CONCEPTUAL DESIGN OF CARGO AIRPLANE

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ABSTRACT

The main goal of the work is to design the military cargo aircraft that fulfils all the requirements. Current work includes weight estimation of an aircraft, selection of airfoil and suitable wing configuration, selection of tail, fuselage sizing and power plant selection. From the available details, weight estimation of the aircraft was started, by assuming the soldiers weight and baggage allowance, with the available empirical relations, weight estimation was done. There are many conditions to select feasible airfoil for the aircraft, with the consideration of design Mach number and design lift coefficient airfoil was selected, then for aircraft flight regime, suitable wing configurations was selected. Primary objective of fuselage is to accommodate the soldiers, crew members in cockpit, and cargo, to place all these in the fuselage, space was sized and proper aisle was given in between with reference to the military standards. Need of Empennage is to provide the stability for aircraft, by checking the required stability for aircraft, horizontal and vertical tail was sized and suitable configuration was selected from the historical trends and requirement. In the design stage feasible engine for the aircraft was selected.

KEYWORDS*: Cargo wing; empennage; fuselage; stability*

1.0 INTRODUCTION

Military cargo transport aircraft is a fixed wing type of the aircraft, which is used to carry soldiers, pallets, guns, jeeps and armaments. During the war time, required things can be carried from one airbase to another. Design for cargo aircraft is different from commercial aircraft, to support loading and unloading of cargo, T- tail configuration is used and since, cargo planes will be landed on countered surface, many number of wheels will be incoporated. In this work the cargo plane is intended to have the following characteristics such

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as, empty mass of 16000 kg, cruise velocity of 120 m/s, ceiling of 10 km, take-off and landing distance of 1050 m, range of 1850 km, loiter for 30 min and should be able to carry payload of 18000 kg and 3 crew members.

2.0 WEIGHT ESTIMATION

Weight of an aircraft plays a vital role both aerodynamically and structurally. Total takeoff weight of aircraft is sum of empty weight, payload weight, crew weight and fuel weight, as expressed in Equation (1) as follows:-

$$W_{TO} = W_E + W_{PL} + W_C + W_f \tag{1}$$

Total takeoff weight can be estimated using the empirical relation (Raymer, 1992) given in Equation (2) as follow:-

$$W_{TO} = \frac{W_{PL}+W_C}{1-\left(\frac{W_f}{W_{TO}}\right)-\left(\frac{W_E}{W_{TO}}\right)} \tag{2}$$

2.1 Weight Break down

To estimate total takeoff weight, payload weight, crew weight, fuel weight and empty weight can be broken down and estimated as follows:

Payload weight:
Payload weight (W_{PL}) for military cargo transport aircraft can be broken down as, by considering 50 soldiers weighing around 80 kg per person, 80 kg baggage weight per soldier, two military jeeps weighing 2300 kg per jeep, one fully loaded pallet weighing 4540 kg, five doctors weighing 80 kg per person and 100 kg surgical instruments. Therefore totally payload weight is 17880 kg.

Crew Weight:
One pilot and two flight engineers add to total crew weight. Each crew are of 80 kg approximately. so, total crew weight (W_C) will be equal to 240 kg.

2.1.3 Fuel weight:

Fuel weight fraction $(\frac{W_f}{W_0})$ can be estimated using the relation as given in Equation (3) as follow:-

$$\frac{W_f}{W_0} = 1.06 \times \left(1 - \frac{W_7}{W_0}\right) \tag{3}$$

Weight fraction $(\frac{W_7}{W_0})$ is estimated through mission profile as shown in Figure 1, and expressed in Equation (4) as follow:-

