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AIRCRAFT CONCEPTUAL DESIGN USING MULTI-OBJECTIVE OPTIMISATION.

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Abstract

Once the market research has been completed and requirements for a new aircraft have been define, the conceptual design phase is launched. During the conceptual design phase, designers have the goal of defining and investigating a number of alternative solutions for which optimisation tool are seldom used. Now a day, design is not limited to single objective optimisation but expanding to multi-objective optimisation. In the aircraft conceptual design using optimisation, requires modal from the various disciplines such as, structure, aerodynamics, propulsion and control which leads to the multidisciplinary design. In this particular study, aircraft which has short range, low altitude and short take-off and landing are considered for which Structure modal, weight modal and aerodynamic modal are prepared for the optimisation purpose. In addition, complete mathematical modal are prepared, which represent the final objective statement, constrain, and fixed parameter and design variables which is used for the design of aircraft used for fire-fighting or agriculture spraying. For study, all the basic terms which are related to aircraft design are briefly discussed. Main objective of the study is to understand conceptual design of the aircraft with the use of multi-objective and multi disciplinary optimisation. During the conceptual design, trade-off between multi disciplinary and multi-objectives is occurred such as aircraft wing loading, wing aspect ratio, payload, take-off and landing distance. With the use of direct search domain algorithm various design objective are optimise simultaneously. Maximise the payload and minimise the take-off distance, minimise aircraft wing weight, minimise the wing aspect ratio and minimise the wing loading are consider as a multi-objective optimisation in further design. In the aircraft conceptual design, wing is the most important part of the aircraft design which is related to multidisciplinary like structure, aerodynamics and control. That's why in the study, wing is the main design variable. Finally, we are designing the aircraft at conceptual level using multi-objective optimisation and compare the value with each set of multi-objective optimisation result.

Keywords— Aircraft conceptual design, Multi objective optimization, Multi disciplinary optimization, Pareto solution, Pareto set, direct search domain.

I. INTRODUCTION

The aircraft design process is divided in to several stages. It starts from market research, followed by conceptual design, preliminary design, detailed design, and product support. Each and every stage is most important in design. However, the conceptual phase is one of the most important parts of the design process. Aircraft conceptual design is the process of determining the new concept and configuration which satisfied the most of the market requirement in which not only the overall shape, size, weight, and performance of the new aircraft as well as engine size and placement, the fundamental aspect of the shape of the wing, lifting surface, location of the main wing, the size, shape and location of the horizontal and vertical tail wing are determined [3],[7].

The overall conceptual design is describe with seven intellectual pivot points. In the aircraft conceptual design using optimization require modal from the various disciplines such as, structure, aerodynamics, propulsion and control. In this case, the approach is called multidisciplinary design optimization. Moreover, in most of the aircraft conceptual design involves trading multiple performance metrics or objectives. In this study represent the multi-objective and multidisciplinary optimization of the short range, low altitude, short take-off and landing distance aircraft. The main function of the aircraft is fire fighting or agriculture spraying. In this way it is expected that resulting design would be optimal to perform its intended function efficiently and effectively. In this study various models are employs to conduct the multi-objective optimization. Any design requires multi-objective optimization during their conceptual phase. For this purpose direct search domain algorithm is used in this particular study. Multi-objective optimization problems in aviation industry are extremely hard to solve in a general manner. Due to the high level integration of aircraft's systems and the increase in complexity of analysis and design methods for the evaluation of system performance, such problems are characterized by multi-disciplinary simulations, objective function and constraints that are expensive to evaluate. In addition, they often have large-dimensional design parameter spaces that further complicate the solution of the problem. Moreover, the resulting optimization problems are frequently non-convex, i.e., multimodal and ill-conditioned. Making complete optimizations of complex aircraft configurations are very expensive [8].

With the help of the multi objective optimization, desire capabilities of the design can achieve at a minimum cost. Aircraft are incredibly expensive compared to any other man-made item. Aircraft are also directly concern with the human life. Multi-

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objective optimization is the process of optimizing more than one objective simultaneously [4,5]. And in the multi disciplinary design and optimization is the process of optimizing a complex system, taking in to account concurrently all the contributing disciplines. In most of the problems to which multi disciplinary optimization is applied, the considered complex system contain internally coupled disciplines.In our particular study we are trying to maximize the payload of the aircraft as well as minimize the take-off distance. In addition, important trade-off between the wing loading and the wing aspect ratio are taking in account during the conceptual design. In this study direct search domain algorithm is used for the multi-objective optimization. [2,13]

II. MULTI-OBJECTIVE OPTIMIZATION OF THE AIRCRAFT USING THE DSD ALGORITHM

A PROBLEM FORMULATION

In this particular study, direct search domain algorithm is used for the multi objective optimization. It is used for the" n" number of objective. But in this particular study bi-objective and three objectives optimization are considered. The design problem is considered in two sets of optimization.

A. First multi-objective optimization problem
1. Maximize the payload.
2. Minimize the take off distance.

B. Second multi-objective optimization problem
1. Maximize the payload.
2. Minimize the wing loading.
3. Minimize the wing aspect ratio.

