PRILIMINARY DESIGN ANALYSIS AND RC MODELLING OF A REGIONAL AIRLINER

GUTTAPALLI SAVARNI MAHARSHI*

ABSTRACT

This project is design analysis and modelling of a regional airliner or commuter aircraft. This project serves to work out the preliminary design and develop computer aided design (CAD) to fabricate a radio controlled (RC) model. Project starts with application of aircraft design process on a commuter class aircraft in which synthesized knowledge in subjects like aerodynamics, structures, propulsion and performance etc. is required. Most of the geometric parameters which drive the aircraft design will be obtained inpreliminary design process. By these parameters Computer Aided Design (CAD) modelis developed in CATIA V5 R19 software.From the Computer Aided Design (CAD) database,design calculationslike center of gravity, neutral point and static margins etc. are carried out. Light weight polystyrene foam board, Radio controlled electronics are used to fabricate model.

Key words: preliminary design, wing loading, CATIA V5, RC Modelling.

* B.Tech. Aeronautical Engineering, India

INTRODUCTION

The process of design of a device or a vehicle, in general involves the use of knowledge in divers fields to arrive at a product that will satisfy requirements regarding functional aspects, operational safety and cost. The design of an airplane, which is being dealt in this paper, involves synthesizing knowledge in areas like aerodynamics, structures, propulsion, systems and manufacturing techniques. The design of an aircraft is a complex engineering task follows as shown in fig.1

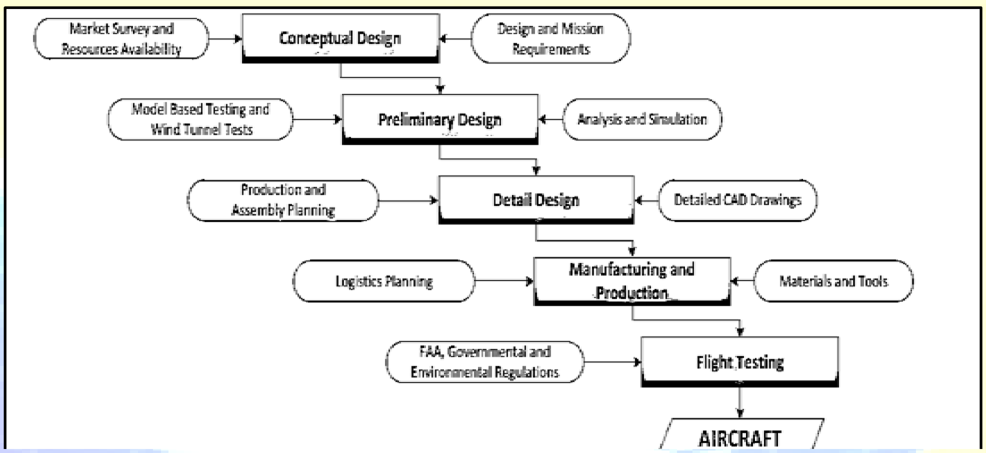


Fig.1: Detailed overview of aircraft design

Preliminary design starts with the assumption of a wing loading that satisfies the selected aircraft class either from previous records or by present relative aircrafts. Three fundamental aircraft parameters that are determined during the preliminary design phase are: Aircraft maximum take-off weight (WTO), Wing reference area (S_W or S_{ref} or S), and Engine thrust (TE or T) or engine power (PE or P).These three parameters will govern the aircraft size, the manufacturing cost, and the complexity of Preliminary Design calculation. A few other aircraft parameters such as aircraft zero-lift drag coefficient and aircraft maximum lift coefficient are estimated in this phase too. In some references, this process and this design phase is referred to as "*initial sizing*". This is due to the nature of the process which literally determines the size of three fundamental features of the aircraft.