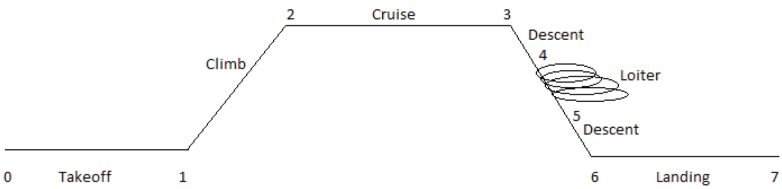

Figure 1 Mission profile considered for military cargo aircraft

$$\frac{W_7}{W_0} = \frac{W_1}{W_0} \times \frac{W_2}{W_1} \times \frac{W_3}{W_2} \times \frac{W_4}{W_3} \times \frac{W_5}{W_4} \times \frac{W_6}{W_5} \tag{4}$$

For military cargo plane, weight fractions are considered as 0.995 $(\frac{W_1}{W_0})$ for take-off, 980 $(\frac{W_2}{W_1})$ for climb, 0.990 $(\frac{W_4}{W_3},\frac{W_6}{W_5})$ for descend and 0.992 $(\frac{W_7}{W_6})$ for landing.

In cruise phase, weight fraction $\frac{W_3}{W_2}$ can be estimated using the relation shown in Equation (5) as follow:-

$$\frac{W_3}{W_2} = e^{\frac{-(R\times C)}{V\times(\frac{L}{D})max}} \tag{5}$$

where, R is the range of the aircraft which is 1850 km, C is the specific fuel consumption of the turboprop engine which is 0.5 in cruise, V is the cruise velocity of aircraft which is 432 kmhr^{-1}, $\frac{L}{D_{max}}$ is the max lift to drag ratio.

$$\frac{W_3}{W_2} = e^{\frac{-(1850\times0.5)}{432\times16}}$$

By estimating weight fractions at different segments, fuel fraction found to be
$\frac{W_f}{W_0} = 0.201$

Empty Weight:

Empty weight fraction can be estimated using the formulae given in Equation (6) as follow:-

$$\frac{W_E}{W_0} = A \times {W_0}^C \times K_{es} \tag{6}$$

where the constants A=0.93, C=-0.07, K_{es}=1 (fixed sweep).

Total Takeoff Weight Estimation:
By substituting the values obtained for crew weight, payload weight, empty weight fraction and fuel weight fraction, Equation (2) becomes,

$$W_0 = \frac{240 + 17640}{1 - 0.93 \times {W_0}^{-0.07} - 0.201}$$

Total takeoff weight W_0 is estimated from above equation by guessing W_0 term on right hand side, therefore take-off mass is 49322 kg.

3.0 CONSTRAINT ANALYSIS

In this work, four constraints are considered separately, for constraining thrust loading and wing loading. Four constraints are take-off, rate of climb, cruise at 6000 m altitude, landing at 80% and 100% weight (Jenkinson & Marchman, 2003).

The basic equation used for calculating thrust loading and wing loading is given by (Jenkinson & Marchman, 2003) as shown in an expression in Equation (7) as follow:-

$$\left(\frac{T}{W}\right)_{T-O} = \left(\frac{\beta}{\alpha}\right) \times \left[\left(\frac{q}{\beta}\right)\left\{\frac{C_{Do}}{W/_{S}} + K\left(\frac{n \times \beta^2}{q}\right)\left(\frac{W}{S}\right)\right\} + \left(\frac{1}{V}\right)\frac{dh}{dt} + \left(\frac{1}{g}\right)\left(\frac{dv}{dt}\right)\right] \tag{7}$$