The main design objective is to maximize the payload (water capacity of tank) of the aircraft. The maximum take-off gross weight of the aircraft is 5700 Kg with an engine power of 1000 horsepower. Design requirement of the aircraft is short range, low altitude, short take-off and landing. The design mission has ten mission segment; warm-up-taxi, takeoff, climb, cruise, descent, loiter, climb, cruise, descent and landing.

Pay load weight selected as the main objective function, which was repeat in all the cases as first objective function. Payload weight equals to the maximum take-off weight minus the fuel weight, dependent weight and fixed weight. According to above formula if we minimize the fuel weight and dependent weight, accordingly, we can maximize the payload weight. The fuel weight represents the yearly recurring costs of aircraft operation. And the aircraft dependent weight represents the over head cost of the aircraft. Thus, an aircraft has a lower fuel weight and dependent weight, may also mean an aircraft with a lower operational cost as well as higher payload weight [10,11].

Take-off distance of the aircraft is depends on the wing loading, thrust to weight ratio, maximum lift co-efficient and take-off parameter. In addition, landing distance is depends on the maximum lift co-efficient and the wing loading.

In our particular study, aircraft main function is agriculture spraying or fire fighting. Because of the purpose of the aircraft, aircraft has to be flying at low altitude, short range aircraft, short takeoff and landing. In addition, aircraft is flying under the higher variable loading condition. According to function of the aircraft wing loading is high but designer try to minimize the wing loading. Furthermore, design aircraft is low altitude aircraft. Because of low altitude requirement and variable loading condition, aspect ratio of the wing should be minimum [7,8,9].

B CONSTRAIN:-

According to market requirement, in this particular study, multi-objective constrain optimization are considered. Design constraints are affected the optimization process[3].

Design constraints are as follows.

1) Maximum cruise speed: < 70 m/s

Maximum cruise speed of the aircraft should not greater than 70 m/s

Thus,

$$G1= [\frac{2}{\varsigma}\frac{W}{S}]^{0.5} [\frac{k}{CDo}]^{0.25} -70$$

2) Wing area < 70 m^2

Wing area of the aircraft should not greater than 70 m^2

Thus,

G2= x(1)*x(2)-70
Where, x(1) = wing span\
x(2)= wing chord

3) Range > 500000 m

Range of the aircraft should be more than 500000 m Thus,

$$G3=V/C_{fs}* L_{OD\text{-}cruise} * \ln (W_i/W_f)- 500000$$

4) Aspect ratio > 5 Aspect ratio of the aircraft should be more than 5 Thus,
G4 = 5-x(1)/x(2)

5) Maximum altitude < 21000 ft

Maximum altitude of the aircraft should not more than 21000 ft For the further optimization process and preparation of the mathematical modal following parameter is considered as fixed parameter [3,6]

.

Table 1. Fixed Parameter [3],[6]

Parameter	**Value**
W or Mo = Gross aircraft weight	5700 kg
σ = ratio of air density at take-off field to sea level	1
Number of Engines	1
Cruise Mach Number	0.208
Maximum Altitude	21000 ft
Total range of aircraft	600 km
Wing sweep (Λ)	0
λ = Ratio of tip to centreline	1
N = 1.65 times the limit maximum manoeuvre acceleration factor	1.65
e = Oswald efficiency factor	0.8
t/c thickness to chord ratio	0.2
T/W=thrust to weight ratio	0.60
Cl max	1.6 to 2.2 = 1.6
Lift to drag ratio $L_{OD\text{-}cruise}$	14.2
E= endurance	1200 sec
C_1 = coefficient depending on type of aircraft	C_1 = 0.00183
C_2 = Coefficient depending upon aircraft	0.6
C_3 = coefficient depending upon type of aircraft and engine	1.40
C_4 = coefficient depending upon type of aircraft	0.12
V (cruise velocity)	70 m/s
M_{eng} = Weight of one engine	567kg (1250 lb)
L = Overall fuselage length	9.47 m
B = Maximum width of the fuselage	3.7m
H = Maximum height of fuselage	3.7m
d=Fuselage diameter	2.8 m

C The Weight Model of the Aircraft –

In the weight modal of the aircraft we are calculating the weight of the each part of the aircraft.

$$W_O = W_{payload} + W_{Fuel} + W_{fixed} + W_{dep}$$

Where,

W_O= Gross aircraft weight during takeoff = 5700 KG
$W_{payload}$ = Payload weight
W_{fixed}= Number of fixed weights (Fuselage, fins, engine weight)
W_{dep}= various dependant weights (wing structural weights, lift surface Weight, Landing gear weights etc.)

In our design gross weight of the aircraft is fixed parameter =5700 KG.
According to gross weight of aircraft we are calculating the each aircraft's component weight [1].

D Payload weight:

$\mathbf{W_{payload}}$ = Objective (Maximise the payload of the aircraft means water tank capacity)
Gross Aircraft weight: $\mathbf{W_O}$ =5700 KG

E Fuel weight:-

Fuel weight**:** - Fuel weight is depends upon the fuel consumptions. Fuel consumption depends upon the mission profile [3].

