Takeoff weight

The value of index *x* depends on the aircraft size: 0 for aircraft with an ultimate load (n_{ult}) < 5 and between 0.001 and 0.002 for ultimate loads of (n_{ult}) >5 (i.e., lower values for heavier aircraft). In general, $x = 0$ for civil aircraft; therefore, (MTOM $\times$ n_{ult}) $x = 1$. The value of index *y* is very sensitive. Typically, *y* is 1.5, but it can be as low as 1.45. It is best to fine-tune with a known result in the aircraft class and then use it for the new design.

$$W_b = 0.038 \times 1.04 \times 1.07 \times (220 \times 9)^{0.0} \quad \left(2 \times 5.16 \times 0.4 \times \sqrt{55.55}\right)^{1.5}$$

$$W_b = 8.79\ k$$

From Kundu equation (8-3) the fuselage weight for UAV having 220 Kg takeoff weight is about 8.79 Kg.

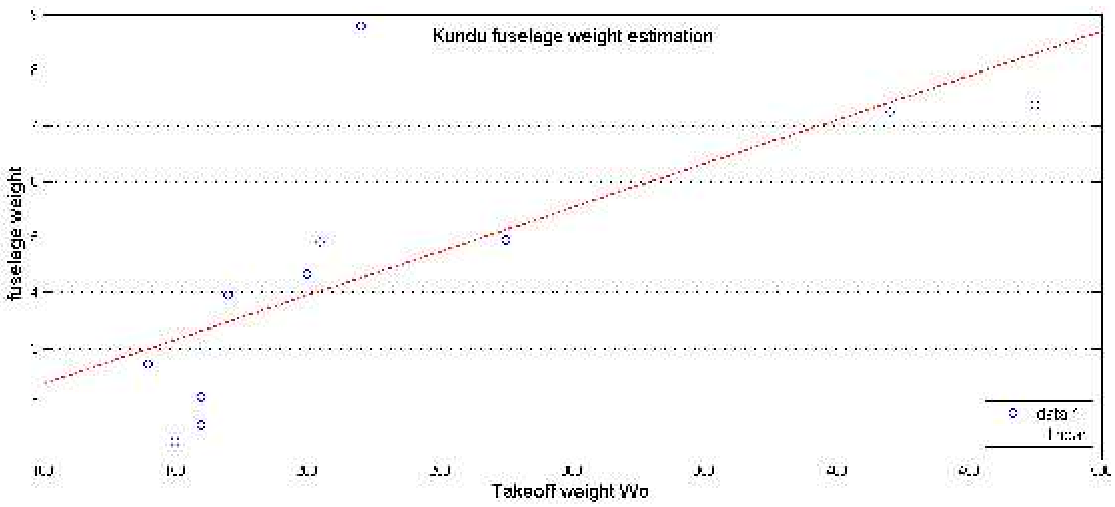


Figure (8-3) Kundu fuselage weight estimation

8.2.4 Roskam method:

The following equation (8-4) for estimating fuselage weight, which suggested by Roskam is called the General Dynamic method:

$$W_b = 10.43 \times k_{in}^{1.4} \times (q/100)^{0.2} \times (W_o/1000)^{0.9} \times (L/H)^{0.7} \quad (8\text{-}4)$$

Wb - Estimated fuselage weight in [lb]

q - dynamic pressure in psf

L - fuselage length

H - fuselage hieght

Kinlet - for inlets in or on the fuselage

Wo - Takeoff weight

By substitution in equation (8-4) we get:

$$W_b = 10.43 \times 1.25^{1.4} \times (16.4/100)^{0.2} \times (485/1000)^{0.9} \times (16.93/1.31)^{0.7}$$

$$W_b = 12.04\ k$$

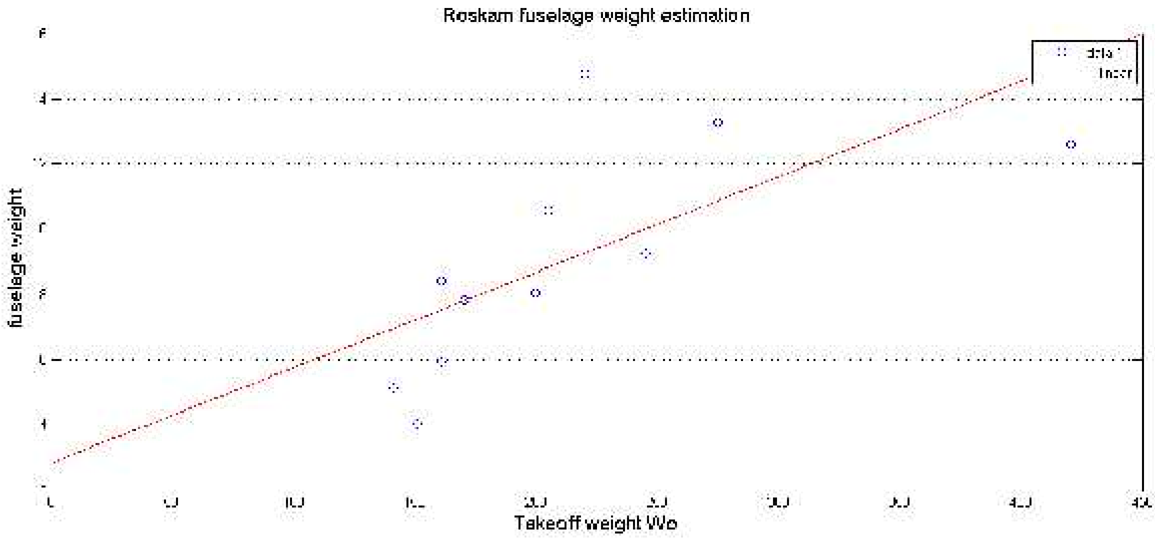


Figure (8-4) Roskam fuselage weight estimation

From Roskam equation shows the fuselage weight for UAV having 220 Kg takeoff weight is about 12.04 Kg.

8.2.5 Raymer method:

Raymer equation (8-5) for general aviation aircrafts to estimate fuselage weight is as follow:

$$W_b = 0.052 \times (W_o \times N_z)^{0.1} \times q^{0.2} \times S_f^{1.0} \times L^{-0.0} \times (L/D)^{-0.0} \qquad (8\text{-}5)$$

S_f - fuselage surface area [ft^2]

L - fuselage length [ft]

q - Dynamic pressure [lb/ft2]

D - fuselage diameter [ft]

Nz - Ultimate load factor

Wo - Takeoff weight. lb

By substitution in Raymer equation (8-5) the result shows that the fuselage weight for UAV having 220 Kg takeoff weight is about 10.8 Kg.

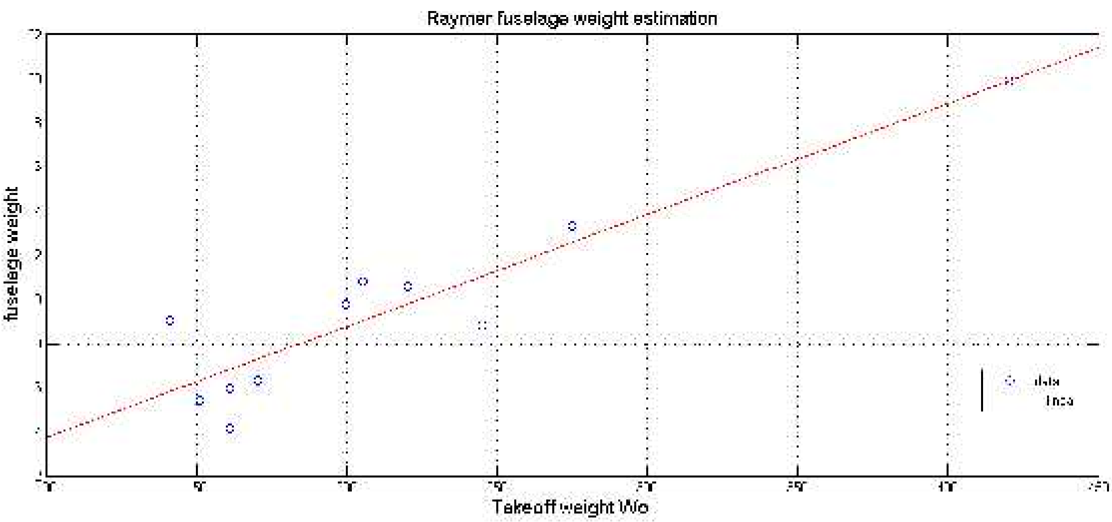


Figure (8-5) Raymer fuselage weight estimation

8.2.6 Usaf method:

Usaf equation (8-6) for general aviation aircrafts to estimate fuselage weight is as follow:

$$W_b = 200 \times \left(\left(W_o \times \frac{N_z}{10^5} \right)^{0.2} \times \left(L_f/10 \right)^{0.8} \times \left((B+H)/10 \right) \times (V_c\ /100)^{0.3} \right)^{1.1} \tag{8-6}$$

L_f - fuselage length (ft)

H - Max fuselage hight

B - Max fuselage width

Vcr - design cruise speed (knots)

Nz - Ultimate load factor

Wo - Takeoff weight. lb

By substitution in Usaf equation (8-6), shows that the fuselage weight for UAV having 220 Kg takeoff weight is about 11.8 Kg.