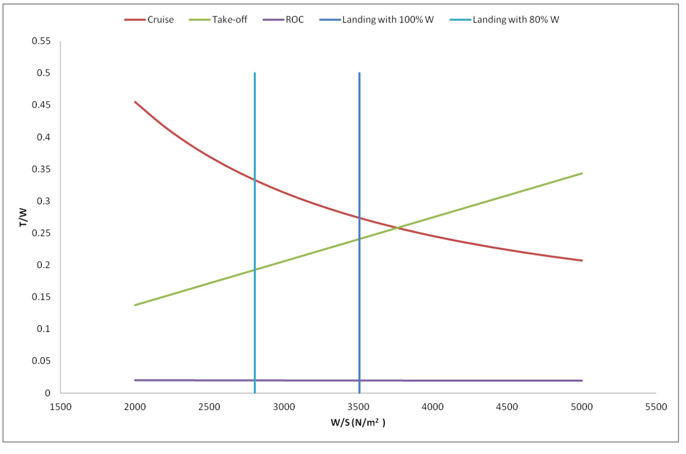

Figure 2 Constraint diagram of military cargo Aircraft

For landing the Aircraft, two constraints should be considered and those are, during emergency, as soon as aircraft take-off it has to land back it may be due to the improper weather conditions or enemy attack so landing weight is considered as same as the take-off weight and other constraint is after completing all the intended mission when it lands back, 80% of the take-off weight should be considered. With those two constraint considerations, design space was selected from Figure 2 between landing with 100% weight and landing with 80% weight, above the cruise line. In the design space, optimum design point is selected and required thrust to weight ratio and wing loading was found.

For Military cargo transport aircraft, highest wing loading with lowest possible thrust loading was selected as, wing loading of 3500 N/m^2 and thrust loading of 0.3.

4.0 AIRFOIL AND WING SELECTION

The selection of the airfoil depends on the Mach regime and the design coefficient of lift. Since, this work is intended to design a military cargo aircraft which typically fly in the subsonic regime (Raymer, 1992). The initial consideration in the selection of airfoil is the design coefficient of lift. This lift coefficient at which aircraft has maximum L/D. In the level flight lift is equal to weight, hence the required design coefficient of lift can be found as given in Equation (8) as follow:-

$$L = \frac{1}{2} \times \rho \times V^2 \times S \times C_l \qquad (8)$$

$$C_l = 0.4$$

The design coefficient of lift is 0.4 which lies within the drag bucket hence the six series airfoil is selected. Within the drag bucket, by incresing the C_l the co-efficient of drag remains minimum. NACA 64-415 airfoil is selected for aircraft. From the variation of t/c for the design Mach number. The design Mach number is 0.5 from the historical trends the t/c is 15% or 0.15. The design coefficient of lift is 0.4. NACA 64-415 is selected because of its favorable characteristics that fulfill the desired requirements of the design.

To calculate wing area, we can use an expression given in Equation (9) as follow:-

$$S = \frac{W_0}{\frac{W}{S}} \tag{9}$$

The W/S is calculated using empirical relation such as,
S = 138 m^2
To determine aspect ratio of the wing, an expression in Equation (10) is given, as follow:-

$$AR = \frac{b^2}{S} \tag{10}$$

$$AR = 9.2$$

Therefore, the wingspan is 35.66 m.
Taper wing will reduce the induced drag and span wise lift distribution is closer to elliptical lift distribution. By giving taper wing will automatically have slit sweep.

From the empirical relation the taper ratio of the wing is 0.65 and thus, the following expressions in Equations (11) and (12) are sought, as follow:-

$$\lambda = 0.65 = \frac{C_t}{C_r} \tag{11}$$

$$C_r = \frac{2 \times b}{A(1+\lambda)} \tag{12}$$

$$C_r = 4.7\ m$$

$$C_t = 3.05\ m$$

From the historical trends, it was determined that aileron span may be 50-90% of the wing span and 15-20% of wing chord (Corke, 2003) hence, for this aircraft 50% of wing span and 20% of the wing chord was choosen and equal to,
Aileron span = 8.75 m each side
Aileron chord = 0.764 m
Area of aileron = 6.68 m^2
According to the Burgess rule, rib spacing is always one-fifth of the chord of the plane. Throughout the span 10 ribs are used. Skin thickness is equal to 0.99 mm.

5.0 FUSELAGE AND LANDING GEAR SIZING

Do determine the length of fuselage for military cargo class of the aircraft, empirical relations are selected and substituted in the empirical relation.

