Estimation of component-wise drag coefficient and Planform Visualization of CG for Various Sub Components

AIRCRAFT DESIGN TABULATED DATASHEET OF AIRCRAFT CONFIGURATION LHW1E1

No. of passengers	405
W_{PL}	67272 kg
range	12000 km
Total Length	69 m

INPUTS AND DESIGN PARAMETERS AND VARIABLES FOR DRAG ANALYSIS

Component Length

The length of the respective component required to calculate Reynolds Number.

Wing/Tail/Fin: Respective MAC- MAC_{Wing}, MAC_{Tail}, MAC_{Fin}

Fuselage: Fuselage Length- $l_{fuselage}$

Engine Nacelle: Nacelle Length (Engine length)- $l_{nacelle}$

Density (ρ)

0.363918 kg/m^3

Atmospheric Density at Cruising Altitude of 11000 meters (36000 feet)

Dynamic Viscosity(μ)- 0.0000143226 Pa.s

Dynamic Viscosity of Atmospheric Air at Cruising Altitude of 11000 meters (36000 feet)

Velocity(v)

0.85 Mach (cruising speed)

Speed of sound at 11000 m= 295.070 m/s

Velocity= 0.85*295.070= 246.5595 m/s

Reynolds Number

R.No= $\rho v l/\mu$

Wetted Area

Effective Area in contact with Freestream Air.

Wing/Horizontal Tail/Vertical Tail

$$S_{wet\ Wing} = 2.003\ S_{exposed} \qquad \text{for } t/c < 0.05$$

$$S_{wet\ Wing} = 2\ S_{exp}\left[1 + 0.25(t/c)_{root}\left\{\frac{1+\tau\lambda}{1+\lambda}\right\}\right] \qquad \text{for } t/c > 0.05$$

where $\tau = \dfrac{(t/c)_{root}}{(t/c)_{tip}}$ and λ is taper ratio of the wing

Taper ratio is 0.3 for all Wing, Horizontal Tail and Vertical Tail.

t/c= 0.127 i.e Aerofoil has 12.7% thickness.

Fuselage

$$S_{wet\ Fus} = \pi d_{Fus}\, l_{Fus}\, \{1 - (2/\lambda_{Fus})\}^{2/3} \{1 + (1/\lambda^2_{Fus})\}$$

d_{fus}= 6m

l_{fuse}= 69m

λ_{fus}= 60/6= 11.5

Nacelle

Nacelle is assumed to be a cone with Diameter 3.6 m and Length of 7.2 m. Wetted area is calculated according to the surface area of cone. Multiplied by 2 since it's a twin-engine aircraft.

Average Skin Friction Coefficient (C_F)

Average skin friction coefficient C_F for any aircraft component like wing, fuselage, nacelle etc. is a function of Reynolds Number.

$$C_{F\ turb} = 0.455/(\log_{10} Re_l)^{2.58} \qquad \text{for } 10^6 < Re_l < 10^9$$

Form Factor (k)

Wing/Horizontal Tail/Vertical Tail

$$\mathbf{K_{Wing}} = \left[1 + \frac{0.6}{x_t}(t/c) + 100 \times (t/c)^4\right] \times \left[1.34 \times M^{0.18} \cos(\Lambda_m)^{0.28}\right]$$

Angle- 29.7°

x_t= Chordwise distance in meters from leading edge to max thickness point on MAC.

Fuselage

$$K_{Fus} = 1 + 60/\lambda^3_{Fus} + \lambda_{Fus}/400$$

Nacelle

$$K_{Nac} = 1 + 0.35/(l/d)_{Nac}$$

Flow Interference Factor (Q)

Flow interference factor Q is usually between 1.03 and 1.06. In this case it is assumed to be 1.03 for all the components.

Drag Coefficient (C_{d0})

$$C_{D0} = \sum K_i\, Q_j\, C_{Fi}\, (S_{wet\ i}/S).$$

COMPONENT-WISE DRAG COEFFICIENT DATASHEET

Component	Characteristic Length (m)	Reynold's Number	Average Skin Friction Coefficient C_F	Wetted Area	Form Factor K	Flow Interference Factor Q	C_{D0}
Wing	11.648	74229582.56	0.002171016	835.7175	1.32	1.03	0.003022359
Horizontal Tail	5.098	32488187.83	0.00244879	176.43	1.30	1.03	0.00338305
Vertical Tail	5.557	35413271.83	0.002417579	118.032	1.27	1.03	0.003262837
Fuselage	69	439718509.3	0.001705073	1153.73	1.06	1.03	0.001877826
Nacelle	7.2	45883670.54	0.002327058	104.3	1.19	1.03	0.002852275

Form factor k for Wing, Horizontal Tail and Vertical Tail are slightly on the higher side because the current design of the aircraft has very large components to account for.

Total Drag Coefficient

Total drag coefficient is the addition of all the C_{D0} values of the respective components.

Total C_{D0}= 0.014398347

Excrescence Drag

Excrescence Drag is usually considered to be 1% of the total drag coefficient. So, we multiply C_{D0} by a factor of 1.01 to get the total drag.