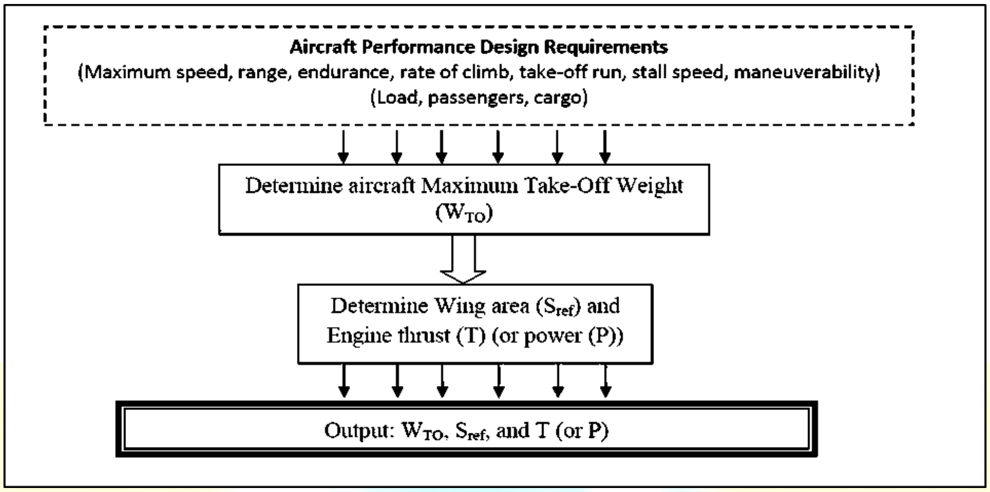


Fig.2: Preliminary design approach

Preliminary Design of a Regional Airliner

Requirements:

- ☑ Gross Still Air Range (GSAR):2000km
- ☑ No. of passengers: 60
- ☑ Cruise Mach number: 0.35~0.4
- ☑ Service celling (or) operating altitude: 8000m. Or 26247 ft.
- ☑ Power plants: turbo prop engines are chosen as they are efficient in the speed regimes between 0.3~0.5

Preliminary weight estimation:

- ☞ No of passengers and range plays a key role in weight estimation. In this case aircraft was considered to carry 60 passengers
- ☞ Gross Still Air Range (GSAR):2000km or 1250nm (for this GSAR the safe range will be around 1400km)
- ☞ 1 cabin crew should be assigned for 30 passengers so we have 2 cabin crew in present case. Including flight crew (captain & first officer) total crew will be 4 members

Total=passengers +crew=60+4=64 members

☞ As per aircraft regulation allowable passenger weight is 110kgf (82kg passenger weight with carry-on baggage+28 kgf check-in baggage)

Therefore payload weight will be

64×110=7040kgf=69038.816N

The gross weight $Wg = \frac{Wpay+Wcrew}{1-\left(\frac{Wf}{Wg}\right)+\left(\frac{We}{Wg}\right)}$

☞ From other relevant aircrafts the "W_g" is taken approximately as 21500kgf and From previous aeronautical research data we can approximate these values for present case

- Aspect ratio: 12
- Taper ratio: 0.5
- Swept back: 0^0
- W/S=350 kgf/m^2

From these values,

$$S = W_g\left(\frac{S}{W}\right) = 21500\times\left(\frac{1}{350}\right) = 61.43\text{m}^2$$

Wing parameters:

$$\text{Wing span: } b = \sqrt{SA} = \sqrt{12\times61.43} = 27.15\text{m}$$

$$\text{Root chord: } C_r = \frac{2}{(1+\lambda)}\times\frac{S}{b} = \frac{2}{1+0.5}\times\frac{61.43}{27.15} = 3.02\text{m}$$

$$\text{Tip chord: } C_t = C_r\times\lambda = 3.02\times0.5 = 1.51\text{m}$$

Revised weight estimation:

$$Wg = \frac{Wpay + Wcrew}{1-\left(\frac{Wf}{Wg}\right)+\left(\frac{We}{Wg}\right)}$$

Estimation of empty weight fraction:

We have the equation $\frac{We}{Wo}$=A.W_o^c

Where W_o= takeoff gross weight, 'A' & 'c' are values depending on aircraft type

In this case for "twin turbo prop" so, A=0.92 &c=-0.05 (from Ref.1 table 3.1)

$$\frac{We}{Wo}=0.92(2.202W_g)^{-0.05}$$

Fuel fraction:

☞ The fuel weight depends on the mission profile and the fuel required as reserve. The mission profile for a civil transport aircraft involves:

- Take off
- Climb
- Cruise
- Loiter before landing
- Descent
- Landing

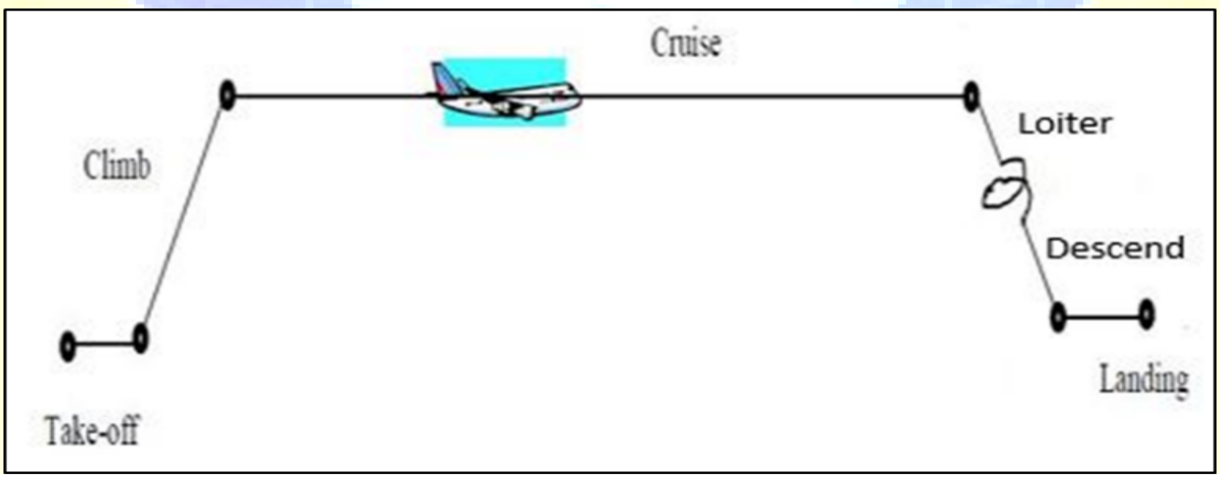


Fig.3: Simple cruise mission of transporter aircraft

Warm-up, taxi & takeoff:

The weight of the airplane at the start of take-off is W_0 and the weight of the airplane at the end of the take-off phase W_1. The ratio $(\frac{W1}{W0})$ is estimated using the guidelines given in Ref 1

$$\frac{W1}{W0}=0.98$$

Climb:

The weight of the airplane at the start of climb is W_1 and the weight of the airplane at the end of the climb phase is W_2. The ratio $(\frac{W2}{W1})$ for this phase isestimated by following guidelines given in Ref 1

$$\frac{W2}{W1}=0.99$$

Cruise:

The weight of the airplane at the start of cruise is W_2 and the weight of the airplane at the end of the cruise phase is W_3. The ratio $(\frac{W3}{W2})$ for the cruise phase of flight is calculated using the following expression from

$$\frac{W3}{W2}=\exp\left(\frac{-R*BSFC}{3600\eta p(\frac{L}{D})}\right)$$

- The safe range=1334km
- For turbo prop driven aircrafts $(\frac{L}{D})_{max}=(\frac{L}{D})_{cruise}=16$
- Allowing additional distance covered due to head wind of 15km/hr. (say) and Provision for diversion to other airport in an emergency
- Service celling is at 262470ft. at M0.35

Speed of sound at 8km or 26247ft. $a=\sqrt{\gamma RT}$

at 8km altitude temperature=236.23k

Pressure=3.5651N/m^2

Density=5.2578 kg/m^3

Speed of sound $a=\sqrt{1.4\times236.23\times287}$

a=308.1m/s

V_{cruise}(u)=108 m/s (Mach number=$\frac{u}{a}$)

Time to cover cruise range of 1334km at V_{cr}=389km/hr. is $\frac{1334}{389}$=3.34 hours

A head wind of 15m/s or 54 km/hr additional distance will be 54×3.34=180.4 km

Allowance for diverging airport=300km

Distance = 180.4+300=480.4km

Total distance R=1300+480.4=1780.4km

BSFC is taken as 0.5/hr.