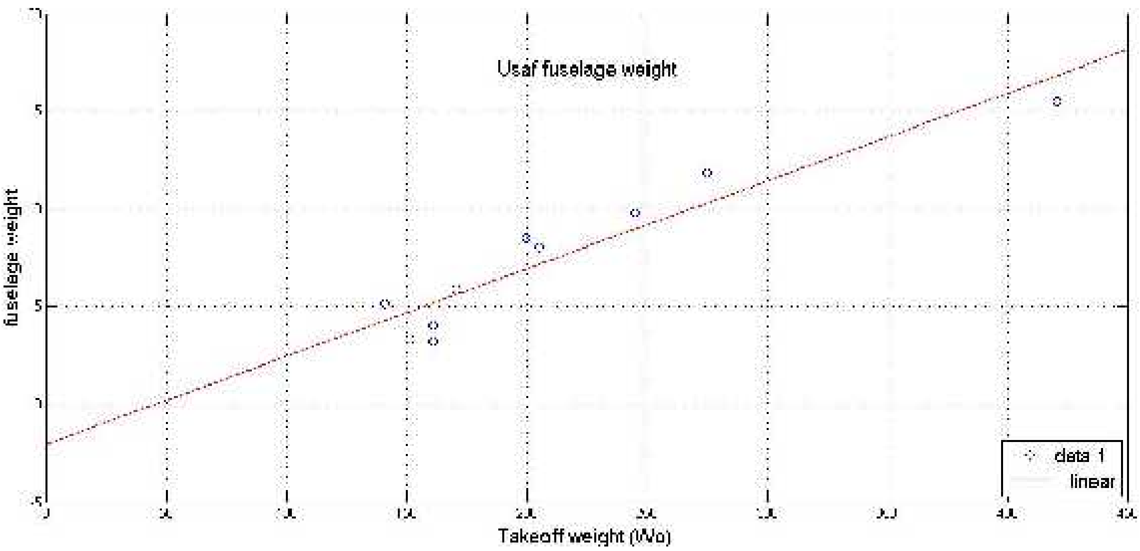


Figure (8-6) Usaf fuselage weight estimation

8.2.7 Kroo method:

Fuselage weight is based on gross fuselage wetted area (without cutouts for fillets or surface intersections and upon a pressure-bending load parameter.

Kroo suggested Desktop Aeronautics to be used for fuselage weight estimation:

$$W_b = (1.051+0.102\times I_{fu} \quad)\times S_f \qquad (8\text{-}7)$$

Wb - Estimated fuselage weight in [lb]

I_{fuse} - fuselage index

S_f - fuselage wetted area

wo - take off weight

The pressure index is: Ip = 1.5E-3 * P * B

The bending index is: Ib = 1.91E-4 N * W * L / H^2

P = maximum pressure differential (lb / sq ft)

B = fuselage width (ft)

H = fuselage height (ft)

L = fuselage length (ft)

N = limit load factor at ZFW

W = ZFWmax - weight of wing and wing-mounted engines, nacelles and pylons.

I_pWhen fuselage is not pressure-dominated: $I_{fuse} = (I_p^2 + I_b^2) / (2\ I_b)$

$I_{fuse} = 5\times10^{-7}$

$$W_b = (1.051 + 0.102 \times (5 \times 10^{-7})) \times 52.35$$

$$W_b = 25\ k$$

Desktop Aeronautics equation (8-7) shows the fuselage weight for UAV having 220 Kg takeoff weight is about 25 Kg. Figure (8-7)

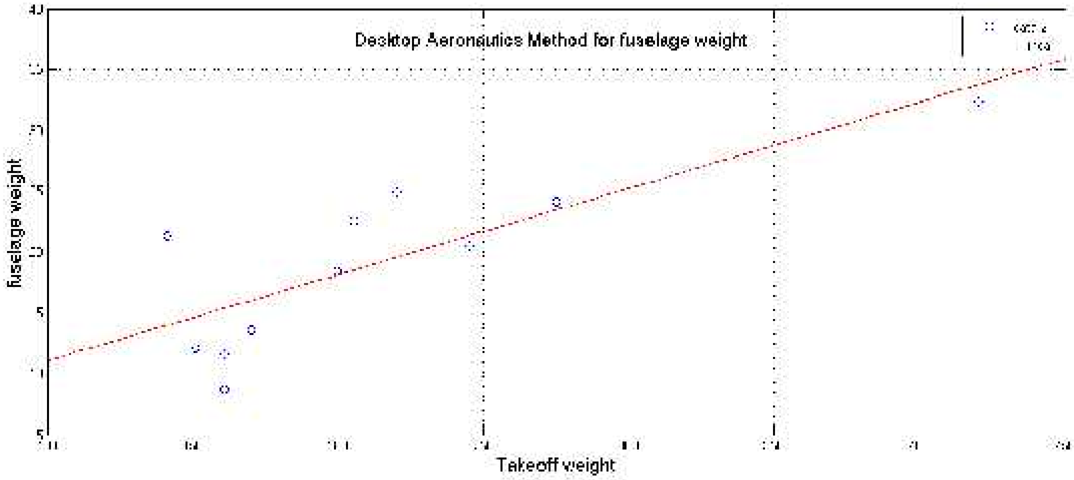


Figure (8-7) Kroo fuselage weight estimation

8.2.8 Jay Gundlach method:

Jay say, a general fuselage structural weight equation for a structure for a semi monocoque or composite shell fuselage for subsonic or transonic UAS weighing between 1 to 800,000 lb is:

$$W_b = 0.5257 \times f \times f \times f \times f \times f \times L^{0.3} \times (W \times N)^{0.4} \times (1.3 \times Vd./100)^2 \quad (8\text{-}8)$$

Wb - Estimated fuselage weight in lb

wc - weight of the components carried within the structure in pounds

Vd - design dive speed in knots equivalent air speed

Fn - Nose gear on the fuselage factor

FP - Pressurized fuselage factor

FM - Main gear on the fuselage factor

FV - Vertical tail on the fuselage factor

Ft - Materials factor

L - fuselage length

wo - max equivalent weight

fm=1.07; fn=1.04; fv=1; ft=1; fp=1;

$$W_b = 39.33\ k$$

Jay Gundlach equation (8-8) shows that the fuselage weight for UAV having 220 Kg takeoff weight is 39.33 Kg. Figure (8-8)

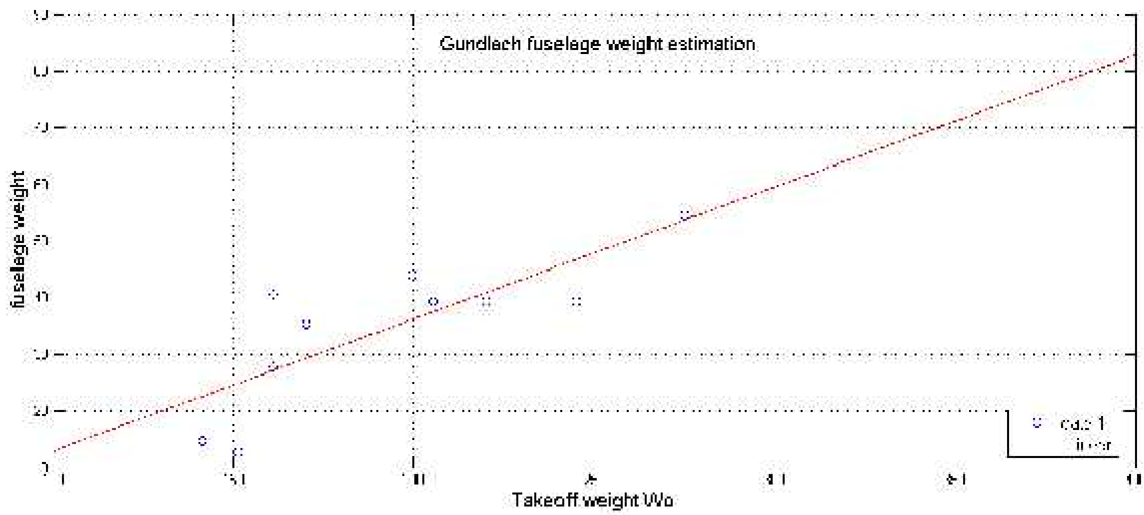


Figure (8-8) Gundlach fuselage weight estimation

9. Landing gear weight estimation:

9.1 Introduction:

Landing gear of the aircraft consists of a set of parts represents entirety undercarriage group, which is composed mainly of strut, wheel, tire, retraction system, shock absorber, and braking system. The weight of the landing gear is largely a function of aircraft weight at landing, and affected by landing gear height, landing gear configuration, landing speed, landing run, retraction system, construction material, and landing ultimate factor. (6)

Landing gear is estimated as a function of takeoff gross weight. (3)

9.2 Methods of Landing gear weight estimation

9.2.1 Howe method:

The following equation applies to general aviation aircrafts weight less than 10000 lb (4545 Kg). (10)

$$W_g = 0.048\times W_o \tag{9-1}$$

$$= 0.048\times 220 = 10.56 \text{ kg}$$

Landing gear is calculated by equation (9-1) for takeoff gross weight 220, and the result is 10.56 kg

9.2.2 Kroo method:

Landing gear is weight fraction of the takeoff weight, and estimated to be about 4.0% of the take-off weight. This is the total landing gear weight including, actuating system, structure, and the rolling assembly consisting of wheels, tires, and brakes. The rolling assembly is approximately 39% of the total gear weight. (9)

$$W_g = 0.04\times W_o \tag{9-2}$$

$$= 0.04\times 220 = 8.8 \text{ kg}$$

Landing gear is calculated by equation (9-2) for takeoff gross weight 220, and the result is 8.8 kg.

9.2.3 Kundu method:

Undercarriage weight depends on an aircraft's takeoff weight. Weight estimation is based on a generalized approach of the undercarriage classes that demonstrate strong statistical relations.