Excrescence Drag Coefficient= 0.01*0.014398347= **0.000143983**

Aircraft Total Drag Coefficient

Total C_{D0}= Component Drag coefficient+ Excrescence Drag coefficient

$C_{D0\text{-total}}$= 0.014398347+0.000143983= **0.01454233**

Limitations: *The current $C_{do\text{-}total}$ of the aircraft does not account for the landing gears and flap deflection. Also, the value of density and dynamic viscosity is assumed to be at the cruising altitude of 11000 m (36000 ft) and the aircraft velocity is at cruising speed that is, 0.85 Mach. Pylons are not considered in this calculation because they account for very insignificant drag value as compared to the nacelle. So, increasing the nacelle drag by a bit was a better option to compensate for the pylons. The drag due to the anti-shock bodies is not considered.*

INDUCED DRAG

Induced Drag Factor k is calculated as follows:

$k = 1/(\pi Ae)$

e- Ostwalt's Efficiency Factor

A- Aspect ratio= b^2/s

$$e = \frac{1}{\left(1+0{,}12\cdot M^6\right)\left\{1+\frac{0{,}142+f(\lambda)\cdot A\cdot(10t/c)^{0.33}}{(\cos\Lambda_{c/4})^2}+\frac{0{,}1\cdot(3\cdot n_e+1)}{(4+A)^{0.8}}\right\}}$$

Where $f(\lambda) = 0{,}005\cdot\left(1+1{,}5\cdot(\lambda-0{,}6)^2\right)$

n_e= Number of Engines under the Wing= 2

From this Formula,

e= 0.87

But as the planform shows, there are winglets present which decrease the induced drag by 5-10%.

Therefore, the final efficiency factor after considering the winglets will be slightly more than the calculated one.

Assume that due to winglets there is a 7.5% increase in efficiency factor e.

New efficiency factor e= 0.87+0.075*0.87= 0.936.

e= 0.936

Induced Drag Factor,

k= 1/(π*0.936*9.6) = 0.0354

k= 0.0354

INCREMENTAL DRAG FOR LANDING GEAR AND FLAPS

Typical values for incremental drag contributions ΔC_{D0} due to landing gears and flap deflections are suggested below for transport class of aircraft for modifying the drag polar for use during ground run, climb and acceleration segments:

$\Delta C_{D0\ LG} = 0.003$ to 0.005

$\Delta C_{D0Flap} = 0.9(c_{Flap}/c)^{1.38}(S_f/S)\sin^2\delta_{Flap}$

For $(c_{Flap}/c) = 0.3$; $(S_{Flap}/S) = 0.2$ and $\delta_{Flap} = 20^o$ we get $\Delta C_{D0\ Flap} = 0.004$

DRAG POLAR

The final aircraft drag polar is defined as:

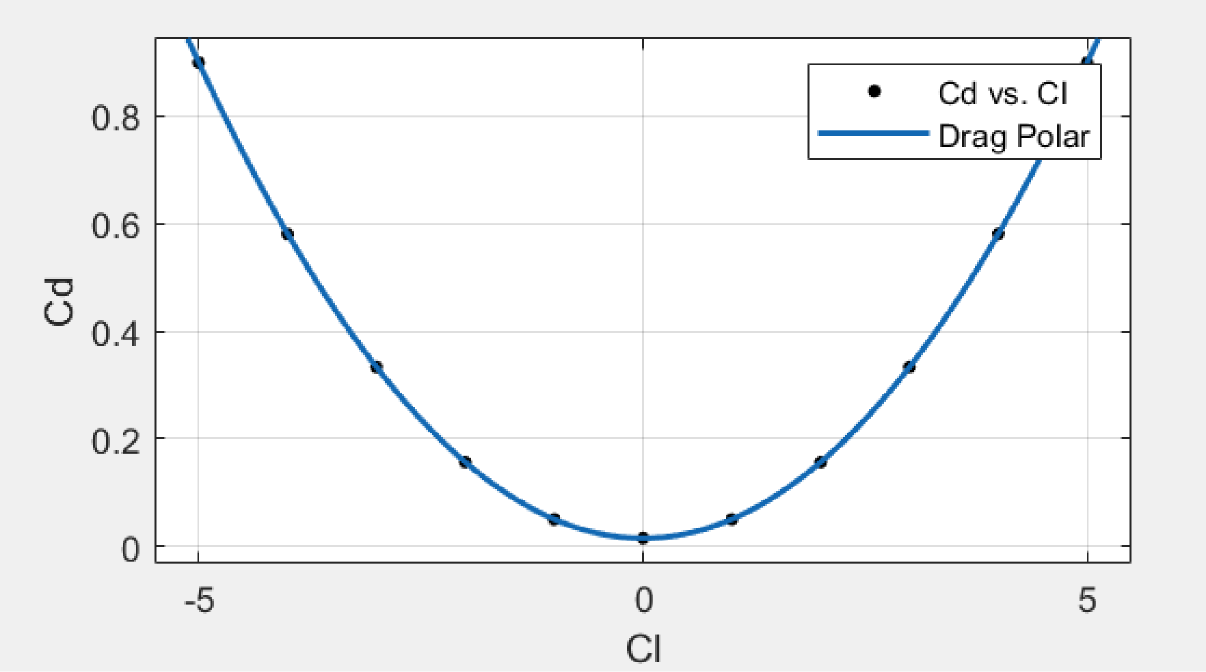
$$C_D = C_{D0} + kC_L^2$$

Cd0= 0.01454233

k= 0.0354

Therefore,

Cd= 0.0145+0.0354C_L^2



The value of C_{D_0} is slightly on the side because the size of aircraft is very large. The value is still under the acceptable limits for the provided amount of thrust.