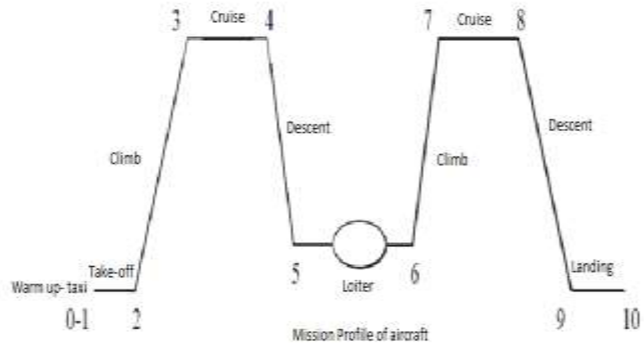


Figure 1. Aircraft mission profile[3]

0→1: Warm-up-Taxi-
1→2: Take off
2→3: Climb
3→4: Cruise
4→5: Descent
5→6: Loiter (Agricultural Spraying or fire fighting)
6→7: Climb
7→8: Cruise
8→9: Descent
9→10: Landing

Idealized mission profile is divided in to ten segments Wf = weight of the fuel required for the mission plus reserve, trapped and emergency flight fuel. Fuel weight can be estimated using fuel fraction [3].

Fuel fraction means above each segment of the mission profile is associated with the weight fraction which can be expressed as the aircraft weight at end of segment divided by its weight at the beginning of that segment. First these fractions are estimated for each segment then they are multiply together to find the total mission weight fraction. 6 % allowance for reserve, trapped and emergency flight fuel are considered. Gross weight of the aircraft is 5700 kg.
Total fuel fraction $(W_f / W_0) = 1.06 * (1 - W_{10} / W_0)$

$$(W_{10} / W_0) = (W_1 / W_0) * (W_2 / W_1) * (W_3 / W_2) * (W_4 / W_3)$$

$* (W_5 / W_4) * (W_6 / W_5) * (W_7 / W_6) * (W_8 / W_7) * (W_9 / W_8) * (W_{10} / W_9)$

E1 Warm up and taxi segment weight fraction = Weight fraction are based on the historical data Initial estimation of the warm-up and taxi weight fraction is as follow.

$(W_1 / W_0) = 0.97*0.995=0.97$

E2 Take off segment weight fraction =Weight fraction is based on the historical data Segment 1-2

$(W_2 / W_1) = 0.996$

E3 Climb segment weight fraction= Weight fraction is based on the historical data. Segment 2-3 and segment 6-7.

$(W_3 / W_2) = (W_7 / W_6) = 0.985$

E4 Cruise segment weight fraction =for cruise, (segment 3-4 and segment 7-8) Fuel fraction for the cruise segment is calculated by brequet range formula [3].

$R_B = (V/C_{fs})*L_{OD\text{-}cruise} * \ln(W_i/W_f)$

$W_4/W_3 = e^{-(c*R_{3\text{-}4})/V*(L/D)}$

$W_8/W_7 = e^{-(c*R_{7\text{-}8})/V*(L/D)}$

Where, range R = 600 km and cruise velocity 70 m/s are known from the market requirements. The specific fuel consumption C is taken from the engine data.
C= TSFC= 1.303 *E-7 kg/N.m
C= 469 g/(kW•h) (0.77 lb/(hp•h))

TSFC=469*(E-3)/(1000*3600) (kg/w*s)
=1.303*E-7 (kg/N.m)
(watt=m^2kg/s^3)
TSFC= 0.6(1/hr)=1.666*E-4(1/s)

This requires an estimation of the lift-to-drag ratio $L_{OD\text{-}cruise}$ .In conceptual design phase a detailed aerodynamic analysis is not necessary since, the shape is not laid out at this stage. However an approximate value was based on data for single propeller driven aircraft.
$L_{OD\text{-}cruise}$ =Maximum Lift over Drag ratio during cruise phase
= 9.6– 14.2 (in-between)
= 14.2
V= cruise velocity = 70 m/s
$R_{3\text{-}4}$=300 km
$R_{7\text{-}8}$=300 km

Total range R=600 km.
$W_4/W_3 = e^{-(c*R_{3\text{-}4})/V*(L/D)} = 0.951$

$W_8/W_7 = e^{-(c*R_{7\text{-}8})/V*(L/D)} = 0.951$

E5 Descent segment weight fraction:-

Weight fraction is based on the historical data Segment 4-5 and segment 8-9

$(W_5 / W_4) = (W_9 / W_8) = 1$

E6 Loiter (fire fighting or agriculture spraying segment):-

Weight fraction is calculated by endurance equation. Segment 5-6.