A fuselage-mounted undercarriage usually has short struts.

$$Wg_{.fus} = 0.04 \times W_o$$

For a fixed undercarriage, the mass is 10 to 15% lighter

$$Wg = 0.04\times Wo - 0.1\times (0.04\times Wo) \tag{9-3}$$

$$W_g = 0.04 \times 220 - 0.1\times (0.04 \times 220) = 7.9 \text{ kg}$$

Landing gear is calculated by equation (9-3) for takeoff gross weight 220, and the result is 7.9 kg

9.2.4 Nikolai method:

The following equation established by Usaf and introduced by Nikolai:

$$Wg = 62.21\times (Wo/1000)^{0.84} \tag{9-4}$$

$$W_g = 62.21\times (220/1000)^{0.84} = 15.36 \text{ Kg}$$

Landing gear is calculated by equation (9-4) for takeoff gross weight 220, and the result is 15.36 kg

9.2.5 Pazmany method:

The following equation applies to general aviation aircrafts weight less than 10000 lb (4545 Kg). (10)

For nose wheel type:

$$W_g = 0.055\times W_o \tag{9-5}$$

$$W_g = 0.055 \times W_o$$

$$W_g = 0.055\times 220 = 12.1\ Kg$$

Landing gear is calculated by equation (9-5) for takeoff gross weight 220, and the result is 12.1kg

9.2.6 Jay Gundlach method:

The following general landing-gear weight equation for all scales of unmanned aircraft is:

$$W_g = F_g\times W_o \tag{9-6}$$

$$W_{LG} = F_g \times Wo$$

Where F_g is the landing-gear mass fraction, which will vary from 0.03 to 0.06. This method obscures landing-gear configuration and dimensions.

An initial F_g value of 0.04 is recommended for aircraft that take off and land on paved runways.

$$W_{LG} = 0.04\times 220 = 8.8\ Kg$$

Landing gear is calculated by equation (9-6) for takeoff gross weight 220, and the result is 8.8 kg.

9.2.7 Raymer method:

The following equations apply to general aviation aircrafts.

Main landing gear weight:

$$W_m = 0.095 \times (N_l \times W_l)^{0.7} \times (L_m/12)^{0.4} \tag{9-7}$$

Nose landing gear weight:

$$W_n = 0.125 \times (N_l \times W_l)^{0.5} \times (L_n/12)^{0.8} \tag{9-8}$$

W_l – landing design gross weight. lb

N_l – ultimate landing load factor.

L_m - main gear height. ft

L_n – nose gear hight. Ft

$$W_l = 485\ lb \ , \qquad N_l = 9, \qquad L_m = 1.3\ ft \ , \qquad L_n = 1.15\ ft$$

$$W_m \ = 0.095 \times (9 \times 374)^{0.7} \ \times (1.3/12)^{0.4}$$

$$W_m \ = 8.89\ k$$

$$W_n \ = 0.125 \times (9 \times 374)^{0.5} \ \times (1.15/12)^{0.8}$$

$$W_n \ = 0.77\ k$$

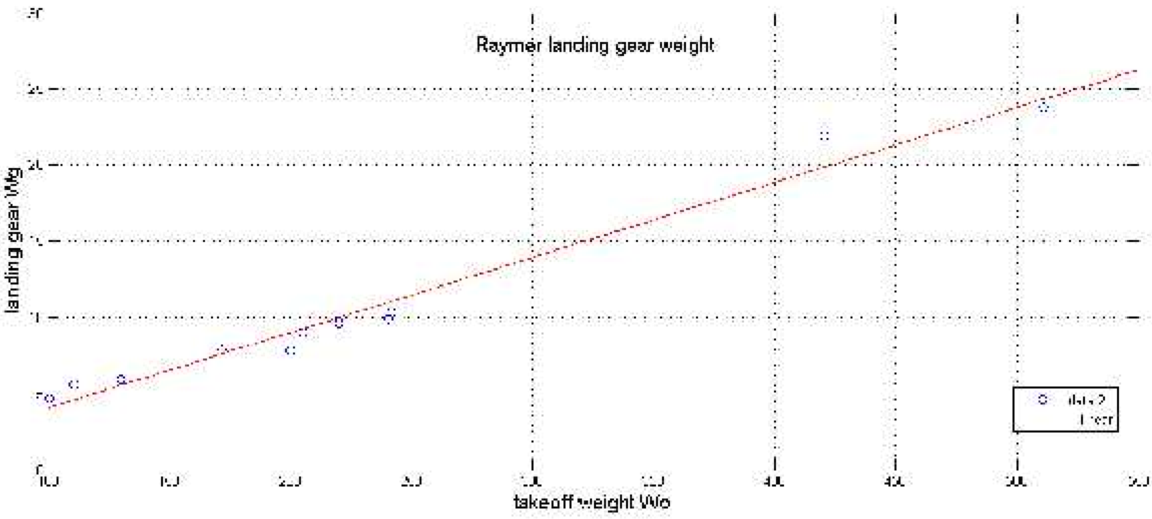


Figure (9-1) Raymer landing gear weight estimation

From figure (9-1), Raymer landing gear weight for UAV having takeoff weight 220 kg is 9.66 Kg.

Landing gear is calculated by equations (9-7) and (9-8) for takeoff gross weight 220, and the result is 9.66 kg.

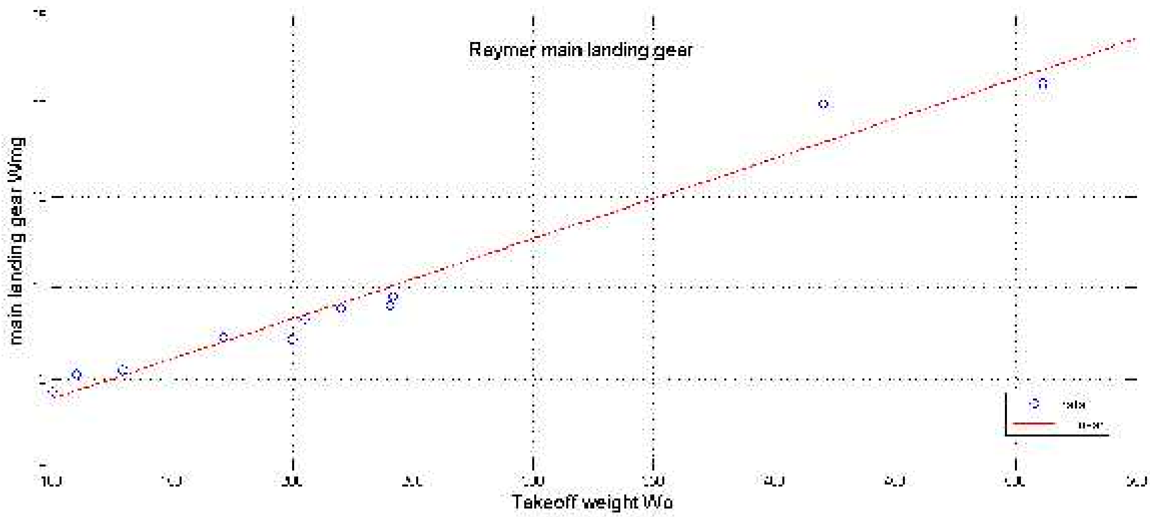


Figure (9-2) Raymer main landing gear weight estimation

Figure (9-2) illustrates main landing gear weight related to takeoff weight, from the figure main landing gear equal to 8.89 kg.

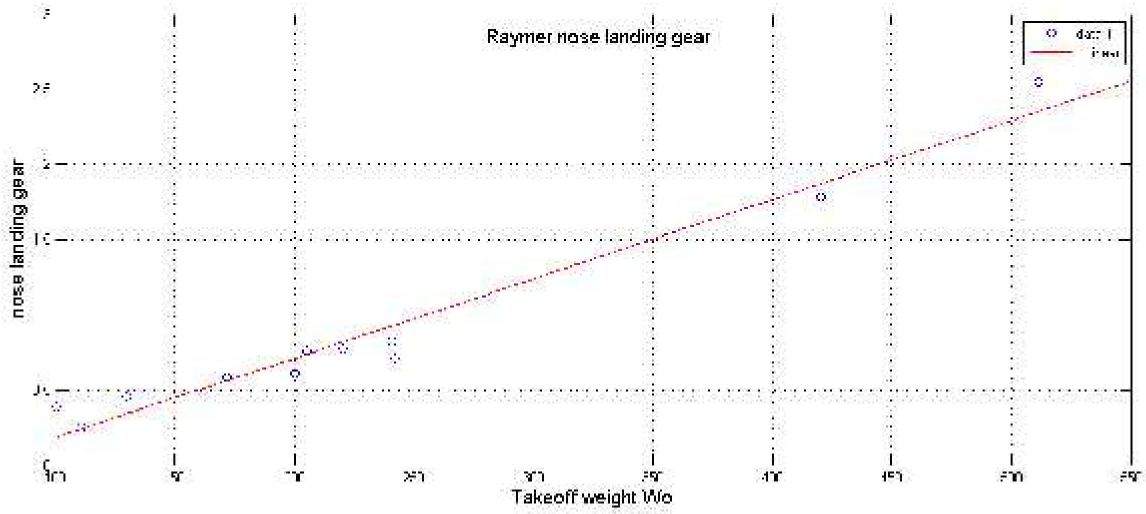


Figure (9-3) Raymer nose landing gear weight estimation

From the figure main landing gear equal to 0.77 kg.

9.2.8 Sadraey method:

Sadraey equation applies to various types of airplanes and it is controlled mainly by landing weight and partly by ultimate load factor, wing span, and aircraft height. The landing gear is calculated as follow:

$$W_{lg} = K_l \times K_{ret} \times K_{lg} \times W_l \times \frac{H_{lg}}{b} \times n_{ult}^{0.2} \quad (9\text{-}9)$$

K_l – landing place factor and is 1.8 for navy aircraft and 1 otherwise.