W_o = 49322 kg = 108736.39 lb o=(overall take-off weight of the aircraft)

$L = 0.23 \times 108736.39^{0.5}$

L = 75.842 ft = 23.5 m

$\therefore$ L/D = 5

Thereby; D = 4.7 m (Maximum diameter of the fuselage)

In the military Aircraft, it is intended to position pallets, jeeps and seats for soldiers and in the initial estimations, position of landing gear retraction box, wing box and APU was neglected. Total length of the fuselage is 23 m it includes cockpit, ramp and cargo compartment. Here one pallet and two military jeeps are carried as cargo, from the MIL standards pallet dimension is 2.64 × 2.13 × 2.91 m, having a mass of 4540 kg and military jeep dimension is 4.5×2.16×1.8 m, having a mass of 2300 kg.

MIL standards say that, to make loading and unloading easy in cargo aircraft, fusealge should be 1.5 m height from the ground and between pallets and jeep 0.15 m of aisle should be given, totally there will be three doors in the fuselage one is for soldiers, one is for pilot and one more will be for cargo loading and unloading. For soldiers and pilot 0.8 m of door should be provided.

More convenienetly Z shaped stringers are used for the design, from the Denis Howe's principle web thickness of the stringer should be same as the skin thickness, generally skin thickness will be 0.99 mm and according to Denis Howe's principle web height will be thirty nine times the thickness, flange width will be sixteen times the thickness. Therefore, total height of the stringer is 39.6 mm and length of the flange is 15.84 mm.

From Barlow equation, an expression as given in Equation (13) is referred to as follow:-

$$P_B = 0.875 \times \left(\frac{2 \times Y_{pt}}{D}\right) \tag{13}$$

Considering aluminum is used as skin, so yield strength will be 269 MPa

$$P_B = 0.875 \times \left(\frac{2 \times 39}{185}\right)$$
$$P_B = -0.36\, psi \text{ or } 2.485 \text{ kPa.}$$

Therefore, 2.48 kPa collapsing pressure will be acting in the airborne flight and in the ground.

Tri cycle arrangement of the landing gear is selected because, main wheels will be aft of the cg and nose wheel will be forward of the cg so aircraft will be stable on the ground. Main tire carries 90% of the total weight of the aircraft, whereas nose landing gear carries only 10% of the total aircraft weight, nose landing gear is used only steering purpose.

$$D_{wheel} = 1.63 \times 49322^{0.315}$$
$$D_{wheel} = 1.15\, m$$
$$W_{wheel} = 0.1043 \times 49322^{0.48}$$
$$W_{wheel} = 0.4\, m$$
$$W_{wheel} = P \times A_p$$

Therefore, an expression in Equation (14) is sought, as follow:-

$$A_p = 2.3\sqrt{W_d}\left(\frac{d}{2} - R_r\right) \tag{14}$$
$$A_p = 2.3\sqrt{0.4 \times 1.15}\left(\frac{1.15}{2} - 0.5\right)$$
$$A_p = 0.117\, m^2$$
$$W_{wheel} = 1.6 \times 10^6 \times 0.117$$
$$W_{wheel} = 19082 \text{ kg/wheel}$$

Oleo shock strut type of the shock absorber was selected, because efficiency of the oleo is maximum compared to other type of strut (say 0.75 to 0.9). Total length of oleo including the stroke distance and fixed portion of the oleo will be approximately 2.5 times the stroke. Historical trends says that stroke in inches approximately equal to the vertical velocity at touchdown, most of the aircraft requires 10 ft/s vertical velocity whereas, 5 ft/s will be very bad landing.