$E = V/(C_{fs}*v)* L_{OD\text{-}loiter} * \ln(W_i/W_f)$

E = loiter time (agriculture spraying or fire fighting purpose 20 min = 1200 sec)

TSFC=469*(E-3)/(1000*3600) (kg/w*s)
=1.303*E-7 (kg/N.m)
(watt=m^2kg/s^3)
TSFC= 0.6(1/hr)=1.666*E-4(1/s)

$L_{OD\text{-}cruise}$ =Maximum Lift over Drag ratio during cruise phase
= 9.6– 14.2 (in-between)
= 14.2

$(W_6 / W_5) = e^{-E*c/(L/D)} = 0.99$

E7 Landing weight fraction estimation:-

Weight fraction is based on the historical data. Segment 9-10

$(W_{10} / W_9) = 0.998$

Thus,

$(W_{10} / W_0) = (W_1 / W_0) * (W_2 / W_1) * (W_3 / W_2) * (W_4 / W_3) * (W_5 / W_4) * (W_6 / W_5) * (W_7 / W_6) * (W_8 / W_7) * (W_9 / W_8) * (W_{10} / W_9)$

$W_{10}/W_0 = 0.97*0.996*0.985*0.951*1*0.99*0.985*0.951*1*0.998$

$W_{10}/W_0 = 0.838$

$W_F/W_0 = 1.06 * (1 - W_{10}/W_0)$
= 1.06* 0.162
=0.171
=0.171 * 5700
=974.7 kg
Total aircraft fuel weight is 974.7 kg.

F *Fixed Weight:*

Fixed weight of the aircraft includes all the weight which is not affecting the design variable. In our particular study the wing span and wing chord are taken as design variable. So weight of the engine, weight of the fuselage and system weight are considered as the fixed weight of the aircraft. Formula for the fixed weight of the aircraft is below [3].

$\mathbf{W_{fixed}} = W_{engine} + W_{fuselage} + W_{sys}$

F1 Weight of the aircraft system W_{sys}:-

Aircraft system includes the all the equipment weight, landing gear weight, passenger furnishing and all the electronics indicators.

$\mathbf{W_{sys}} = C_4 M_0$
$= 0.12*5700 =$
684 kg

Where,
M_0 = Total aircraft mass = 5700 kg
C_4 = coefficient depending upon type of aircraft
For general aviation, single propeller engine types, (C_4 = 0.12)

F2 Weight of the engine W_{engine}:-

Weight of the engines includes all the engine which are installed in the aircraft as well as housing weight of the engine and propeller weight.
In our particular study single propeller driven nine piston radial engine is used. Weight of the engines is includes the propeller weight.

$\mathbf{W_{engine}} = nC_3 M_{eng}$
$= (1*1.40*567) =$
793.8 kg

Where,
n = Number of engines=single engine =1
M_{eng} = Weight of one engine
= 567kg (1250 lb)
C_3 = coefficient depending upon type of aircraft and engine
For general aviation, single piston - engine types
C_3 = 1.40

F3 Weight of the fuselage $W_{fuselage}$:-

Weight of the fuselage includes the compartment cell and the cockpit as well as fuel tank and hopper tank. The formula for the fuselage weight is given below [3].

$\mathbf{W_{fuselage}} = C_2\,[L(B + H)\ V_D^{0.5}]^{1.5}$

Where,
C_2 = Coefficient depending upon type of aircraft
=For single propeller driven engine and light aircraft
= 0.6
L = Overall fuselage length =9.47 m
B = Maximum width of the fuselage = 3.7m
H = Maximum height of fuselage
=3.7m
V_D= design max. Speed
=70 m/s
Fuselage diameter =2.8 m

$\mathbf{W_{fuselage}} = C_2\,[L(B + H)\ V_D^{0.5}]^{1.5}$
$=0.06[9.87(3.7+3.7)*8.37]$
$=906.90$ Kg

$\mathbf{W_{fixed}} = W_{engine} + W_{fuselage} + W_{sys} =$
793.8+ 906.90+684
= 2384.7 kg

G Dependent Weight:

Dependent weight includes the lifting surfaces weight. Main wing is responsible for the producing the lift forces. Rectangular shape unswept wing are used in the design of the aircraft. Because of the unswept wing sweep angle is zero as well as taper ratio is 1 because of the rectangular wing shape[3].

$\mathbf{W_{dep}} = W_{liftsurfaces}$

$\mathbf{W_{liftsurfaces}}$ is total weight of the wing and the horizontal and vertical stabilisers.

$\mathbf{W_{liftsurfaces}}=$
$C_1[A^{0.5}S^{1.5}\sec\Lambda_E(1+2\lambda/3+3\lambda)*M_0/S*N^{0.3}*(V_D/(t/c))^{0.5}]^{0.9}$ kg

Where,
M_0 = Total aircraft mass= 5700 kg
A = Wing aspect ratio= wing span / wing chord
S = Wing area = variable in m^2
Λ_E = Effective sweep, usually at the 0.25 chord sweep
=0 deg (upswept wing are used)
λ = Ratio of tip to centreline chords of the wing
= chord length at tip/ chord length at root
=1
N= 1.65 times the limit maximum manoeuvre acceleration factor
V_D = Design maximum (diving) speed, m/s= 70 m/s
t/c = thickness to chord ratio at wing centreline =0.2
= Naca 4416 airfoil are used in the wing design.
C_1 = coefficient depending on type of aircraft
For General aviation purpose aircraft, Single propeller driven engine
(Gross aircraft weight M_0 < 5700 kg)
C_1 = 0.00183