Substituting the values in equation

$$\frac{W3}{W2}=\exp\left(\frac{-1780.4*0.5}{3600*0.8*16}\right)=0.98$$

Loiter:

$$\frac{W4}{W3}=\exp\left(\frac{-E\times BSFC\times V}{1000\times \eta p\times\left(\frac{L}{D}\right)}\right)$$

- Loiter endurance is taken as 30 minutes
- For turboprop $(\frac{L}{D})_{loiter}$=0.866($\frac{L}{D}$)=0.866×16=13.85
- BSFC= 0.6* ηp= 0.8

$$\frac{W4}{W3}=\exp\ (-0.5\times0.5\times\frac{389.1}{1000}\times0.8\times\ (13.85)\ =0.99$$

Landing: from Ref.1 table 3.2

$$\frac{W5}{W4}=0.995$$

✶ **Total**=0.98×0.99×0.98×0.99×0.95=**0.894**

- Allowing reserve fuel of 6%

$$\text{Fuel fraction}=\frac{Wf}{Wg}=1.06(1-\frac{W5}{W0})=1.06(1-0.894)=0.112$$

Gross weight (W_g):

$$Wg = \frac{Wpay+Wcrew}{1-\left(\frac{Wf}{Wg}\right)+\left(\frac{We}{Wg}\right)} \text{so,} \quad W_g = \frac{7040}{1-0.112-0.92(2.202Wg)-0.05}$$

Iteration method:

Table1: Gross weight iterations

W_g	W_e/W_0	W_g
21000	0.537694	20096
20500	0.538343	20134
20250	0.538673	20153
20180	0.538766	20159
20160	**0.539793**	**20160**

Estimation of Wing Loading $(\frac{W}{S})$ and Thrust Loading $(\frac{T}{W})$:

Estimation of wing loading and thrust loading is based on various considerations

Landing distance consideration:

☞ From aeronautical charts the balanced field length for such weight is taken as 1250m

$$S_{land}=1250\text{m}$$

☞ From ramyer[1] text C_{Lmax} of single slotted flap is 2.2

$$W=L=\frac{1}{2}\rho V^2_{stall} S C_{Lmax}$$

$$\left(\frac{W}{S}\right) = \frac{1}{2}\rho V^2_{stall}C_{Lmax}$$

$$\text{Approach velocity } V_a=\sqrt{\frac{Sland}{0.3}}$$

$$V_a=1.3 V_s \& V_{td}=1.15V_s$$

$$V_a(\text{in knots})=\sqrt{\frac{4101}{0.3}}=116.91\text{kts}=51.96\text{m/s}$$

$$\text{Stall speed } V_s=\frac{Va}{1.3}$$

$$V_s=\frac{51.96}{1.3}=39.96=40\text{m/s}$$

$$\left(\frac{W}{S}\right) = \frac{1}{2}(1.2250)*40^2*2.2=2156 \text{ N/m}^2$$

For such airplanes W_{land}=0.85$W_{takeoff}$

$$\left(\frac{W}{S}\right)_{TO}=\frac{1}{0.85}\left(\frac{W}{S}\right)_{land}=\frac{1}{0.85}(2156)=2536.4\ N/m^2$$

- ✶ Allowing ±10% variation in V_{stall} gives 2371.6 $N/m^2<\left(\frac{W}{S}\right)<2790\ N/m^2$

Maximum speed considerations V_{max}:

- ☞ Maximum velocity is decided on bases of Mach number.