MASS AND CG BUILD-UP OF THE AIRCRAFT AND ITS COMPONENTS

A. Components with Known CG

Sr.No	Component Name	Mass (lbs)	Distance from Nose to CG (ft)	Moment at CG (lbf.ft)
1	Wing	40196.345	107.555	4323317.886
2	Fuselage	76725.624	101.870	7816039.317
3	Horizontal Tail	5650.980	208.786	1179845.51
4	Vertical Tail	2938.563	216.843	637206.8166
5	Main Landing Gear	20587.395	124.507	2563274.789
6	Nose Landing Gear	2930.168	38.484	112764.5853
7	Propulsion System	38800	75.547	2931223.6
8	APU	2310	226.377	522930.87

B. Items with Notional CG at Centroid of Passenger Cabin

9	AC System	4950	101.870	504256.5
10	Furnishings	11176	101.870	1138499.12
11	Operational Items	4400	101.870	448228
12	Associated Crew Mass	19624	101.870	1999096.88

Total Mass of 12 Components= 230289.075 lbs

Total Moment of 12 Components= 24176683.9

CG of Components 1-12= (24176683.9/230289.075) = 104.984068 ft

C. Components whose CG is NOT known

13	Surface Controls	15614.707	-	-
14	Instrumentation	900	-	-
15	Hydraulics	2303.612	-	-
16	Electrical	4920	-	-
17	Electronics	1125	-	-

Total Mass of Operationally Empty Aircraft = 255152.394 lbs

CG of Operational Empty Aircraft = 104.984068 ft =105 ft

Operationally Empty Mass from Previous Session = Empty Mass + Associated Crew Mass

=101000+8940 kg= 109940 kg= 242376.195 lbs

Percentage Difference in Both Values = 5% i.e ACCEPTABLE

Percentage Fraction of Component to Operationally Ready Empty Aircraft

Operationally Ready Empty Weight = 255152.394 lbs

Sr. No	Component	Percentage Fraction
1	Wing	15.75
2	Fuselage	30.07
3	Horizontal Tail	2.21
4	Vertical Tail	1.15
5	Main Landing Gear	8.07
6	Nose Landing Gear	1.15
7	Propulsion System	15.21
8	APU	0.91
9	AC System	1.94
10	Furnishings	4.38
11	Operational Items	1.72
12	Associated Crew Mass	7.69
13	Surface Controls	6.12
14	Instrumentation	0.35
15	Hydraulics	0.90
16	Electrical	1.93
17	Electronics	0.44

Percentage fraction of Structural Components to Empty Weight (Without Crew)

Empty Weight = 222666.87 lbs

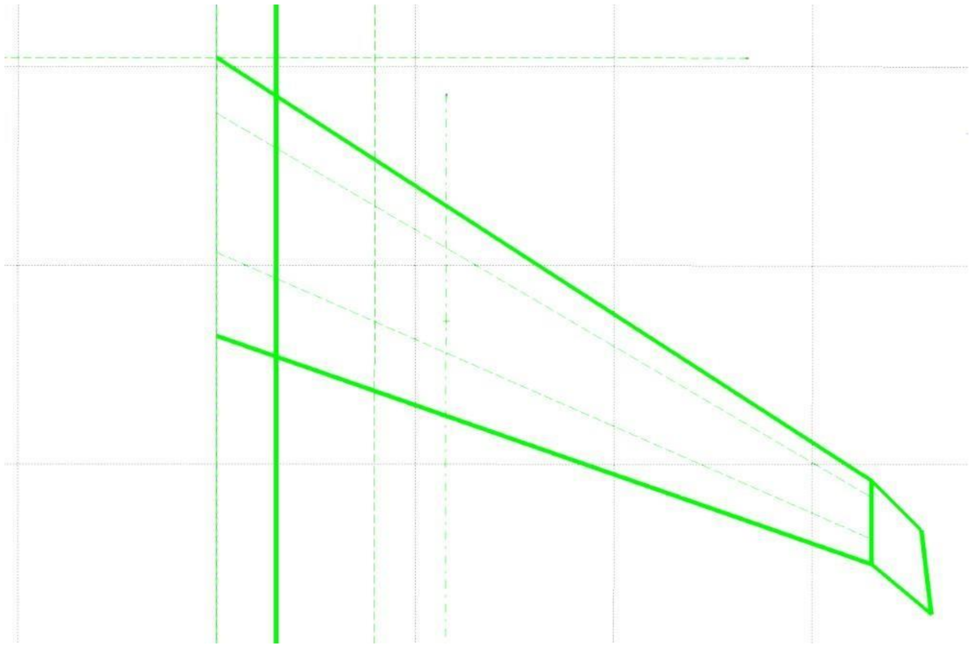
Sr. No	Component	Percentage Fraction
1	Wing	18.05
2	Fuselage	34.46
3	Horizontal Tail	2.54
4	Vertical Tail	1.32
5	Main Landing Gear	9.25
6	Nose Landing Gear	1.32
7	Propulsion System	17.43
8	APU	1.04