K_{ret} - 1 for fixed landing gear and 1.07 for retractable landing gear.

K_{lg} – Landing gear weight factor, and ranges between 0.48 and 0.62 for general aviation and home built aircrafts.

b – Wing span.

H_{lg} – Height. of Landing gear

W_{lg} – landing weight of aircraft.

$$W_{lg} = 1 \times 1 \times 0.55 \times 170 \times \frac{0.4}{6.5} \times 9^{0.2}$$

= 8.93 kg

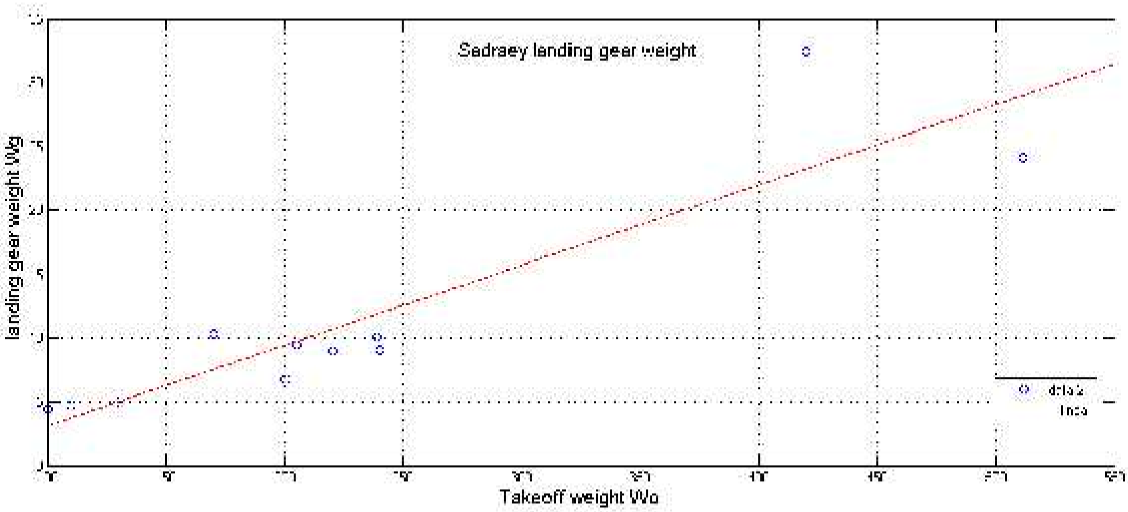


Figure (9-4) Sadreay landing gear weight estimation

From figure (9-4), Sadraey landing gear weight for UAV having takeoff weight 220 kg is 8.93 Kg.

$$Wmg = 0.8 \times Wg \quad (9\text{-}10)$$

$$Wmg = 0.8 \times 12.75 = 7.14 \text{ Kg}$$

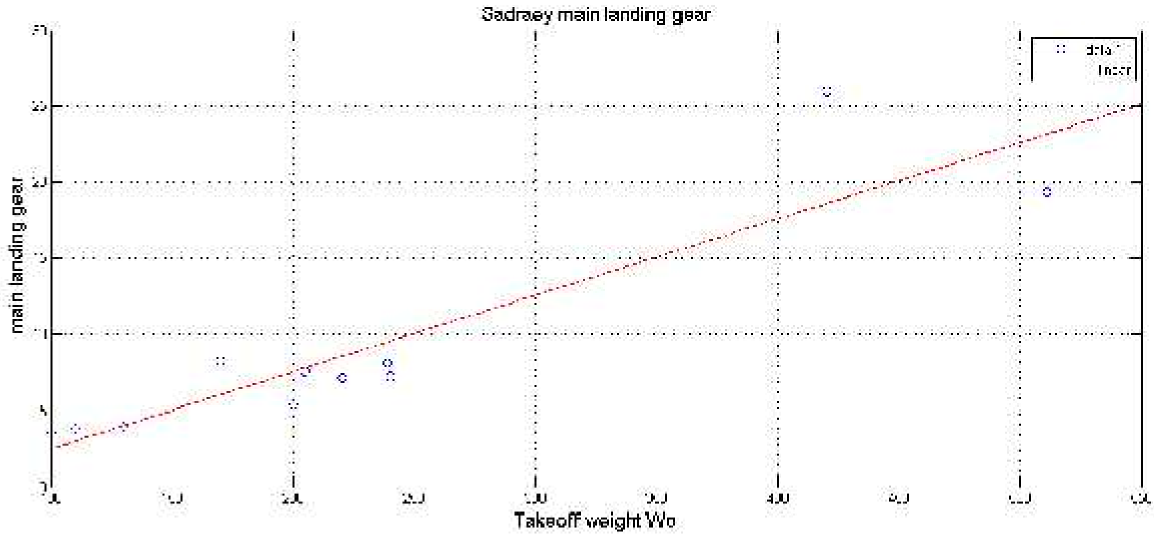


Figure (9-5) Sadreay main landing gear weight estimation

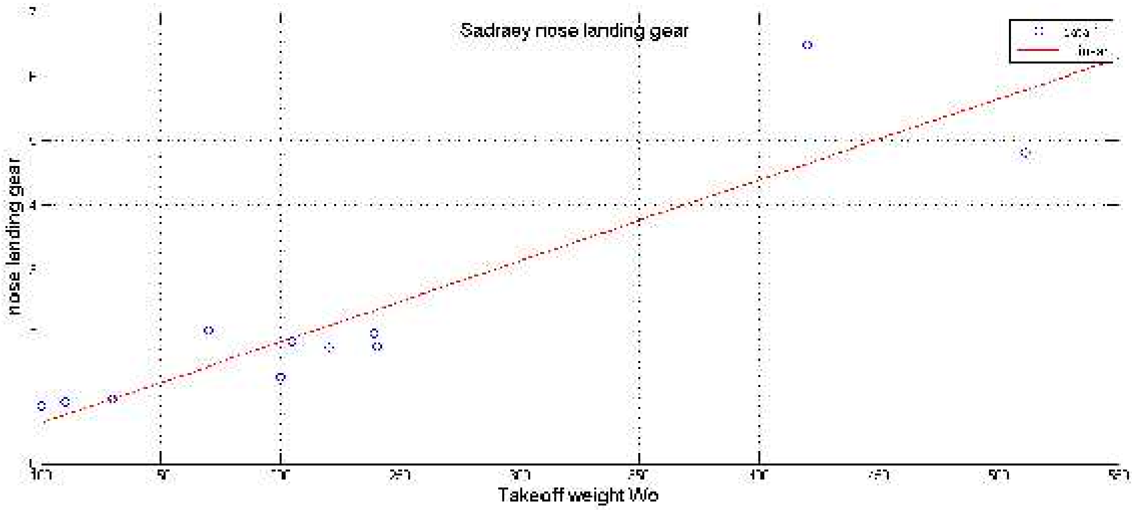


Figure (9-6) Sadreay nose landing gear weight estimation

$$Wng = 0.2 \times Wg \qquad (9\text{-}11)$$

$$Wng = 0.2 \times 12.75 = 1.785 \text{ Kg}$$

9.2.9 Usaf method:

The following equation (9-12) is applied to light and utility type airplanes with performance up to about 300 knots. (3)

$$W_g = 0.054 \times L_m^{0.5} \times (W_l \times N_u)^{0.6} \qquad (9\text{-}12)$$

$$W_g = 0.054 \times 1.3^{0.5} \times (374 \times 9)^{0.6} \quad \text{lb}$$

$$W_g = 15.94\ ll = 7.23\ k$$

From figure (9-7), Usaf landing gear weight for UAV having takeoff weight 220 kg is 7.23 Kg.

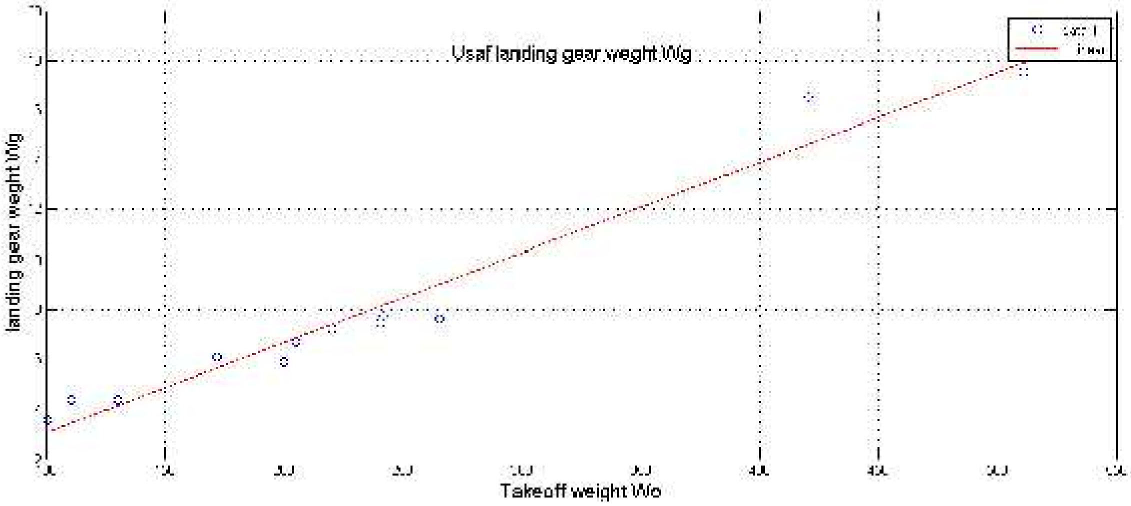


Figure (9-7) Usaf landing gear weight estimation

10. Results and discussion:

10.1 Selected formulas for component weight estimation:

From previous design calculations, we found that the empty weight fraction $\frac{W_E}{W_0}$ of the aircraft is 0.585 which is equal to 128.7 Kg, the empty weight includes wing weight, empennage weight, fuselage weight, carriage weight, and engine group weight.