$$M_{max}=M_{cruise}+0.04==0.4+0.04=0.404*308.1=124.4724\ m/s$$

- ☞ The drag polar is alternatively given by

$$C_D=F_1+F_2P+F_3P^2$$

$$\text{Where } F1=C_{fe}\left(1+\frac{Sht}{S}+\frac{Svt}{S}\right)\left(\frac{Swet}{S}\right)\ w$$

$$F_2=\left(\frac{CD.o-F1}{\frac{W}{S}}\right)$$

$$F_3=\frac{K}{q2}$$

- ☞ As we assumed the values from relevant aircrafts

$$\frac{Sht}{S}=0.29\ \frac{Svt}{S}=0.21$$

$$C_{D.o}=C_{fe}\left(\frac{Swet}{S}\right)$$

- ☞ For equivalent tapered or trapezoidal wing with 0^0sweep

$$C_r=C_r-\left(\frac{Cr-Ct}{\frac{b}{2}}\right)$$

$$C(y)=3.02-1.111y$$

☞ Taking fuselage diameter as 2.85(based on similar aircraft's aisle height, width of seats & no of seats in cabin)

☞ The chord y=1.425m is the root chord of exposed wing

$$C_{r\ (exposed)} = 3.02 - 0.111(1.425) = 2.86m$$

$$\text{Semi span of exposed wing } b_e = \left(\frac{27.15}{2}\right) - \left(\frac{2.85}{2}\right) = 12.15m$$

$$(S_{exp})_{wing} = \left(\frac{Crexp+Ct}{2}\right) * \frac{be}{2} * \text{no of wings}$$

$$(S_{exp})_{wing} = \left(\frac{2.86+1.51}{2}\right) * 12.15 * 2 = 53.09m^2$$

☞ Wetted area of exposed wing

$$S_{wet} = 2 * S_{ew} * (1 + 1.2(t/c)_{avg})$$

$(t/c)_{avg}$–average thickness to chord ratio

$$S_{wet} = 2 * 53.09 * (1 + 1.2(0.120)) = 121.46m^2$$

$$C_{D.O} = C_{fe}\left(\frac{Swet}{S}\right) \qquad C_{fe=}\text{skin friction drag}$$

$$C_{D.O} = 0.00652\left(\frac{121.46}{61.43}\right) = 0.01289$$

$$F_1 = 1.5 * 0.005022 = 0.007533$$

☞ For straight wing aircraft 'e'(Oswald span efficiency)=$1.78(1-0.045A^{0.68})-0.64$

☞ Here $e=1.78(1-0.045(12)^{0.68})-0.64=1.120$

$$K = \frac{1}{\pi Ae}$$

$$K = \frac{1}{\pi * 12 * 1.120} = 0.0236$$

$$\left(\frac{L}{D}\right)_{max} = 16 \text{ for turbo prop driven aircrafts}$$

$$C_{D.o} = 1/4K \left(\frac{L}{D}\right)^2_{max} = \frac{1}{4 * 0.0236 * 162} = 0.0413$$

$$C_D=0.0413+0.0236C_L^2$$

$$C_{fe}=\frac{0.0413}{6.33}=0.00652$$

$$F_2=\frac{0.0413-0.01289}{3433}=8.275\times10-6\ m^2/N$$

The value depends on dynamic pressure at V_{max}

$$q_{max}=\frac{1}{2}\rho\ V_{max}=0.5*0.364*157.1=3629.4 \qquad (V_{max}=0.404*389=157.1m/s)$$

$$F_3=\frac{K}{q2}=\frac{0.0236}{(3629.4)2}=1.7910^{-9}\ m^4/N^2$$

☞ The relation for the thrust required for V_{max} is

$$T_{vmax}=q_{max}\ (F_1/P+F_2+F_3P)$$

$$\left(\frac{W}{S}\right)_{optimum}=\sqrt{\frac{F1}{F3}}=\sqrt{0.007533/1.79*10^{-9}}=2068.8\ N/m^2$$

$$T_{Vmax}=3629.4\left(\left(\frac{0.007533}{2068.8}\right)+8.275*10^{-6}*1.79*10^{-9}*2068.8\right)=0.01321$$