Assumptions and Conditions:

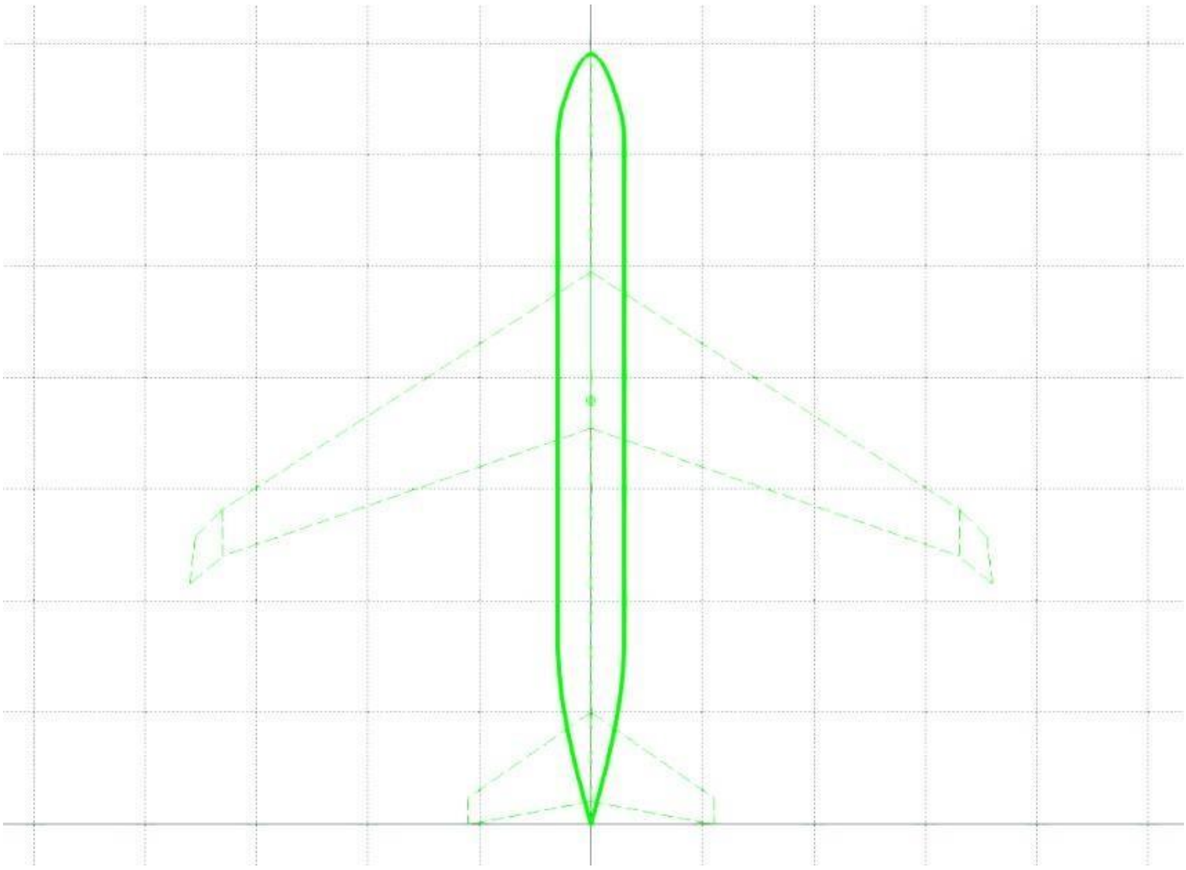
- Empty Weight and Operationally Ready Empty Weight are different. Empty Weight does not have the Associated Mass added due to the cabin crew.
- Technology factor of 0.75 has been used to scale down the component weight found from databases like Roskam, Nicholai, Raymer who have used metal as the base material unlike now when more advanced light weight composites are used.
- Slight manipulation in extra mass has been done to keep it under the acceptable limit.
- Number of Cabin Crew = 22
- Amenities' weight has been assumed for a long-haul flight (Transatlantic Flight) with 4 meals and drinks with some extra snacks.