10.2 Wing weight formulas selection:

Many formulas were used in this thesis to find the wing weight estimation and the results were shown on charts. Because of all these formulas are not established for unmanned aircrafts, so in some cases we see unreasonable and extreme results for wing weight estimation. But in some few cases we got reasonable results.

Comparison of all the results that we got from the equations of previous methods for wing weight estimation of UAV having 220 kg takeoff weight, and the results extracted from the figures, we can see that the formulas of Raymer, Usaf, Turenbeek, Gundlach gave satisfied values of the estimated wing weight .

Wing weight represents about 17-27% of the empty weight. (24)

Wing area in our case is relatively big compared to the aircraft empty weight because of low wing loading value, so we will consider the wing weight to be around 27%. We concluded from the foregoing that the Jay Gundlach formula which founded by Gerard for manned sailplanes or Raymer formula give the best results between all of the other formulas.

Usaf and Torenbeek formulas show acceptable results, but these results give value which are not enough for this big wing. Usaf wing weight is equal to 22.37% of the empty weight and Torenbeek wing weight is equal to 22.63%. Gundlach formula outcome is meet our request but the fitting line slope in Figure (10-1) shows a big difference between 8.5% at 100 kg and 17.6 % at 500 kg takeoff weight which make it unreliable. The result we got from Raymer formula is the best between the previous formulas. Raymer equation offers 33.85 kg for our aircraft wing weight, this equal to 26.3% of Ww/We. Raymer fitting line slope in figure (10-2) shows 10% of Ww/Wo at 100 kg and 12.2 % at 500 kg. For 220 kg the Ww/Wo the wing weight fraction is 15.38% at 220 kg.

Jay Gundlach equation for sailplane:

$$W \quad = 0.0038 \times (N_z \times W_o)^{1.0} \times A^{0.3} \times S_w^{0.2} \times (1+\lambda)^{0.2} \times (t/c)^{0.1}$$

$$W \quad = 0.0038 \times (9 \times 220)^{1.0} \times 8^{0.3} \times 5.27^{0.2} \times (1+1)^{0.2} \times (15)^{0.1}$$

$$W \quad = 35.14\ K$$

This weight represents 27.3% of the aircraft empty weight.

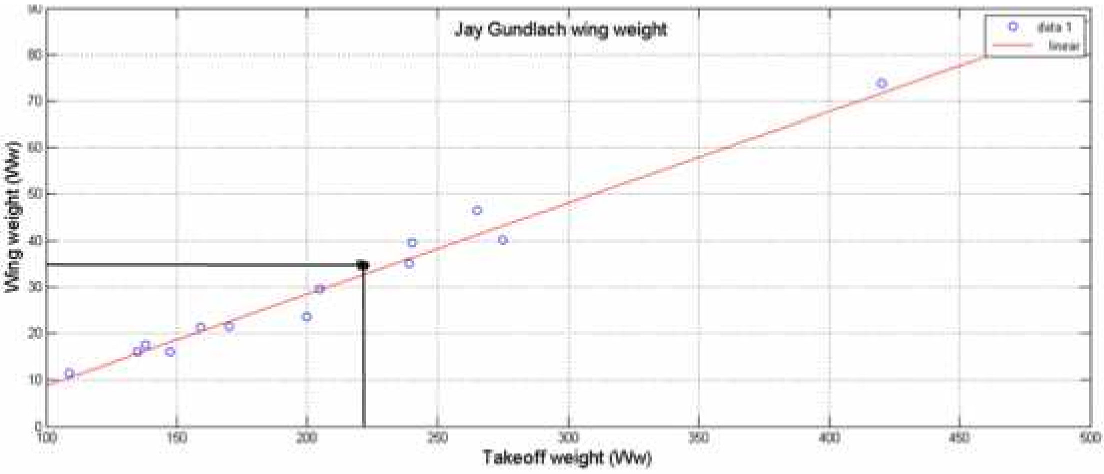


Figure (10-1) Gundlach wing weight estimation

Raymer equation for general aviation:

$$W_w = 0.036 \times S_w^{0.7} \times W_f^{0.0} \times \left(\frac{A}{c \quad {}^2\Lambda}\right)^{0.6} \times q^{0.0} \times \lambda^{0.0} \times \left(\frac{100 \times t/c}{c \quad \Lambda}\right)^{-0.3} \times (N \times w_d)^{0.4}$$

$$W_w = 0.036 \times 56.7^{0.7} \times 113^{0.0} \times \left(\frac{8}{c \quad {}^2 0}\right)^{0.6} \times 16.3^{0.0} \times 1^{0.0} \times \left(\frac{1 \quad \times 0.1}{c \quad 0}\right)^{-0.3} \times (9 \times 220)^{0.4} = 33.85\ k$$

33.85/128.7 = 26.3 %

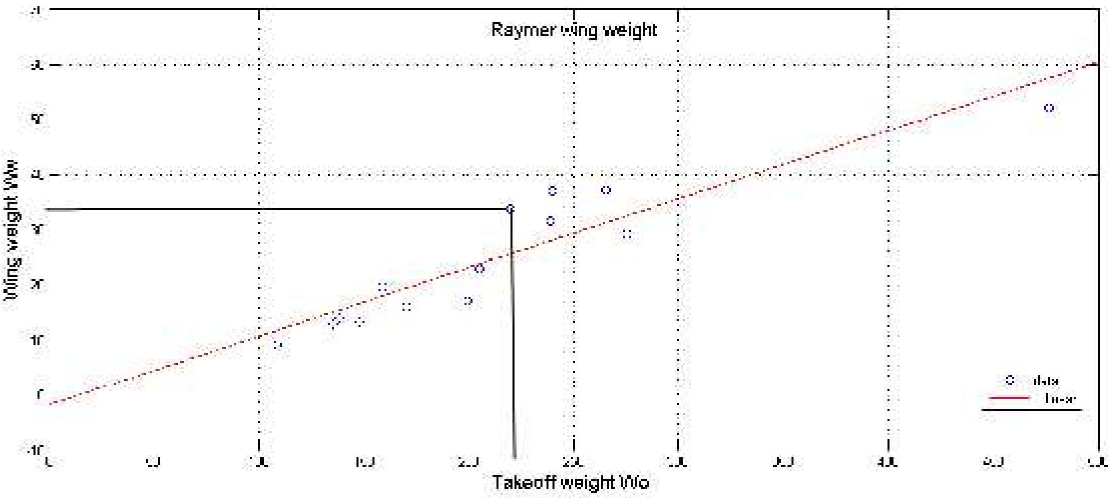


Figure (10-2) Raymer wing weight estimation

10.2.1 New formula for wing weight estimation for TUAV:

In this thesis we introduced a new formula to find wing weight estimation for TUAV's in conceptual design phase. This formula is derived from Gerard equation for sailplanes, but our formula shows more accurate results than Gerard for wing weight estimation.

This formula is experimented by choosing randomly fifteen aircrafts wings parameters. The results in Figure (10-3) show that the fitting line slope gives the same wing weight fraction at both ends of the slope line which equals 14.2% Ww/Wo along the slope line from 100 kg up to 500 kg takeoff weight.

$$W = 0.03 \times (N_z \times w_o)^{0.8} \times A^{0.2} \times S^{0.1} \times (1+\lambda)^{0.2} \times t^{0.1} ;$$

By substitution in the equation for parameters values that we got for our aircraft which weighs 220 kg, the result is as follow:

$$W = 0.03 \times (9 \times 220)^{0.8} \times 8^{0.2} \times 5.27^{0.1} \times (1+1)^{0.2} \times 0.15^{0.1}$$

$$W = 34.52\ k$$

This value represents 15.69 % of takeoff weight $\frac{W_w}{W_o}$, and 26.82 % of empty weight $\frac{W_w}{W_e}$.

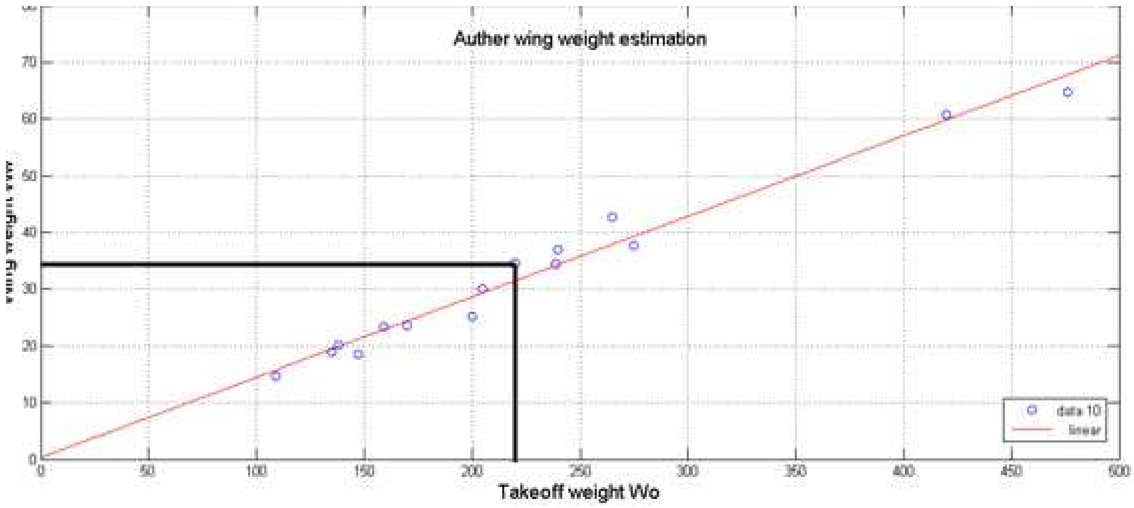


Figure (10-3) The new formula for wing weight estimation

10.3 Analysis of Fuselage weight results:

All the results extracted from the formulas for fuselage weight estimation are not satisfied. The best result for our case is gotten by Gundlach which equal to 39.33 Kg; this weight represents 17.88% of takeoff weight and 30.56% of empty weight. This attributed to that all the other equations are established for manned aircraft fuselage. The most obvious difference between the components of manned and unmanned aircrafts is that the fuselage shape and configuration. For UAV's there is no human inside it, so there is no need for crew or passengers' equipment, seats, and doors. Some parameters are neglected within fuselage weight design for UAVs such as; pressurization reinforcement weight, and doors weight. The fuselage diameter, depth, or width in UAV's is much lower than in manned aircrafts. All of these factors have clear effect on the UAV fuselage weight.