$\mathbf{W_{liftsurfaces}}=$
$C_1[A^{0.5}S^{1.5}\sec\Lambda_E(1+2\lambda/3+3\lambda)*M_0/S*N^{0.3}*(V_D/(t/c))^{0.5}]^{0.9}$ kg

$\mathbf{W_{liftsurfaces}}$
$=0.00183[(B/C)^{0.5}*(B*C)^{1.5}\sec\Lambda_E(1+2\lambda/3+3\lambda)$
$*M_0/S*N^{0.3}*(V_D/(t/c))^{0.5}]^{0.9}$

$\mathbf{W_{dep}} = 0.00183[(B/C)^{0.5}*(B*C)^{1.5}\sec\Lambda_E(2850/B*C)$
$*1.162*(70/(t/c))^{0.5}]^{0.9}$

Weight modal of the aircraft

$W_O = W_{payload} + W_{Fuel} + W_{fixed} + W_{dep}$
Where,
W_O= Gross aircraft weight = 5700 kg
$W_{payload}$ = Design objective 1
=Maximise the payload.
W_{Fuel} =974.7 kg
W_{fixed} =2384.7 kg
$\mathbf{W_{dep}} = 0.00183[(B/C)^{0.5}*(B*C)^{1.5}\sec\Lambda_E(2850/B*C) *1.162 *(70/(t/c))^{0.5}]^{0.9}$

Final statement of the design objective - 1 (payload)

$W_{payload} = W_O - W_F - W_{fixed} - W_{dep}$

$W_{payload} = 5700-974.7-2384.7-(0.00183[(B/C)^{0.5}*(B*C)^{1.5}\sec\Lambda_E(2850/B*C) *1.162 *(70/(t/c))^{0.5}]^{0.9})$

$W_{payload} = 2340.6-(0.00183[(B/C)^{0.5}*(B*C)^{1.5}\sec\Lambda_E(2850/B*C) *1.162 *(70/(t/c))^{0.5}]^{0.9})$

Variable
B= wing span
=x (1) – variable
C= Wing chord
= x (2) – variable

H Take off distance:-

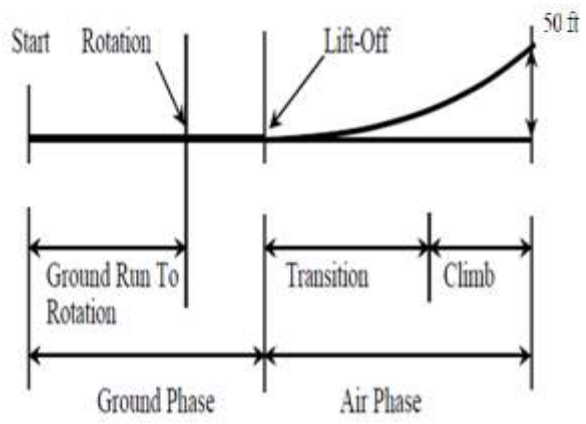


Figure 2. Aircraft take-off field [3]

Fire fighting or Agricultural aircraft is usually based at a temporary airstrip. And it is not as good as other commercial aircraft"s airstrip. Take-off length is also depends on the nature of the take-off surface. Because of rough surface the rolling resistance of the wheels causes the take-off run to increase. That"s why minimizing the take-off distance is very important objective in this design study [6].
The evaluation of takeoff performance can be examined in two phases, the ground and air phase. The ground phase begins at brake release, includes rotation, and terminates when the aircraft becomes airborne. The air phase is begins from leaving the ground until reaching an altitude of 50 ft.[3]

However, in the conceptual design, detail study of take-off distance is not considered. According to empirical formula of take-off distance, take-off distance is depends upon take-off parameter. During the conceptual design, take-off distance is calculated by using the wing loading and the thrust to weight ratio. At this stage we are using the take-off parameter and basic terms to finding the take-off distance. For this, we use historical data and take-off parameter, TOP, which has been found to correlate the take-off distance for wide range of aircraft.[3]

The take-off parameter is define as

$TOP=(W/S)_{TO}*(1/CL_{max})* (W/T)_{TO}*(1/\sigma)$

TOP= Take off parameter
(T/W)= Thrust to weight ratio
(W/S)= Wing loading
=Gross aircraft weight / wing area
CL_{max}=Maximum lift coefficient
σ =Ratio of the air density at the take off site to that at sea level.

In our particular study, for the short range aircraft, the value for the take off parameter is as below.

$TOP=(W/S)_{TO}*(1/CL_{max})* (W/T)_{TO}*(1/\sigma)$
TOP= Take off parameter
(T/W)= Thrust to weight ratio = 0.60
(W/S)= Wing loading
=Gross aircraft weight / wing area
CL_{max}=Maximum lift coefficient=1.6
σ =Ratio of the air density at the take off site to that at sea level.=1
Thus, take-off distance of the aircraft is as below,

$S_{To}= 20.9(TOP)+87\sqrt{(TOP)(\frac{T}{W})}$

Objective 2:- Minimize the take off distance

$S_{To}= 20.9(TOP)+87\sqrt{(TOP)(\frac{T}{W})}$
Where,
The first and second coefficients have units of $f/(\frac{lb}{f^2})$ and

$f/(\frac{lb}{f^2})^{0.5}$, respectively

$20.9\ f/(\frac{lb}{f^2}) = 1.305\ m/(\frac{kg}{m^2})$

$87\ {}^{f}\!/_{(\frac{lb}{f^2})} = 12\ {}^{m}\!/_{(\frac{kg}{m^2})^{0.5}}$

After the converting all coefficient unit in to the SI unit the final equation for the take off distance is as below.