Minimum fuel for range (W_{fmin}) consideration:

$$W_{fmin}=R/3.6*\sqrt{\frac{\rho}{2}}*BSFC*\sqrt{q_{cr}}\ (F_1/P+F_2+F_3P)$$

☞ Range R=2000 & BSFC=0.5lb/(lb/hr)

$$V_{cr}=108m/s$$

$$Q_{cr}=0.5*0.364*108^2$$

$$F_3=\frac{0.0236}{2123}=1.1116*10^{-5}$$

$$F_1=0.007533$$

$$F_2=8.275*10^{-6}\ m^2/N$$

$$P_{optimum}=2068.8\ N/m^2$$

$$W_f=\frac{2000}{3.6}*\quad\frac{0.346}{2}*0.5*\sqrt{\quad 2123*(\frac{0.007533}{2068}+8.275*10^{-6}*1.1116*10^{5}*2068.8)}$$

$$W_f=0.02040$$

- Choice of wing loading is between $2068.8<(\frac{W}{S})<2790\ N/m^2$

Final choice of wing loading:

- For final choice of wing loading, take-off requirements are considered
  - ☑ High wing loading=shorter wing + low drag + low empty weight & high thrust loading
  - ☑ Low wing loading=large wing + more drag + more weight & low thrust loading
- We have assumed balanced field length of 1250m=4101ft, For this balanced field length take-off parameter is $\left(\frac{\frac{W}{S}}{\sigma CLTO\left(\frac{BHP}{W}\right)}\right)$ is 150

$$\sigma=1 \text{ at sea level}$$

$$C_{LTO}=0.8*2.2/1.21=1.45$$

$$BHP/W=550\eta p/\gamma(hp/W)=\frac{550\times0.8}{354.3}\times0.2=0.248$$

$$140=\frac{\frac{W}{S}}{1\times1.45\times0.2}$$

$$(\frac{W}{S})=44lb/ft^2=2112.4\ nm^{-2}$$

Thrust requirement for takeoff:

- Typical value of $(\frac{hp}{W})$ for twin turboprop is 0.2

$$T_{reqTO}=0.2*20160*9.81=39553N=40KN$$

☞ Allowing more 25KN as thrust requirement for V_{max}

$$T_{req}=65KN$$

☞ As we are using two turbo prop engines the requirement will be shared as

$$T_{req}=32.5KN \text{ for each engine}$$

Aircraft design parameters:

Wing design:

☞ Weight & wing loading from the calculations are

$$W=20160kgf\ (197568.6N)\ \&\left(\frac{W}{S}\right)=2112.4\ N/m^2,\ S=\frac{197568}{2112.4}=93.53m^2$$

$$\text{Span } b=\sqrt{12*93.53}=33.5m^2$$

$$\text{Root chord } C_r=\frac{2*93.53}{1+0.5*33.5}=3.72m$$

$$\text{Tip chord } C_t=C_r\lambda=3.72*0.5=1.86$$

$$\text{Mean aerodynamic chord (MAC)} =\frac{2}{3}\frac{(1+\lambda+\lambda 2)}{1+\lambda}, MAC=2/3\left(\frac{1.75}{1.5}\right)*1.86=3.255m$$

Fuselage layout:

☞ From data $\frac{lf}{b}$=0.84 is considered

L_f=0.84*33.5=28.14m

☞ No seating classes are divided in this aircraft, all seats are economy class only

☞ Depending on number of seats, aisle width, aisle height & diameter of fuselage

- Nose length=1m
- Cockpit length=2.5m

- Cabin length=15.2m
- Rear lengths=10.2m

Total=28.14m

Tail surface: We have assumed $\frac{Sht}{S}$=0.29 $\frac{Svt}{S}$=0.21

$$S_{ht}=0.29*93.5=27.1m^2$$

$$S_{vt}=0.21*93.5=19.6m^2$$

$$b_h=\sqrt{A_h S_h}=\sqrt{5*271}=11.6m$$

$$b_V=\sqrt{A_V S_V}=\sqrt{1.7*19.6}=5.7m$$

Chord lengths

- $C_{rh}=\frac{2Sh}{bh(1+\lambda h)}=\frac{2*27.1}{11.6(1+1.5)}$ =3.1m
- C_{th}=3.1*0.5=1.55m
- $C_{rv}=\frac{2Sv}{bv(1+\lambda)}$ =2*19.6/5.7 (1+0.31) =5.24m
- C_{tv}=5.24*0.31=1.62m

Location of power plant:

Turboprop engines are located lower side of the wing with pylons to reduce drag &fuel will be near.34% of b/2 is the location to fix the engines.