10.3.1 New formula for Fuselage weight estimation for TUAV:

Although Gundlach equation result for our case was almost satisfied. But it is still inaccurate for some aircrafts as shown in Figure (8-8). Random samples of some UAVs for fuselage weight estimation gave fuselage weights range from 12 to 21% of takeoff weight. This wide range makes it unreliable for many cases. Another thing is seen in Gundlach formula that is the diameter, height, or width of the fuselage is not announced in the formula, for example if we decided to increase the fuselage diameter by 25 % the weight will not increase and stay as it is which makes it unreliable and it may give fake results.

Introducing the speed of the aircraft in the equations of fuselage weight estimation calculations is not practically effective, especially for low speed aircrafts. At low speeds of aircrafts, the drag is reduced and doesn’t have high effect on the thrust. So introducing the aircraft speed in these formulas may not be effective.

In this thesis a new formula for fuselage weight estimation is established. This formula is founded for Tactical Unmanned aerial vehicles. It is valid for TUAVs having takeoff weight ranges between 100 – 500 Kg, and maximum speed less than 300 Km/hr, and altitude does not exceed 6000 m.

This formula is experimented just for the previous parameters. However it may be valid to use for higher or lower takeoff weights, and higher altitude, but in subsonic speeds range.

This equation first mainly relies on the aircraft takeoff weight, whereas the fuselage of the aircraft connects and carries all the components of the aircraft together, (wing, empennage, landing gear, and engine). Dimensions of the fuselage are a main part of the formula represented by fuselage length and diameter and indirectly fuselage surface area.

$$W_b = 0.55 \times (L^{0.3} \times D^{0.3} \times W_o^{0.4})^{1.5}$$

W_b- Fuselage weight. Kg

L – Fuselage length. m

D – Equivalent diameter. m

W_o – aircraft takeoff weight. Kg

$$W_b = 0.55 \times (5.16^{0.3} \times 0.4^{0.3} \times 220^{0.4})^{1.5}$$

$$W_b = 36.43\ kg$$

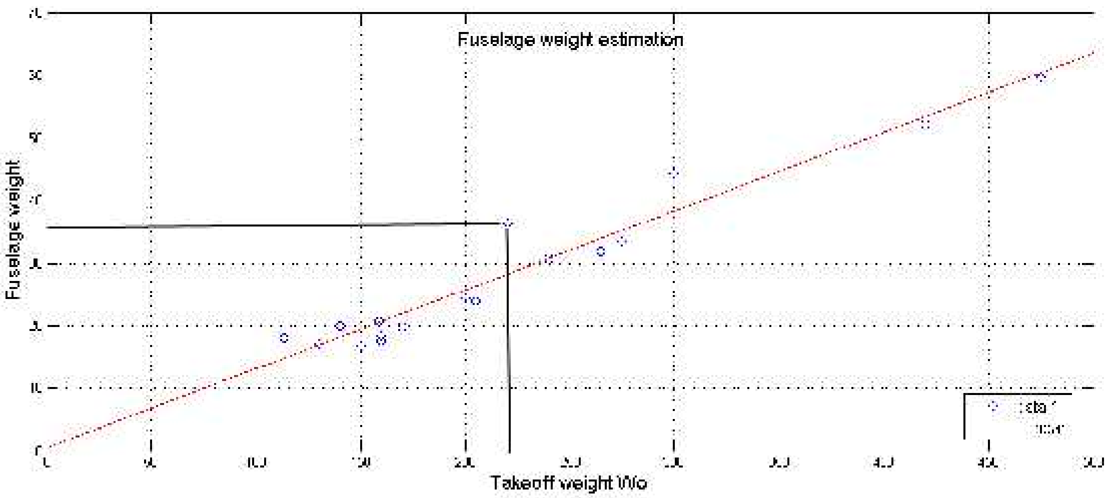


Figure (10-4) The new formula for fuselage weight estimation

From Figure (10-4), the results of this formula for different UAVs sizes have takeoff weights range from 100 – 500 Kg lying between 11 – 15 %.

Fitting line slope shows the same value all along the line which is approximately 12.7% of takeoff weight.

For our case, UAV with 220 Kg takeoff weight, the fuselage weight is 36.43 Kg which is 15.56% of the takeoff weight and 28.3% of the empty weight.

10.4 Empennage weight formulas selection:

For empennage weight estimation many formulas were used for finding the estimated empennage weight, and the results were explained on charts to get the best one. The formulas used for empennage weight mostly were founded for general aviation aircrafts and because of presence a lot of tail group shapes and designs. So in some cases we see unreasonable and extreme results for empennage weight estimation, but in some few cases we got acceptable results such as in Jay Gundlach, Torenbeek and Usaf formulas.

Empennage group has no specific design criteria, because some UAVs have special empennage shapes, some of them have no horizontal tail instead they have delta wing, in some cases the vertical tail is much bigger than the horizontal tail and in some case there are two vertical tails, …. Etc.

Jay Gundlach formula for empennage weight estimation:

$$W = w_a \times (S_h + S_v)\ \text{lb}, \quad w_a = 1.0\ lb/ft^2$$

$$W = 18.7\ lb$$

$$W = 8.5\ Kg$$

This weight represents 6.6% of the aircraft empty weight.

The empennage weight from Torenbeek is:

$$W_e = 0.04 \times (9 \times (5.7 + 13)^2)^{0.7}$$

$$W_e = 16.8\ lb = 7.63\ kg$$

This weight represents 5.93% of the aircraft empty weight.

The empennage weight from Usaf equal to 9.72 kg

This weight represents 7.55% of the aircraft empty weight.

10.4.1 New formula for empennage weight estimation:

These equations of our work for horizontal and vertical tails weight estimation are modified of Cessna equations. They should be applied to small size and low performance aircrafts which has maximum speed less than 350 km/hr.

10.4.2 Horizontal tail:

The equation from Cessna for horizontal tail weight estimation, originally established for general aviation aircrafts. The author equation (7-16) now is valid for UAV's.

For our case, Wo = 220 kg

$$W_h = \frac{1.46 \times W^{0.8} \times Sh^{0.1} \times A_h^{0.1}}{57.5 \times t_r^{0.2}}\ lb \quad (7\text{-}16)$$

$$W_h = \frac{1.46 \times 485^{0.8} \times 13^{0.1} \times 5^{0.1}}{57.5 \times 0.19^{0.2}}\ lb$$

$$W_h = 14.31\ lb = 6.5\ kg$$

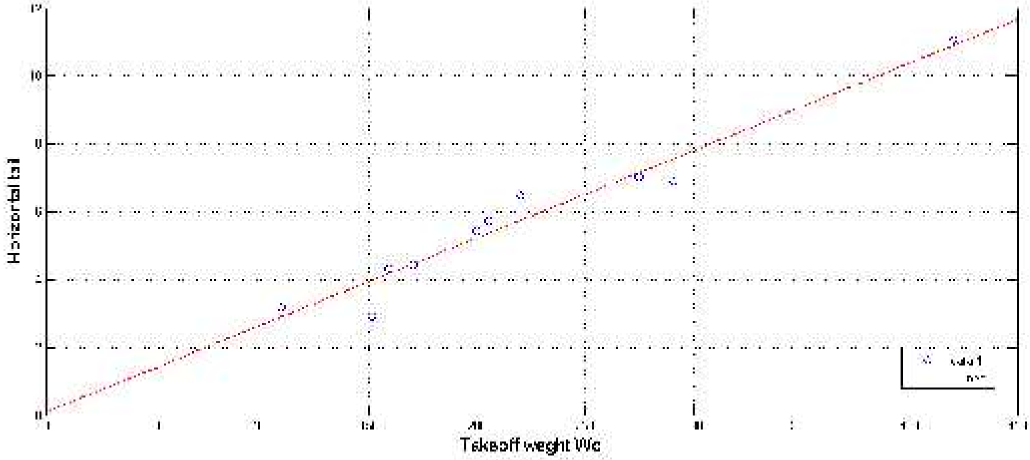


Figure (10-5) The new formula for horizontal tail weight estimation

10.4.3 Vertical tail:

The equation from Cessna for horizontal tail weight estimation, originally established for general aviation aircrafts. our equation (7-17) now is valid for UAV's.