$TOP=(W/S)_{TO}*(1/CL_{max})*(W/T)_{TO}*(1/\sigma)$

$S_{To}= 1.305(TOP)+12\sqrt{(TOP)(\frac{T}{W})}$

S_{To} = take-off distance.
TOP= Take off parameter
(T/W)= Thrust to weight ratio
= 0.60 (according to Thomas crock page no 55 (table number 3.2)

(W/S)=Wing loading

= Gross aircraft weight / wing area

$= {}^{5700}\!/_{(wing\ span\ *\ wing\ chord)}$

CL_{max}=Maximum lift coefficient=1.6
σ =Ratio of the air density at the take off site to that at sea level.=1

I Aircraft wing loading:-

Wing loading is depends on the total aircraft weight as well as wing geometry and wing area.

Wing loading =W/S
Where,
w = aircraft gross weight = 5700 Kg
S = wing area = B*C

J Aircraft wing aspect ratio:-

Wing aspect ratio is equal to ratio of the wing span to wing chord for particular rectangular aircraft wing geometry. In this particular study, rectangular wing geometry is used for the aircraft design [3].

Wing aspect ratio $A=\frac{B}{C}$
B= wing span, C= wing chord

III RESULT AND ANALYSIS

A Trade off between payload and take-off distance:-

Objective final statement:-
Design objectives:-

MAXIMISE $W_{payload}$
MINIMISE S_{To}

$W_{payload} = (2340.6 – (0.00183[(B/C)^{0.5}*(B*C)^{1.5}\sec\Lambda_E(2850/B*C)*1.162 *(70/(t/c))^{0.5}]^{0.9}))$

$TOP=(\frac{5700}{B*C})_{TO}*(\frac{1}{1.6})*1.66*(1/1)$

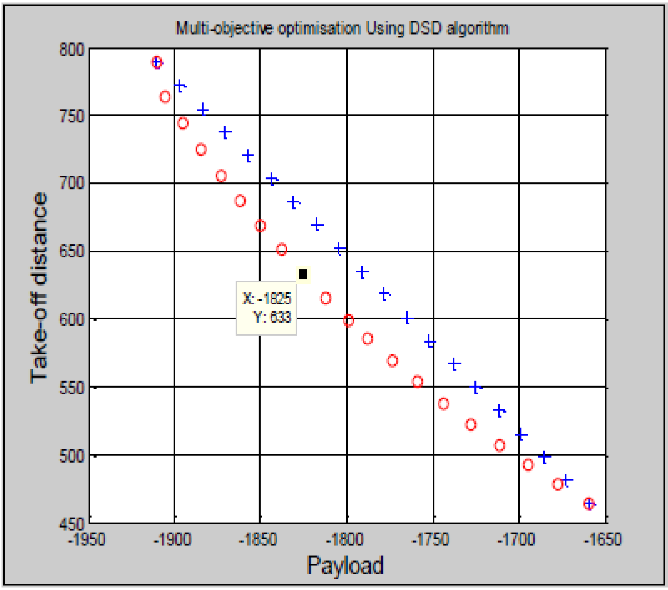
$S_{To}= 1.305(TOP)+12\sqrt{(TOP)(0.6)}$

TABLE-2. DESIGN VARIABLE

Design Variables	Definitions(units)	Bounds [min, max]
B (x(1))	Wing span [m]	[15; 25]
C(x(2))	Wing chord [m]	[1.8;2.5]

Design constrain:-
1) Maximum cruise speed: < 70 m/s

$G1= [\frac{2}{\varsigma}\frac{W}{S}]^{0.5}[\frac{k}{CDo}]^{0.25} -70$
2) Wing area < 70 m^2
G2= x(1)*x(2)-70
3) Range > 500000 m
$G3=\frac{V}{Cfs}*L_{OD\text{-}cruise}*\ln(W_i/W_f) -500000$
4) Aspect ratio > 5
G4 = 5-x(1)/x(2)
5) Maximum altitude < 21000 ft



Graph 1 Trade-off between payload and take-off distance

Discussion:-

According to the above graph and result, Whenever payload of the aircraft is increase, take-off distance of the aircraft is also increase. However, objective function of the design is maximising the payload and minimise the take-off distance. Whenever payload is maximising, other objective take-off distance is also maximise. Compromisations between the objectives are necessary. In the above graph the red line shows the Pareto solution and the blue line show the utopia plane [4,12].

In the above graph, direct search domain algorithm is used. There are two anchor points are found in above trade-off.