PLANFORM VISUALISATION OF CG FOR VARIOUS COMPONENTS

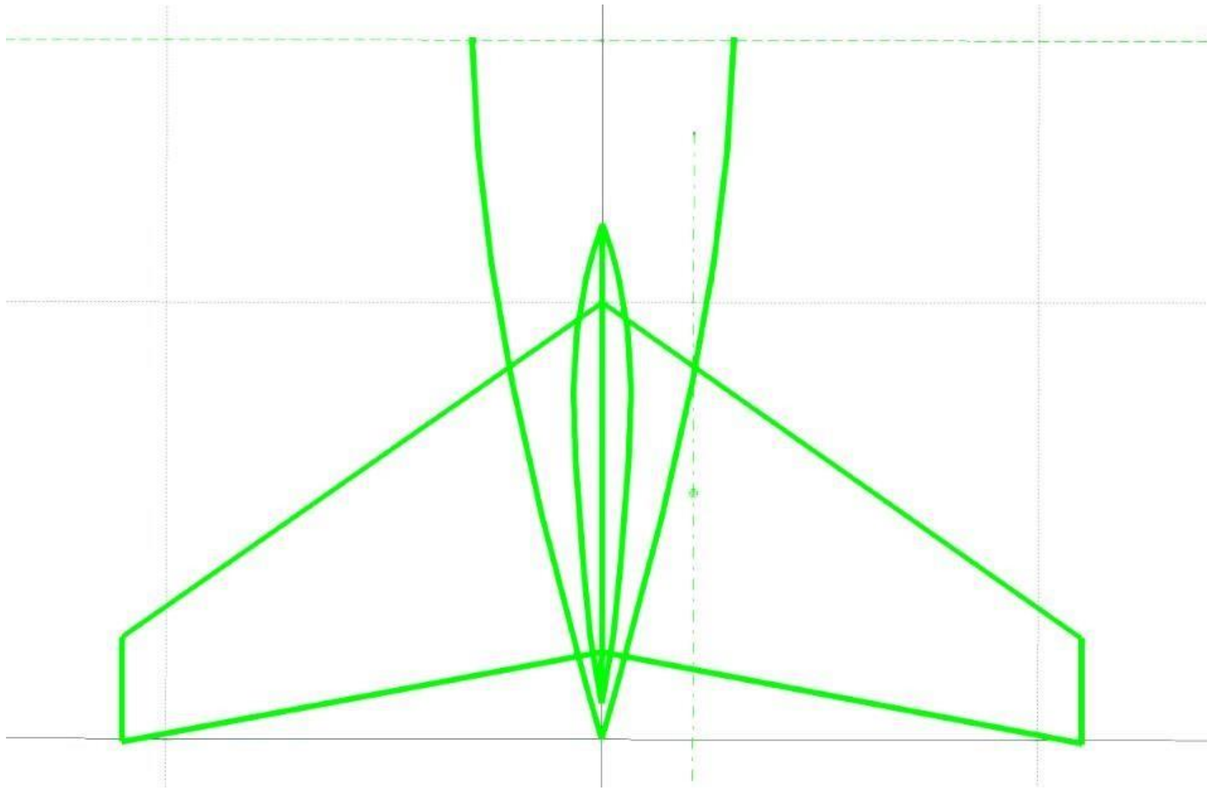
WING



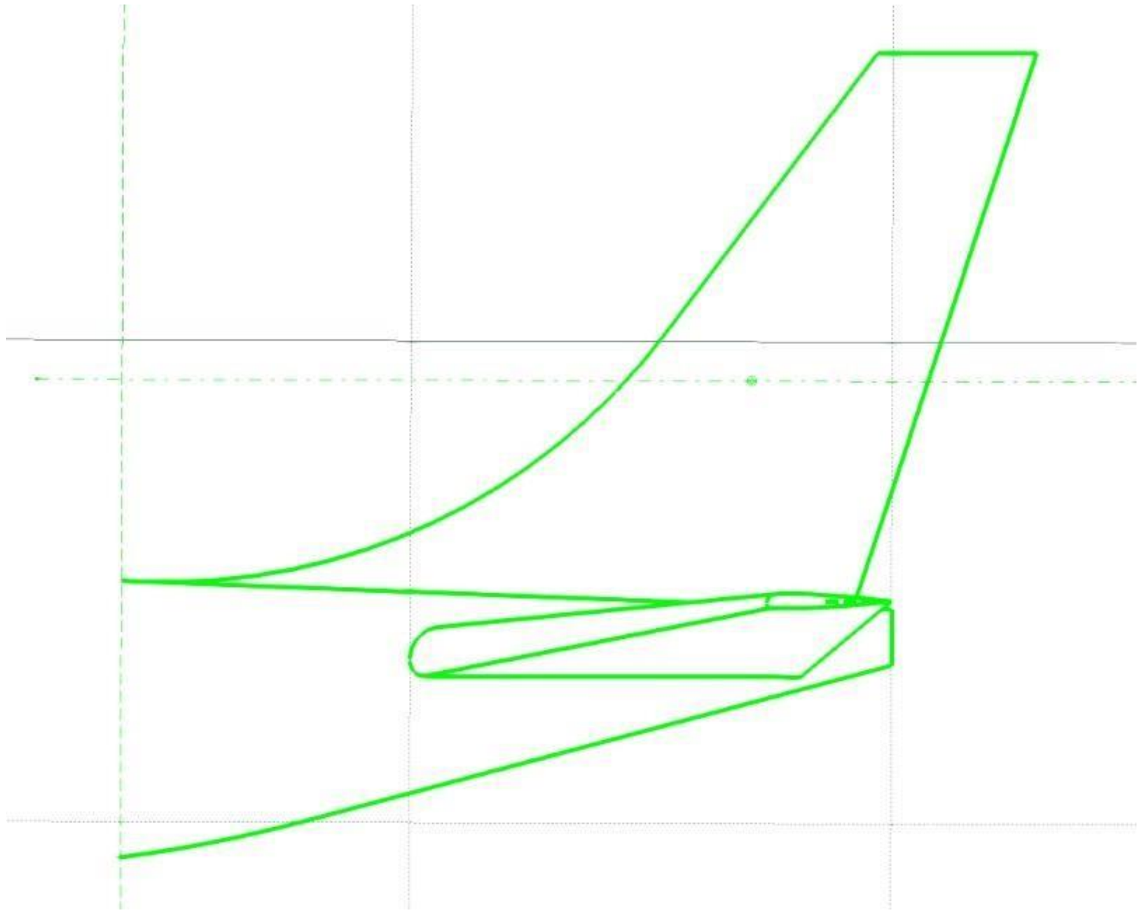
FUSELAGE



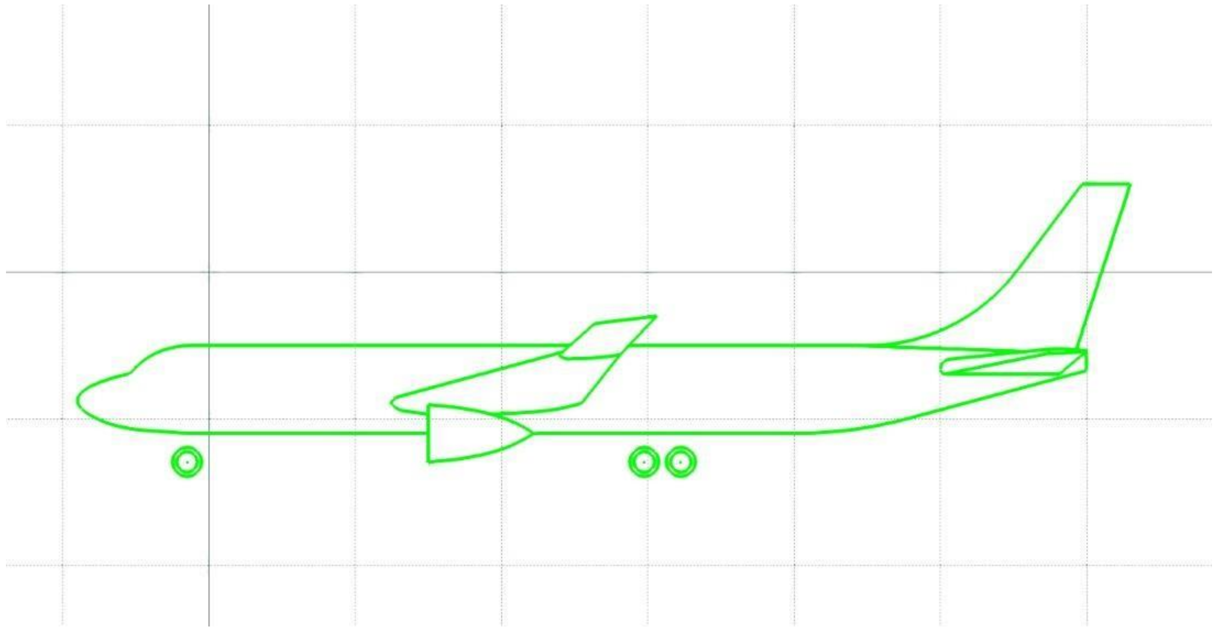