$$W_v = \frac{0.039 \times W_0^{0.5} \times S_v^{1.2} \times A_v^{0.4}}{15.6 \times t_r^{0.7} \times \left(\cos \Lambda_{1/4}\right)^{0.8}} \ lb \qquad (7\text{-}17)$$

$$W_v = \frac{0.039 \times 485^{0.5} \times 5.7^{1.2} \times 1.4^{0.4}}{15.6 \times 0.24^{0.7} \times (\cos 20)^{0.8}}$$

$$W_v = 5.53 \ lb = 2.5 \ kg$$

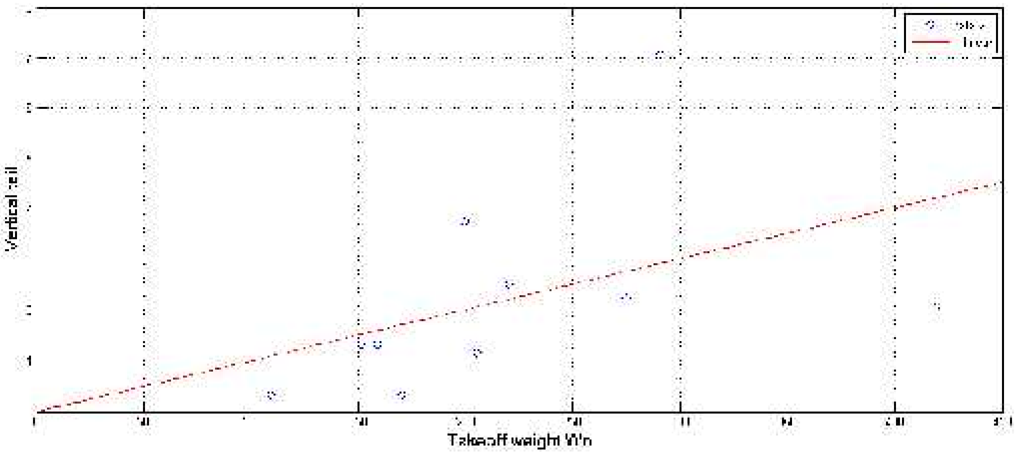


Figure (10-6) The new formula for vertical tail weight estimation

10.4.4 Empennage group weight estimation:

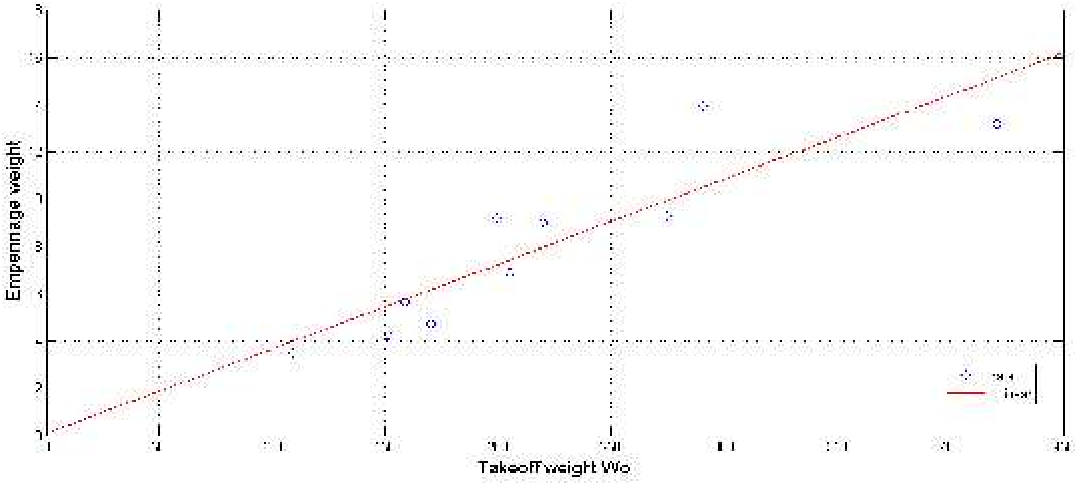


Figure (10-7) Empennage group

Summation of horizontal and vertical tails is shown of Figure (10-7)

Empennage weight estimation equals:

$$W_h + W_v = 6.5 + 2.5 = 9\ k$$

10.5 Landing gear formulas selection:

The estimated weight of the landing gear is much easier to calculate as the most formulas used in this part is directly related to the takeoff weight of the UAV, such as Howe, Kroo, Kundu, Nikolai, Pazmany, and Usaf.

The method that we followed in this thesis is Kundu formula:

$$W_g = 0.04 \times W_o - 0.1 \times (0.04 \times W_o)$$

$$= 7.9 \text{ Kg}$$

This weight represents 6.15% of empty weight and 3.6% of takeoff weight.

10.6 Engine selection:

Engine manufacturing companies offer a big number of engines which are designed for the aircrafts. There are many types of engines, such as 4-stroke engine,2-stroke engine, rotary engine. Each engine has its specifications for engine power, weight, specific fuel consumption, dimensions, and so on.

In our aircraft we will chose 2-stroke engine because of its low weight and high performance.

Table (10-1) two stroke engines specifications

Engine	type	Weight (kg)	Power (kw)
Radne raket 120	2-strok, 1-cylinder	6.8	10
Hirth F 36	2-strok, 1-cylinder	12.7	11
KFM 107	2-strok, 1-cylinder	15.2	19
Zanzottera Mz 34	2-strok, 1-cylinder	17	21
Jpx pul 425	2-strok, 2-cylinder	17	19
Hirth F 33	2-strok, 2-cylinder	20.4	21
Kawasaki	2-strok, 2-cylinder	22	28
Rotax 447	2-strok, 2-cylinder	27.8	29.5
2 si 460	2-strok, 2-cylinder	27	26
Rotax 227	2-strok, 2-cylinder	29.5	19
Hirth F 263	2-strok, 2-cylinder	31.8	22
Hirth 2702	2-strok, 2-cylinder	34.5	30

Rotax 447:

Rotax 447 will be chosen for our aircraft. The engine power is 29.5 kw and its maximum weight is 45.8 kg which represents 35.59% of the empty weight. (26)

10.6.1 The Rotax 447 engine specifications:

Table (10-2) Rotax 447 engine specifications

Performance	
Maximum power	39.6HP / 29.5kW @6500 RPM
Maximum torque	34.0ft-lb / 46NM @6000 RPM
Maximum RPM	6800 RPM
Combustion Chambers	
Bore	2.66" / 67.5mm
Stroke	2.40" / 61.0mm
Displacement	26.64cu.in. / 436.5cm³
Compression ratio	Theoretical: 9.6 Effective: 6.3
Weight	
Engine with carburetor	61.3lbs / 27.8Kg
Exhaust system	10.8lbs / 4.9Kg
Air filter	0.4lbs / 0.2Kg
No gearbox, no electric starter	72.5lbs / 32.9Kg
B gearbox, no electric starter	82.4lbs / 37.4Kg
B gearbox, electric starter	93.2lbs / 42.3Kg
C gearbox, no electric starter	90.1lbs / 40.9Kg
C gearbox, electric starter	100.9lbs / 45.8Kg
E gearbox	97.2lbs / 44.1Kg

10.7 The Final components weight estimation:

The main UAV structure components are the wing, fuselage, empennage, landing gear, and the engine. These components are the collection of the empty weight.

$$W_e = W_w + W_f \quad + W_e \quad + W_g + W_e$$

Sum of the UAV structure components weights is:

$$W_e = 34.52 + 36.43 + 9 + 7.9 + 45.8$$

$$W_e \simeq 134\ k$$

The calculated empty weight increased by approximately 5 Kg from that we found before.

Conclusion:

Design process for a complex system of the UAV passes through three design phases. First of these phases is that conceptual design phase. The result conducted from this phase is overall shape and size of the UAV. Preliminary design phase and detail design phase contribute to the results of conceptual design phase. If the results of conceptual design phase are not accurate enough it may result in bad UAV design and necessity to repeat the all phases. It is of outmost importance that conceptual design phase specifying the shape and size of the UAV as accurately as possible. Weight of the UAV and weight of the components of UAV are the most important input parameters in the complete design process, which influence all the other design parameters of the UAV. This thesis is devoted to accurate estimation of UAV takeoff weight and to accurately estimate the UAVs component weights.

Conceptual design process, as a first one in new flaying vehicle creation is strongly dependent on correct estimation of flaying vehicle takeoff weight. There are plenty data for manned aircrafts to estimate aircrafts takeoff weight as well as aircraft components weight. Databases for UAVs are still unavailable and those which are available are frequently incorrect and erroneous. It is required thus to analyze all available formulas for component weight estimation and to select or modify them to be applicable for UAV design process. This thesis is specially concentrated on tactical UAVs.

In this thesis available information about existing UAVs are collected and statistically processed in order to get basis for new component weight estimation formulas development. Many of statistical charts are presented together with data fitting. These charts can be used as a first guess of UAV design parameter definitions.

Various weight design formulas are applied to estimate component weights of UAVs. Since these formulas are developed for manned aircrafts (which are much heavier than tactical UAVs) they sometimes give unacceptable estimations (to high values or to small values than it could be expected by common sense). That's why critical considerations of available formulas have to be done, and formulas which gave unacceptable results are eliminated from our concern. Often we need to improve these formulas to better fit to UAV design process. In this thesis all available formulas are applied to tactical UAVs and selection among them is performed.

Finally modification in coefficients of the available formulas is done in order to get better fit UAV design process. These modified formulas are the main contribution of this thesis. Application of these formulas estimates more accurately component weight of the UAVs. This leads to more reliable and accurate conceptual design phase. Three suggested formulas for component weight estimation including wing, empennage, and fuselage are introduced:

For wing

$$W = 0.03 \times (N_z \times w_u)^{0.8} \times A^{0.2} \times S^{0.1} \times (1+\lambda)^{0.2} \times t_l^{0.1} \; K$$

For empennage

$$W_h = \frac{1.46 \times W^{0.8} \times Sh^{0.1} \times A_h^{0.1}}{57.5 \times t_r^{0.2}} \; li$$

$$W_v = \frac{0.039 \times W_u^{0.5} \times S_v^{1.2} \times A_v^{0.4}}{15.6 \times t_r^{0.7} \times \left(c \; \Lambda_{1/4}\right)^{0.8}} \; li$$

For fuselage

$$W_b = 0.55 \times (L^{0.3} \times D^{0.3} \times W_u^{0.4})^{1.5} \; Kg$$

Throughout the thesis all formulas are applied to typical tactical UAV. Very good agreement between component weight estimation and takeoff weight estimation is obtained, since summed component weights are very close to estimated total takeoff weight.