Length of the stroke = 10 inch
Length of oleo = 2.5×10
Length of oleo = 0.7 m;

Therefore, an expression given in Equation (15) is sought, as follow:-

$$D_{oleo} = 1.3\sqrt{\frac{4 \times L_{oleo}}{P\pi}} \tag{15}$$

Typically, oleo type of shock absorber has 1.27×10^7 pa of pressure,

$$D_{oleo} = 0.04\sqrt{L_{oleo}}$$
$$D_{oleo} = 0.03\ m$$

6.0 EMPENNAGE SIZING

Primary objective of the empennage is to provide stability for the aircraft, with that reference airfoil will be selected and later on control surface for the empennage will be sized and airfoil for those control surfaces will be selected. Selection of tail depends on the aircraft requirement. Since, Military cargo transport is intended to design, T-tail arrangement is preferred over the other configurations because fuselage will be closer to the ground and when ramp is opened in the aircraft it allows the direct loading and unloading of jeeps, pallets and armaments. There are two options for positioning the tail, aft position is preferred over the canard due to ease of construction and historical trends for military cargo aircrafts have been used aft position.

6.1 Horizontal tail design

Taper ratio, aspect ratio and tail volume coefficient was found from the historical trends,

A.R. = 4

λ=0.45

In general area ratio varies from 0.25 to 0.35,

$$\frac{S_{HT}}{S_W} = 0.3$$

During the wing design, area was calculated and found to be 138 m^2,

$$S_{HT} = 41.4\ m^2$$
$$A.R. = \frac{b^2}{S_W}$$

b = 12.86 m

Tapered horizontal tail was intended to design because, it has a influence on the tail efficiency, aircraft stability and control, performance, aircraft weight and centre of gravity.

Therefore, an expression given in Equation (16) is sought, as follow:-

$$C_r = \frac{2 \times S}{b(1+\lambda)} \tag{16}$$

$$C_r = \frac{2 \times 41.4}{12.86(1+0.45)}$$

$$C_r = 4.4\ m$$

$$\lambda = \frac{C_t}{C_r}$$

$$C_t = 2\ m$$

Symmetric or negative cambered airfoil should be used for horizontal tail because, tail should be able to create positive and negative lift. For the ease of manufacturing of horizontal tail symmetric airfoil will be best. So, NACA symmetric airfoil will be selected. During the selection of airfoil, following points were considered, horizontal tail should never stall and wing must stall before the tail, symmetric airfoil should be selected in such a way that it should behave in similar manner at positive and negative angle of attack, lift coefficient should be as large as possible, overall drag should be less and pitching moment should be minimum.

Lift produced by NACA 0020 airfoil is lesser than the NACA 0022 airfoil, where drag produced is almost same, but pitching moment is less in the NACA 0020 airfoil. Hence, NACA 0020 airfoil was selected for horizontal tail.

6.2 Vertical tail design

Primary functions of vertical tail are to provide directional stability and directional trim, for vertical tail also symmetric airfoil was preferred. A military cargo transport aircraft is a multiengine aircraft, hence one engine inoperative, vertical tail should be strong enough to control the aircraft. From the historical trends,

A.R. =0.9

λ =0.8

In general area ratio varies from 0.15 to 0.25, and this yield

$$\frac{S_{VT}}{S_W} = 0.2$$

$$S_{VT} = 27.6\ m^2$$

$$A.R. = \frac{b^2}{S}$$

$$b_{VT} = 4.9\ m$$

The objective of vertical tail airfoil is to produce stability and control, it is not a lifting surface. Following points to be considered while selecting airfoil, airfoil should be symmetric because, it should behave in similar passion at positive and negative angle of attack and to reduce the structural weight of empennage, thickness of airfoil should as less as possible.

As mentioned above primary objective of vertical is directional stability ($C_{n\beta} > 0$),

Empirical expression for directional stability (Nelson, 2000) is as given in Equation (17) below:-

$$\eta_V\left(1 + \frac{d\sigma}{d\beta}\right) = 0.724 + 3.06\left(\frac{S_V/_S}{1+cos\lambda_{\frac{c}{4}}}\right) + 0.4 \times \left(\frac{Z_w}{d}\right) + 0.009 \times AR_W \tag{17}$$

$$\frac{d\sigma}{d\beta} = 1.6;$$

Therefore, the following expression as given in Equation (18) is considered, as follow:-