HORIZONTAL TAIL



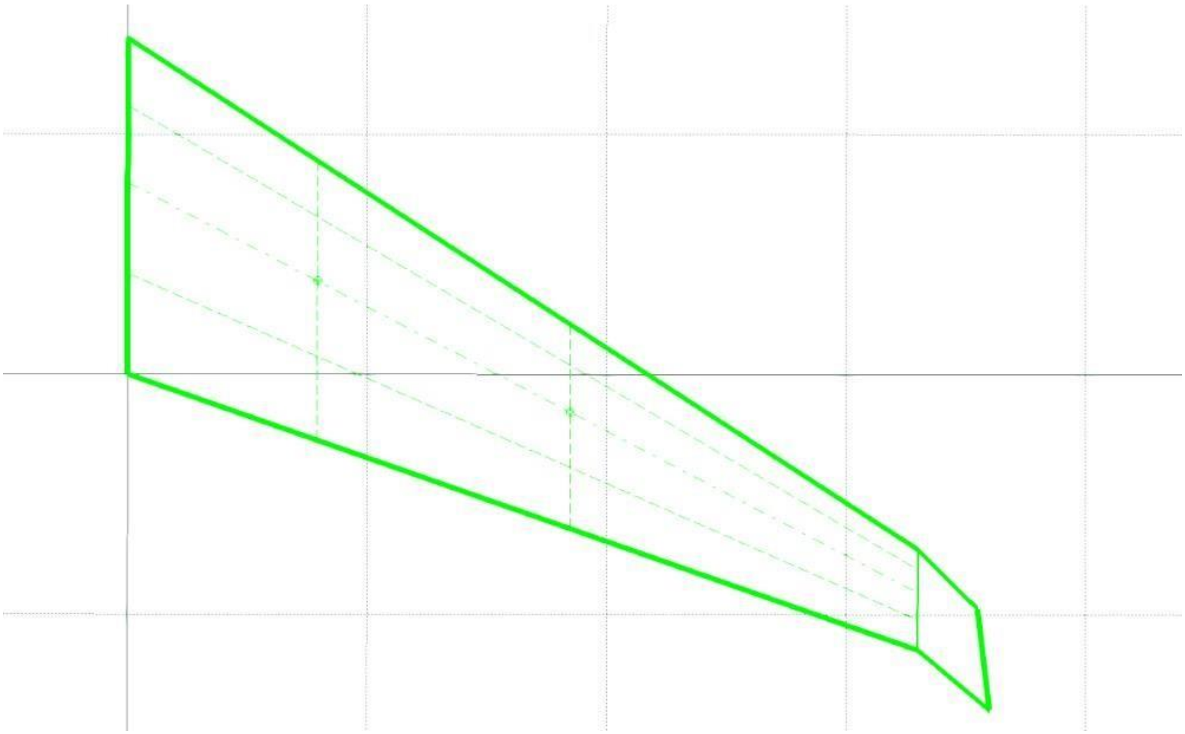
VERTICAL TAIL



LANDING GEAR



INBOARD AND OUTBOARD FUEL TANKS



FUEL TANK CONFIGURATION, MASS AND CG OF FUEL TANKS

Fuel Tank Capacity:

Mass of Fuel= **121352.616 kg (121.3 Tons)**

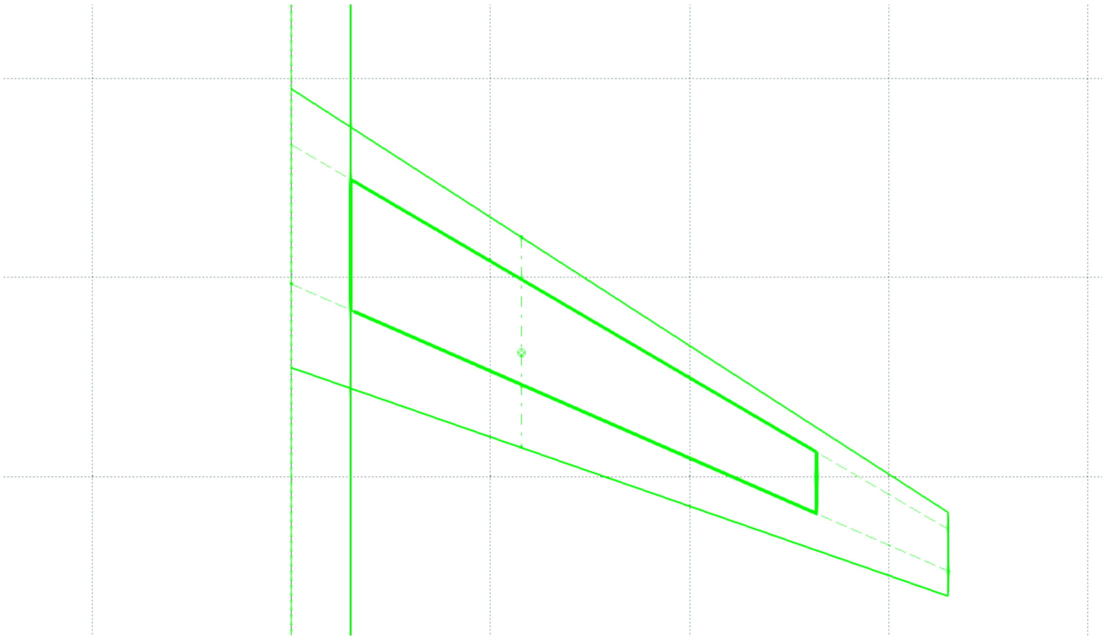
Volume of Fuel Tank= **169.95 m^3**

Mass of Fuel in Inboard Tanks= 60% of 121352.616= **72811.57 kg (72.8 Tons)**

Volume of Inboard Tanks= 60% of 169.95= **101.97 m^3**

Mass of Fuel in Outboard Tanks=40% of 121352.616= **48541.0464 kg (48.5 Tons)**

Volume of Outboard Tanks= 40% of 169.95= **67.98 m^3**



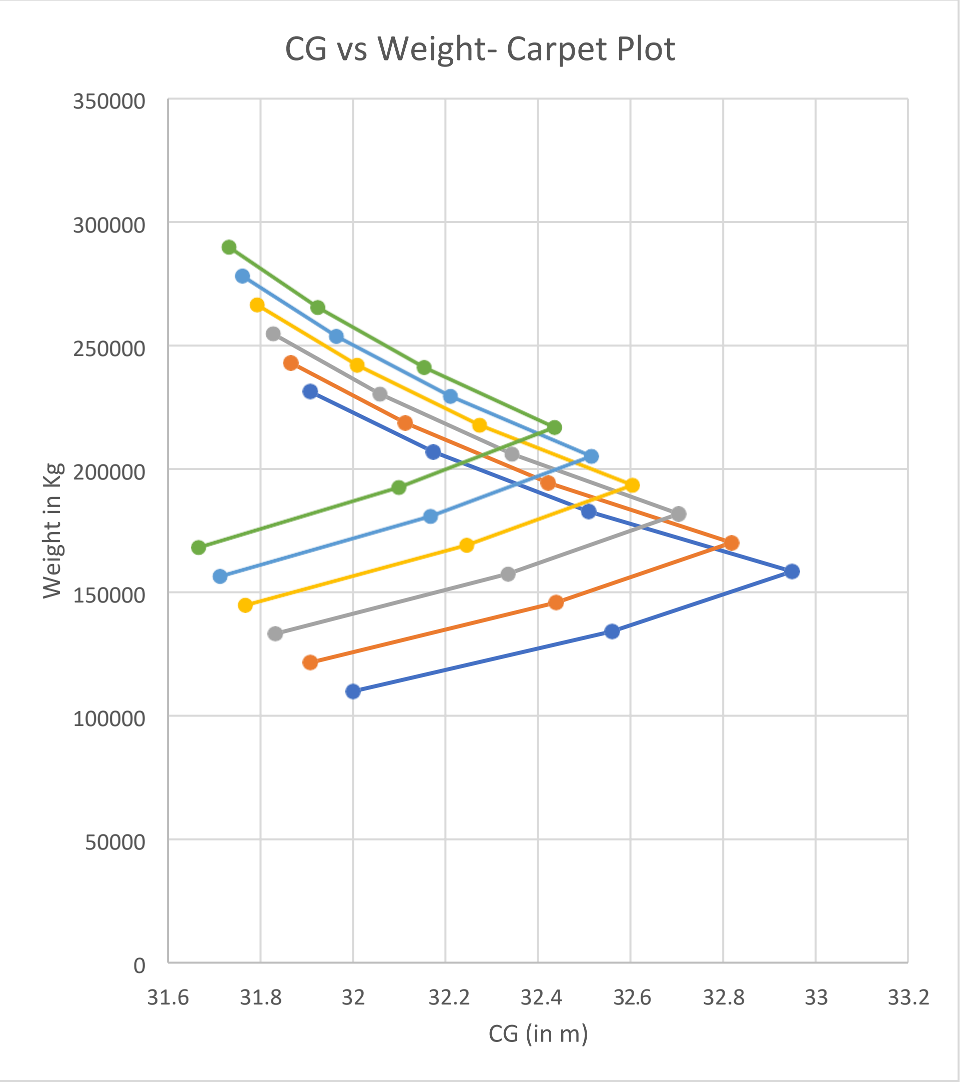
Parameter	Value
M_{empty}	109940 kg
X_{empty}	32 m
M_{IB}	Given in Table Below
M_{OB}	Given in Table Below
X_{IB}	29.641 m
X_{OB}	35.1 m

Percentage Fuel Consumed	Fuel in Inboard Tank	Fuel in Outboard Tank	CG Location
0	72811.57 kg	48541.04 kg	31.71 m
20	48541.04 kg	48541.04 kg	31.89 m
40	24270.52 kg	48541.04 kg	32.11 m
60	0	48541.04 kg	32.38 m
80	0	24270.52 kg	32.05 m
100	0	0	31.63 m

MASS AND CG OF PAYLOAD

Following table is calculated for payload percentage from 100% to 0 with an decrement of 20% per iteration.

Percentage Payload	Payload Mass (kg)
100	58332
80	46665.6
60	34999.2
40	23332.8
20	11666.4
0	0



The values of CG and Weight for each loading combination of aircraft (%Fuel+%Payload) can be traced from the above plot. The above Carpet Plot shows that for the current aircraft configuration, the CG traverses aft ward when emptying the inboard tank and forward when emptying the outboard tanks.