(1) 1^{st} anchor point value is (-1910.3,788.9)

(2) 2^{nd} anchor point value is (-1659,464.11)

According to above anchor point, if we performed single objective optimisation and maximise the payload up to 1910.3 kg, simultaneously take-off distance also increase to 788.9 meter. In addition, if we choose second objective as single objective optimisation than we get minimum take-off distance as 464.11 meter. But simultaneously payload weight is also decrease.

Above well-distributed pareto set point are obtained with the use of direct search domain algorithm. In the above table, all the points are pareto point, which are solution of the above problem. [12,13,14]

TABLE 3. DESIGN VARIABLE DIRECTION ACCORDING TO OBJECTIVE

Design variables	Wing span	Wing chord
Design Objective	B	C
Maximise payload	↓	↓
Minimise take off distance	↑	↑

According to above table, In the first objective function, the payload are maximised at the same time design variables are decrease. Means both wing span and wing chord are decrease And takes the value of lower bound of the variable.

In the second objective function take-off distance are minimise at the same time design variable are increase. Wing span and wing chord takes the value of upper bound of the variable.

With the use of fmincon fuction of the matlab we are find the single objective optimisation value and the obtain value is consider as a anchor point in the optimisation. Anchor point is nothing but single objective optimisation value.

Finally, it can be seen that the tendency of the wing span and wing chord is to increase in order to achieve the minimum take-off distance and similarly they have opposite effect on maximum payload objective function i.e. tendency of the wing span and wing chord is to decrease in order to obtain maximum payload. This shows that variables are increased in one objective function and decreased in other objective function which gives opposite effect hence this gives the strong trade-off between the two objective functions.

B Tradeoff between payload, wing loading and wing aspect ratio:-

Objective final statement:-

Design objectives:-

MAXIMISE $W_{payload}$
MINIMISE (W/S) wing loading
MINIMISE (A) Aspect ratio

$W_{payload}$ = (2340.6– (0.00183[(B/C)$^{0.5}$*(B*C)$^{1.5}$secΛ_E(2850/B*C) *1.162 *(70/((t/c))$^{0.5}$]$^{0.9}$))

(W/S) = 5700/(B*C)

A=(B/C)

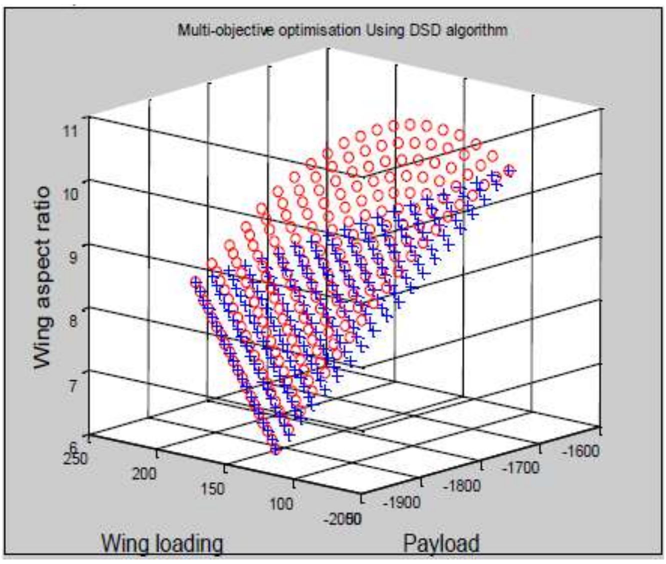
Design Variables:

TABLE-4. DESIGN VARIABLE

Design Variables	Definitions(units)	Bounds [min, max]
B (x(1))	Wing span [m]	[15; 25]
C(x(2))	Wing chord [m]	[1.8;2.5]

Design constrain:-

1. Maximum cruise speed: < 70 m/s
   $G1= [\frac{2}{\varsigma}\frac{W}{S}]^{0.5} [\frac{k}{CDo}]^{0.25} -70$
2. Wing area < 70
   G2= x(1)*x(2)-70
3. Range > 500000 m
   $G3=\frac{V}{Cfs}*L_{OD\text{-}cruise}* \ln(W_i/W_f) -500000$
4. Maximum altitude < 21000 ft



Graph 2:- Trade off between payload, wing loading and wing aspect ratio

According to the above graph, it shows the well-distributed trade-off between the payload, wing loading and wing aspect ratio. Because of the variable loading condition in the design aircraft, aspect ratio and wing loading are the important objective for the design. In addition, when the multi-objective optimisation performed during the conceptual phase of design, all the above pareto set point are the solution of the above problem. However, selection of the design point among the above pareto set point is depends on the market requirement and the performance of the aircraft[4]

In the above graph, direct search domain algorithm are used. There are three anchor point found in above trade-off.