$$C_{n\beta} = -K_n \times K_{R1} \times \left(\frac{S_{fs} \times l_f}{S \times b}\right) + V_V \times C_{L_{\alpha V}} \times \eta_V \times \left(1 - \frac{d\sigma}{d\beta}\right) \tag{18}$$

$$C_{L_{\alpha V}} = 0.00129/deg$$

0.00129 is the minimum lift required from the vertical tail, based on this, thinner standard airfoil should be selected. By checking the historical trend for military cargo transport aircraft vertical tail configuration and minimum lift required NACA 0009 airfoil was selected. Whereas, NACA 0009 airfoil produces $C_{L\alpha V}$ of 0.09 per deg, which is more than the minimum lift required so NACA 0009 airfoil can be used for vertical tail.

6.3 Control surface sizing

Elevator should be long enough to produce the desired lift, so elevator chord to tail chord ratio can be taken as,

$$\frac{C_E}{C_H} = 0.35 \qquad (19)$$

MAC of horizontal tail is considered,

$C_E = 0.35 \times 3.2$
$C_E = 1.12\ m$

For ease of manufacturing,

$$\frac{b_E}{b_H} = 1 \qquad (20)$$

From the empirical relations,

$$\frac{C_R}{C_V} = 0.38 \qquad (21)$$

Whereas, MAC of vertical tail is 4.16 m,

$C_R = 1.6\ m$

Rudder should be long enough to control the aircraft, so leaving a gap of 1 m for hinges either side, so span of a rudder will be 4.64 m.

7.0 ENGINE SELECTION

With the constrained altitude and Mach no, two types of engines can be used such as turboprop and piston prop engines. But specific weight of piston prop is high compared to turboprop engines. These engines makes the less noise than jet engines, also costs less than jet engines. It has the least environmental chemistry impact from fuel consumption vs payload. They provide high thrust at low speeds.

The power required for an aircraft is calculated at an altitude of 8000 meters and maximum cruise velocity of 120 m/s.

$$P_{required} = T_{required} \times V \qquad (22)$$

$$P_{required} = \frac{1}{2}\rho V_{\infty}^{3} S C_{D0} + \frac{W_0^2}{\frac{1}{2}\rho V_{\infty} S}\left(\frac{1}{\pi e AR}\right) \tag{23}$$

$= 6799 kW$

$= 7000\ kW\ (approx).$

For the required power, Europrop international TP400-D6 is the power plant selected. Coincidentaly it is used in Airbus A400M atlas military transport aircraft. The TP400 is the most powerful single-rotation turboprop.

The positioning of prop-driven engine on the wing in twin engine configuration often results in most attractive design from a structural and aerodynamic point of view. It can be either under wing configuration or over wing configuration, engine will induce a flutter to wing structure. While an under wing configuration will not have such negative impact. So, that engine placement under wing is been selected where the positioning of wing is at the 35% over the length of wing.

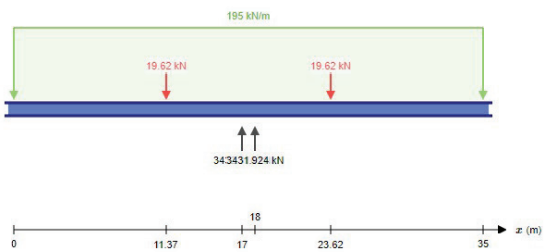

Figure 3 Free body diagram of wing, engine weight is considered at 35% of wing length

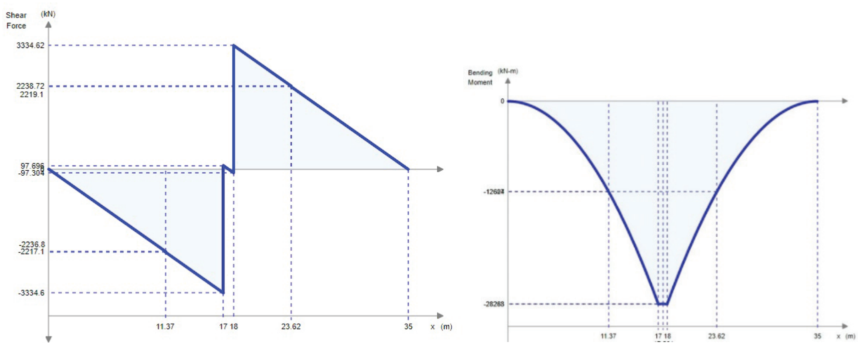

Figure 4 Shear force and bending moment diagram of wing, where engine weight is considered at 35% of wing length