Parameter	Value
Distance of Front CG from Nose	31.65 m
Distance of Aft CG from Nose	32.9 m
Distance of Front CG from leading edge MAC	7.423 m
Distance of Aft CG from leading edge MAC	9.873 m
Difference	1.25 m

PERCENTAGE COMPARISION OF ALL THE COMPONENTS

Parameter Name	Calculated Parameter (From Mass-CG Analysis)	Calculated Parameter (From Take-off Weight Analysis)	Original Parameter w.r.t Reference Aircraft	Percentage Deviation (of mean) from Reference Aircraft
Seating Capacity	405	405	300	+25.92%
Max Take-Off Weight	289000 kg	278000 kg	242000 kg	+17.14%
Max Payload Weight	58332 kg	67272 kg	45600 kg	+37.72%
Max Fuel Weight	121352.616 kg	109728 kg	109185 kg	+5.82%
Empty Weight	109940 kg	101000 kg	129000 kg	-18.24%
Length	69 m	69 m	63.7 m	+8.32%
Wingspan	66 m	66 m	60.3 m	+9.45%
Thrust	988.6 kN	988.6 Kn	632 kN	+56.42%
Range	12000 km	12000 km	11750 km	+2.12%
Aspect Ratio	9.6	9.6	9.7	-1.03%

References

1. Suri, D. (2019). Design of an Optimized Inlet Shroud for a Flanged Diffuser.
2. Radhakrishnan, J., & Suri, D. (2018, June). Design and Optimisation of a Low Reynolds Number Airfoil for Small Horizontal Axis Wind Turbines. In IOP Conference Series: Materials Science and Engineering (Vol. 377, No. 1, p. 012053). IOP Publishing.
3. Suri, D., Radhakrishnan, J., & Nayak, R. (2019). Lattice Boltzmann Method to Analyse Fluid Flow Around a Circular Cylinder.