(1) 1st anchor point value is (-1910.26,211.11,8.33)

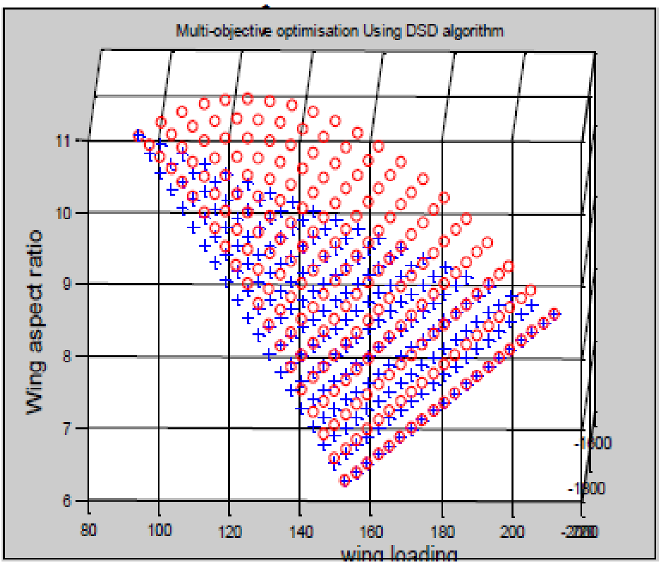
(2) 2nd anchor point value is (-1659.09,91.20,10)

(3) 3rd anchor point value is (-1910.26,152,6)

Above well-distributed pareto set point are obtained with the use of direct search domain algorithm. In the above graph, all the points are pareto point, which are solution of the above problem.

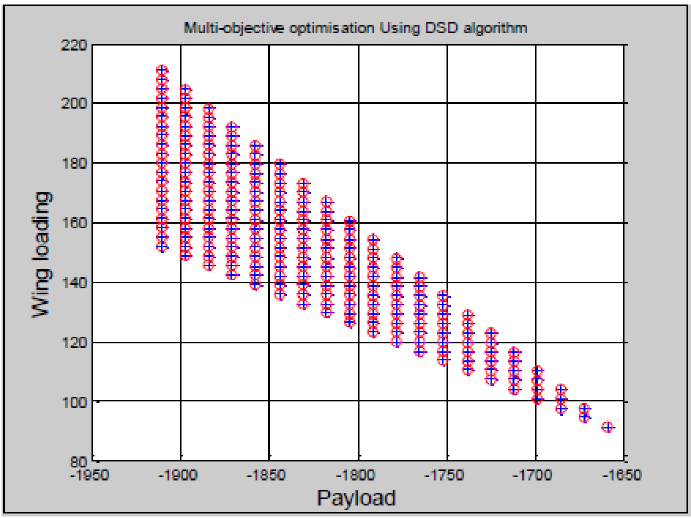
Now, above optimization are compare by each of the objective. In above graph 3, wing loading and wing aspect ratio are compared with each other. Wing loading is depends on wing aspect ratio. With the minimization of the wing aspect ratio, wing loading is maximized. However our objectives are minimizing the wing loading and wing aspect ratio. According to above graph strong trade-off occurred during the conceptual phase of the design.

In below graph 4, wing loading and payload are compared with each other. When the payload are maximize, at the same time wing loading are also maximized. However our objectives are minimizing the



Graph:3 Comparison between wing loading and wing aspect ratio

Wing Loading and maximize the payload. According to graph strong trade-off occurred during the conceptual phase of the design. As above, each objective has strong trade-off occurred between them. Below table shows the variable direction during the optimization of the above problem. Minimum value of the wing spans and wing chord are responsible to maximize the payload. In addition, however minimization of wing loading is depends on wing area. If wing area is increase at the same time wing loading is minimize [12,4,14].



Graph:4 Comparison between payload and wing loading

Furthermore, according to design requirement, design aircraft is low altitude. Because of low altitude of the aircraft, wing aspect ratio should be minimum. According to above graph strong trade-off occurred between above objectives.

TABLE-5. DESIGN VARIABLE DIRECTION ACCORDING TO OBJECTIVE

Design variables / Design objective	Wing span	Wing chord
	B	C
Maximise payload	↓	↓
Minimise wing loading	↑	↑
Minimise wing aspect ratio	↓	↑

IV CONCLUSION

In this study, Conceptual design of an Aircraft has been performed. Aircraft structure modal, weight modal, initially estimate of the weight and the mathematical modal are prepared. Direct search domain algorithm is very efficient to perform multi-objective optimization during the aircraft conceptual design. Thus, the program is capable of carrying out aircraft conceptual design. In this particular study two objectives and three objective design optimization studies are carried out. In this way "n" objective optimization are also carried out. Thus, integration of the multi-objective optimization and the aircraft mathematical modal during the conceptual design are successful. The baseline configuration is chosen Poland"s M18 dromader aircraft.

First bi-objective optimization problems are solved to maximize payload weight and minimize the take-off distance. After that three objective optimization problem are performed, which represent the well distributed Pareto- frontier points. According to the market requirement and required performance of the aircraft, the design point will be select from the Pareto set. Multi-objective optimization is a powerful tool to achieve the best design. But the accuracy of the optimal solution is depends upon the input parameter and the mathematical model.

After the investigation of the aircraft design process at conceptual phase, each objective are contradict with each other. In order to introduce optimization process in the conceptual design, mathematical model is important aspect. The conceptual design phase is peculiar in that contributing analysis, which are needed in order to model the behaviour of the aircraft, are in fact hundreds of generally simple models which are associated with thousands of variables. For performing trade-off analysis the contributing analysis need to be assembled according to the particular design study to be performed.

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