References:

1- Paul Jackson. Jane's All the World's Aircraft. 2004-2005.
2- Jay Gundlach. Designing Unmanned Aircraft Systems: A Comprehensive Approach. Published by the American Institute of Aeronautics and Astronautics, Inc. 2012.
3- Daniel P. Raymer. Aircraft Design: A conceptual Approach. Published by the American Institute of Aeronautics and Astronautics, Inc. Second Edition 1992.
4- Jan Roskam. Aircraft Design. Part 5. Component Weight Estimation. 1989.
5- Ajoy Kumar Kundu. Aircraft Design. Published in the United States of America by Cambridge University Press, New York. 2010.
6- Mohammad H. Sadraey. Aircraft Design: A system Engineering Approach. A john Wiley & Sons, Ltd Publications. 2012.
7- Denis Howe. Aircraft Conceptual Design Synthesis. Professional Engineering Publishing Limited London and Bury, UK. 2000.
8- Lloyd R. Jenkinson, Paul Simpkin, Darren Rhodes Civil. Jet Aircraft Design. Published by Arnold, a member of the Hodder Headline Group, 338 Euston Road, London NWl 3BH.
9- Ilan Kroo. Aircraft Design: Synthesis and Analysis. This textbook is copyright by Desktop Aeronautics, Inc. 2001.
10- Darrol Stinton. The Design of the Aeroplane. Printed and bound in Great Britain by William Clowes Limited, Beccles and London. 1991.
11- Kendra LB Cook. The silent force multiplier: the history and role of UAVs in warfare. In 2007 IEEE Aerospace Conference.
12- Vlad Paul Galantai. Design and Analysis of Morphing Wing for Unmanned Aerial Vehicles. Department of Mechanical and Industrial Engineering, University of Toronto 2010.
13- Marasco.A.J. Control of Cooperative and Collaborative Team Tactics in Autonomous Unmanned Aerial Vehicles using Decentralized Model Predictive Control. Master's Thesis, Electrical Engineering, Royal Military College- Canada,2012.
14- Time line of uavs http://www.pbs.org/wgbh/nova/spiesfly/uavs.html .

15- E. Bone and C. Bolkcom. Unmanned aerial vehicles: Background and issues for congress. Defense Technical Information Center (DTIC) Document, 2003.

16- D.M. Marshall. Dull, dirty, and dangerous: The FAA's regulatory authority over unmanned aircraft operations. Issues in Aviation Law & Policy, page 10085, 2004.

17- R. Kumar. Tactical reconnaissance: UAVs versus manned aircraft. Technical report, Defense Technical Information Center (DTIC) Document, 1997.

18- Teng Choon Hon Adrian. Design and Performance Evaluation Study of a Prototype of A tactical Unmanned aerial vehicle. December 2007

19- Ahmed Tajmour Hafez. Design and Implementation of Modern Control Algorithms for Unmanned aerial vehicles.

20- https://en.wikipedia.org/wiki/List_of_unmanned_aerial_vehicles .

21- Toby Clark, Peta Johannsen , Zhao Lu, Miteshu Ramnarain. Maritime Surveillance UAV, Final Report. UNIVERSITY OF ADELAIDE. 2012.

22- Egbert Torenbeek. synthesis of subsonic airplane design. Torenbeek.1976

23- http://www.oceanica.ufrj.br/ocean/cursosead/materiaiscompositos/compositematerials/f_aerospace_applications.pdf

24- Al-Shamma, Omran, and Ali Dr. Rashid. Aircraft Weight Estimation In Interactive Design Process, 72nd Annual Conference, St. Louis, Missouri, 2013.

25- Gerard, W. H. “Prediction of Sailplane Wing Weight for Preliminary Design,” Weight Engineering, A98 37633, Spring 1998, p. 9.

26- http://www.rotaxservice.com/rotax_engines/rotax_447ULs.htm.

27- K. L. Sahoo, C. S. Sivaramakrishnan, A. K. Chakrabarti. (2000). Some studies on Al-8.3Fe-0.8V-0.9Si Alloy. Journal of Material Processing Technology 135 (2003) 253-257.

28- A. Essari, F. Sariuglo, Z.Petrovic, A.Sedmac, I. Samardzic. Effect of Aging on Mechanical Properties of Al-8Si-8Fe-1.4V/SiCp Composites. ISSN 0543-5846, volume 55(2) 189-192 (2016).

Потписани Абдулхаким Ессари

број уписа D 27/08

Изјављујем

да је докторска дисертација под насловом

ПРОЦЕНА ТЕЖИНА КОМПОНЕНТИ ТАКТИЧКИХ БЕСПИЛОТНИХ ЛЕТЕЛИЦА У ФАЗИ КОНЦЕПТА

- резултат сопственог истраживачког рада,
- да предложена дисертација у целини ни у деловима није била предложена за добијање било које дипломе према студијским програмима других високошколских установа,
- да су резултати коректно наведени и
- да нисам кршио/ла ауторска права и користио интелектуалну својину других лица.

Потпис докторанда

У Београду, ___________________

Прилог 2.

Изјава о коришћењу

Овлашћујем Универзитетску библиотеку „Светозар Марковић“ да у Дигитални репозиторијум Универзитета у Београду унесе моју докторску дисертацију под насловом:

ESTIMATION OF COMPONENT DESIGNWEIGHTS IN CONCEPTUAL DESIGN PHASE FOR TACTICAL UAVs

која је моје ауторско дело.

Дисертацију са свим прилозима предао/ла сам у електронском формату погодном за трајно архивирање.

Моју докторску дисертацију похрањену у Дигитални репозиторијум Универзитета у Београду могу да користе сви који поштују одредбе садржане у одабраном типу лиценце Креативне заједнице (Creative Commons) за коју сам се одлучио/ла.

1. Ауторство
2. Ауторство – некомерцијално
3. Ауторство – некомерцијално – без прераде
4. Ауторство – некомерцијално – делити под истим условима
5. Ауторство – без прераде
6. Ауторство – делити под истим условима

(Молимо да заокружите само једну од шест понуђених лиценци, кратак опис лиценци дат је на полеђини листа).

Потпис докторанда

У Београду, ________________________

1. Ауторство - Дозвољавате умножавање, дистрибуцију и јавно саопштавање дела, и прераде, ако се наведе име аутора на начин одређен од стране аутора или даваоца лиценце, чак и у комерцијалне сврхе. Ово је најслободнија од свих лиценци.

2. Ауторство – некомерцијално. Дозвољавате умножавање, дистрибуцију и јавно саопштавање дела, и прераде, ако се наведе име аутора на начин одређен од стране аутора или даваоца лиценце. Ова лиценца не дозвољава комерцијалну употребу дела.

3. Ауторство - некомерцијално – без прераде. Дозвољавате умножавање, дистрибуцију и јавно саопштавање дела, без промена, преобликовања или употребе дела у свом делу, ако се наведе име аутора на начин одређен од стране аутора или даваоца лиценце. Ова лиценца не дозвољава комерцијалну употребу дела. У односу на све остале лиценце, овом лиценцом се ограничава највећи обим права коришћења дела.

4. Ауторство - некомерцијално – делити под истим условима. Дозвољавате умножавање, дистрибуцију и јавно саопштавање дела, и прераде, ако се наведе име аутора на начин одређен од стране аутора или даваоца лиценце и ако се прерада дистрибуира под истом или сличном лиценцом. Ова лиценца не дозвољава комерцијалну употребу дела и прерада.

5. Ауторство – без прераде. Дозвољавате умножавање, дистрибуцију и јавно саопштавање дела, без промена, преобликовања или употребе дела у свом делу, ако се наведе име аутора на начин одређен од стране аутора или даваоца лиценце. Ова лиценца дозвољава комерцијалну употребу дела.

6. Ауторство - делити под истим условима. Дозвољавате умножавање, дистрибуцију и јавно саопштавање дела, и прераде, ако се наведе име аутора на начин одређен од стране аутора или даваоца лиценце и ако се прерада дистрибуира под истом или сличном лиценцом. Ова лиценца дозвољава комерцијалну употребу дела и прерада. Слична је софтверским лиценцама, односно лиценцама отвореног кода.

Изјава о истоветности штампане и електронске верзије докторског рада

Име и презиме аутора **M.Sc. Abdulhakim Essari**

Број уписа D 27/08

Студијски програм **Докторске студије**

Наслов рада **ESTIMATION OF COMPONENT DESIGN WEIGHTS IN CONCEPTUAL DESIGN PHASE FOR TACTICAL UAVs**

Ментор проф. **др Златко Петровић, дипл. инж.**

Потписани **M.Sc. Abdulhakim Essari**

изјављујем да је штампана верзија мог докторског рада истоветна електронској верзији коју сам предао/ла за објављивање на порталу **Дигиталног репозиторијума Универзитета у Београду.**

Дозвољавам да се објаве моји лични подаци везани за добијање академског звања доктора наука, као што су име и презиме, година и место рођења и датум одбране рада.

Ови лични подаци могу се објавити на мрежним страницама дигиталне библиотеке, у електронском каталогу и у публикацијама Универзитета у

Потпис докторанда

У Београду, ________________________