8.0 EMPTY MASS ESTIMATION

Wing weight for cargo/transport aircraft can be estimated using the empirical relation (Raymer, 1992)

$$W_{Wing} = 0.0051 \times (W_{dg} \times N_z)^{0.557} \times S_W{}^{0.649} \times A^{0.5} \times (\frac{t}{c})^{-0.4}_{root} \times (1+\lambda)^{0.1} \times (cos\Lambda)^{-1.0} \times S_{csw}^{0.1} \quad (24)$$

$$W_{Wing} = 2100\ kg$$

Weight of horizontal tail structure can be estimated using the relation

$$W_{H.T} = 0.0379 \times K_{uht} \times (1+\frac{F_W}{B_h})^{-0.25} \times W_{dg}{}^{0.639} \times N_Z^{0.10} \times S_{ht}^{0.75} \times L_t^{-1.0} \times K_y^{0.704} \times (cos\Lambda_{ht})^{-1.0} \times A_h^{0.166} \times (1+\frac{S_e}{S_{ht}})^{0.1} \quad (25)$$

$$W_{H.T} = 770\ kg$$

Vertical tail mass can be estimated through use of empirical relation

$$W_{V.T} = 0.0026 \times (1+\frac{H_t}{H_v})^{0.225} \times W_{dg}{}^{0.556} \times N_Z^{0.536} \times S_{vt}^{0.5} \times L_t^{-0.5} \times K_Z^{0.875} \times (cos\Lambda_{vt})^{-1.0} \times A_v^{0.35} \times (\frac{t}{c})^{-0.5}_{root} \quad (26)$$

$$W_{V.T} = 100\ kg$$

Mass of the fuselage can be computed with the use of formulae

$$W_{fuselage} = 0.3280 \times k_{door} \times K_{Lg} \times (W_{dg} \times N_z)^{0.5} \times L^{0.25} \times S_f^{0.302} \times (1+K_{ws})^{0.04} \times (\frac{L}{D})^{0.10} \quad (27)$$

$$W_{fuselage} = 3600\ kg$$

Landing gear mass can be estimated through the use of relations

$$W_{landing\ gear} = W_{main\ landing\ gear} W_{nose\ landing\ gear} \quad (28)$$

$$W_{main\ L.G} = 0.0106 \times k_{mp} \times W_l{}^{0.888} \times N_l^{0.25} \times L_m^{0.4} \times N_{mw}^{0.321} \times N_{mss}^{-0.5} \times V_{stall}^{0.1} \quad (29)$$

$$W_{nose\ L.G} = 0.032 \times k_{np} \times W_l{}^{0.646} \times N_l^{0.2} \times L_n^{0.5} \times N_{mw}^{0.45} \quad (30)$$

$$W_{L.G} = 2220\ kg$$

Mass of engine controls can be calculated by the formulae

$$W_{engine\ \ controls} = 5.0N_{en} + 0.80L_{ec} \quad (31)$$

$$W_{engine\ \ controls} = 25\ kg$$

Starter's mass can be estimated using the relation

$$W_{starter} = 49.19 \times \left(\frac{N_{en} \times W_{en}}{1000}\right)^{0.541} \quad (32)$$

$$W_{starter} = 80\ kg$$

Mass of the fuel system can be calculated with the use of empirical relation

$$W_{fuel\ system} = 2.405 \times V_t^{0.606} \times \left(1 + \frac{V_i}{V_t}\right)^{-0.1} \times \left(1 + \frac{V_p}{V_t}\right) \times N_t^{0.5} \quad (33)$$