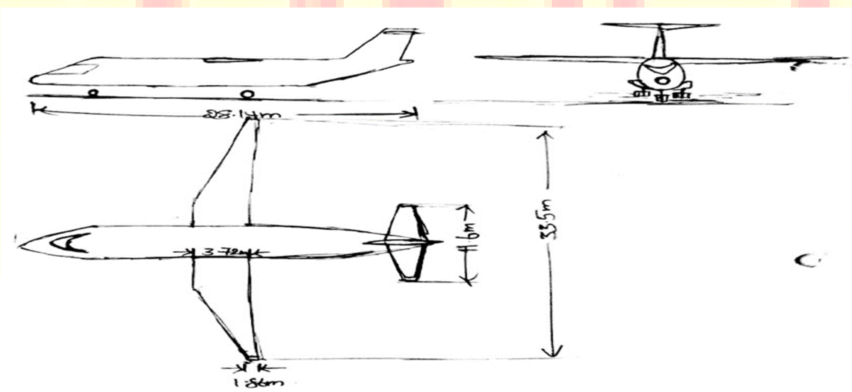


Fig.4: Three view diagram of the aircraft

Developing designs into CAD models:

CATIA (Computer Aided Three-dimensional Interactive Application) V5 R19 software was used to develop the CAD model.

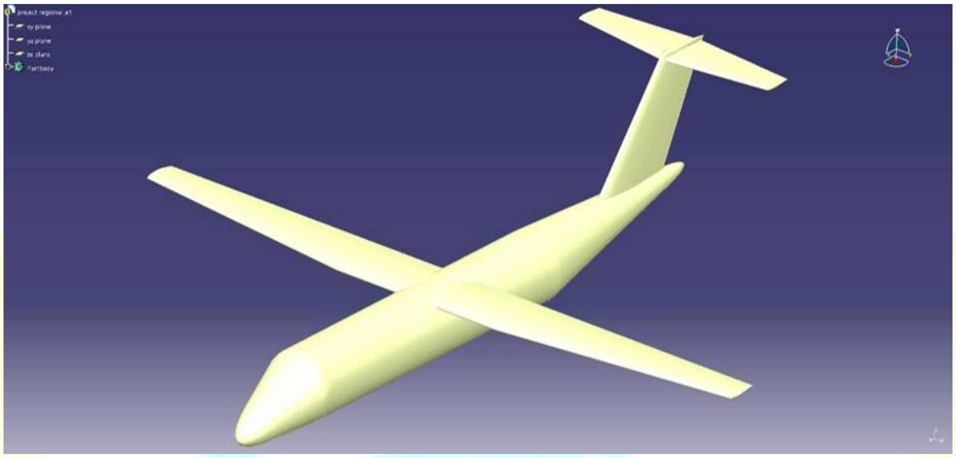


Fig.5: CAD model of the aircraft

RC Aircraft Design Measures:

In model aircraft design there are some critical measures that will play key role in flying. Those measures are the Static Margin (SM), Center of Gravity (CG), Horizontal Stabilizer Volume (VH), Vertical Stabilizer Volume (VV), and Cubic Wing Loading (CWL), Power-to-Weight Ratio (P/W) or Thrust-to-Weight Ratio (T/W). In order to find these key measures we must first find a few other measures beforehand. These measures would be the dimensions and areas of the main wing, horizontal and vertical stabilizers, the aerodynamic centers (AC), Neutral Point (NP), and the distance between the two centers.

$$\text{Wing area}=41\left(\frac{11+11}{2}\right)=41*11=451\text{cm}^2$$

$$\text{Horizontal tail area}=13.25*\left(\frac{7.8+4.3}{2}\right)=13.25*6.05=80.160\ \text{cm}^2$$

$$\text{Vertical tail area}=15.5*\left(\frac{9.5+8}{2}\right)=15.58.75=135.6\ \text{cm}^2$$

☞ Aerodynamic center for wing is at **2.8cm** from leading edge.