$$W_{fuel\ system} = 628\ kg$$

Total mass of avionics can be approximated by the formulae,

$$W_{avionics} = 1.73 \times W_{uav}^{0.983} \quad (34)$$

$$W_{avionics} = 500\ kg$$

The mass of each troop seat is approximated as 5 kg. There are totally 50 seats so, the total weight of troop seats will be equal to,

$$W_{troop\ seats} = 250\ kg$$

There are two turboprop engines which weigh 2000 kg each. So, the total engine mass is equal to,

$$W_{engine} = 4000\ kg$$

Total mass of the equipment's is given by the relation (Raymer, 1992),

$$W_{equipment} = 4.509 \times N_c^{0.541} \times K_r \times K_{tp} \times N_{en}\left(L_f + B_W\right)^{0.5} \quad (35)$$

$$W_{equipment} = 100\ kg$$

Therefore, in total empty mass of cargo transport aircraft is 14373 kg.
Centre of gravity is more sensitive to the weight of the aircraft, depending upon the aft or forward cg cargo can be positioned. To calculate the cg of aircraft body co-ordinate system is selected.

$$Aircraft_{cg} = \frac{\sum W_w x_w + W_f x_f + W_{HT} x_{HT} + W_{VT} x_{VT} + W_E x_E + W_L x_L}{\sum W_w + W_f + W_{HT} + W_{VT} + W_E + W_L} \quad (36)$$

Centre of gravity (from leading edge of wing) = 3.7 m

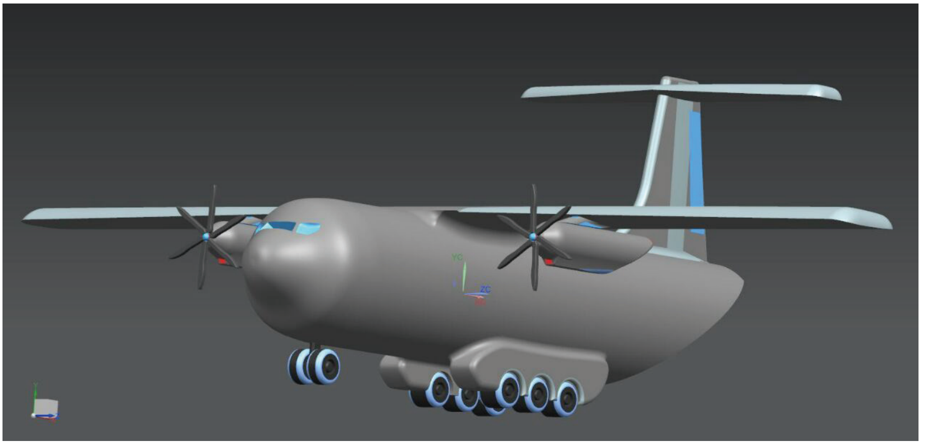

Figure 5 Conceptualy designed military cargo aircraft

CONCLUSION

Initial constraints of the project are empty mass should be of 16000 kg, cruise velocity of 120 m/s, ceiling of 10 km, range of 1850 km, take-off and landing distance of 1050 m, loiter for 30 min, 18000 kg payload and 3 crew members. To sustain the weight and provide sufficient lift for the aircraft NACA 64-415 airfoil with 9.2 aspect ratio for wing was selected, from the emperical relations control surface sizing was done, from the Burgess rule, rib spacing was done. To accommodate the soldiers, crew members, armaments and cargo, fuselage fitness ratio of 5 was selected. For stability requirements NACA 0020 airfoil was selected for horizontal tail and NACA 0009 was selected for vertical tail. From the constraint analysis, power required for aircraft was found to be 7000 kw, for this power matching, TP-400 engine was selected. After sizing the each component of aircraft, empty mass was found to be 14373 kg and cg was 3.7 m from leading edge of the wing.

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