☞ Aerodynamic center for horizontal tail is at **2.1cm** from leading edge.

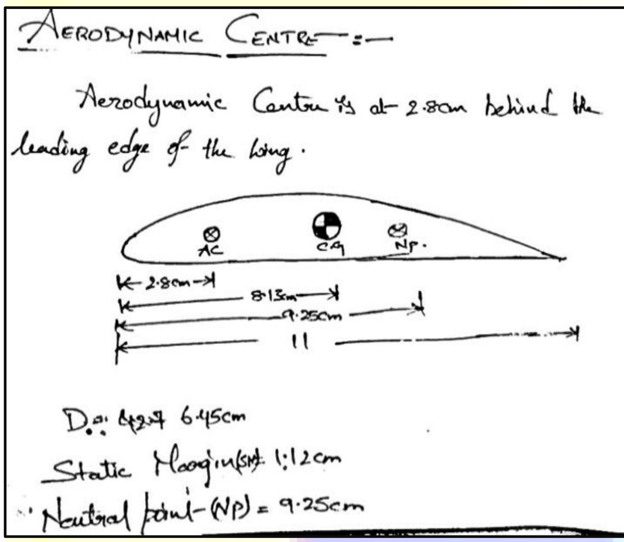
☞ Distance between AC & NP, D=$L*\left(\frac{stab\ area}{wing\ area+stab\ area}\right)$=42.7*$\left(\frac{80.16}{451+80.16}\right)$=6.45cm

Static Margin:

The Static Margin is a measure of stability of your plane. Most planes have a SM of 5% to 15% of the MAC which means the CG is 5% MAC to 15% MAC in front of the NP.

$$\text{Static Margin(SM)} = \frac{MAC}{\sim 10} = \frac{11.2}{10} = 1.12\text{cm}$$

We can now use D to find the location of NP by subtracting it from L.



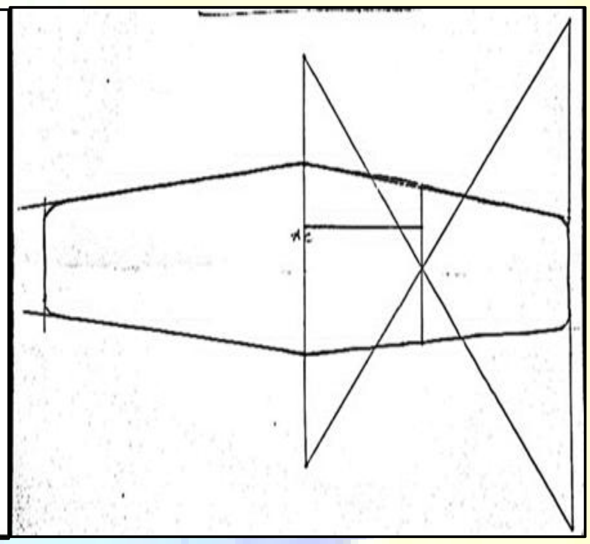


Fig.6: Aerodynamic measures of wing **Fig.7: Aerodynamic center of horizontal tail**

Fig.8: Aircraft model and its first flight

Conclusion:

Preliminary design procedure was worked out on a regional class aircraft with all pre-defined requirements that are meant to be fulfilled by every aircraft with in that class. The advancements in aerodynamically structured aircraft made aviation more efficient than before. By preliminary design parameters CAD model of the aircraft is developed in CATIA V5 R19 software. Radio controlled aircraft is developed by the classic aero modeling theory and aerodynamic calculations and the plan was worked out on extruded polyurethane foam boards. Mean aerodynamic chords, neutral point, center of gravity etc. calculations are well defined that makes neither nose heavy nor tail, thus aircraft balances perfectly.

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