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Conceptual Design and Sizing of an Amphibian Transport Aircraft

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This paper presents results of a study related to conceptual design and sizing of an amphibian passenger transport aircraft for connecting some coastal cities and islands in south India. The operating requirements for the seaplane were based on a recently published feasibility study for the island-coast connecting market for transport and tourism purposes in South India. Based on a study of the design data related to existing seaplanes, DHC-6 Twin Otter was taken as a baseline aircraft, and redesigned to meet the requirements. The paper describes the procedure used for arriving at the values of key parameters of the aircraft, sizing of the various components, and estimation of the drag and weight breakdown. It was seen that the requirements on Landing Distance and Missed Approach Gradient dictated the Wing and Power loading, and the aircraft met or exceeded all the design requirements that were specified.

Nomenclature

$\alpha_{deadrise}$ = deadrise angle (deg)
b_H = hull beam width (m)
$b_W, \bar{C}_W$ = wing span and mean aerodynamic chord (MAC) (m)
c_{VT}, c_{HT} = volume coefficient of vertical and horizontal tail
E = loiter time (s)
γ_{MA}, γ_{SSCG} = missed approach and second segment climb gradient
K = trailing-edge (TE) nondimensional angular deflection rate
L_{fus} = Fuselage length (m)
l_H, l_V = distance between wing c/4 MAC and HT,VT quarter chord point (m)
l_{H+F}, h_{H+F} = hull+fuselage length and height (m)
L/D = lift to drag ratio
n_{eng} = Number of engines
P/W = power loading, power to weight ratio (W/g)
R = range (m)
sfc = specific fuel consumption (s^{-1})
S_{HT}, S_{VT} = surface area of horizontal tail and vertical tail (m^2)
$Sref$ = reference area, wing area (m^2)
T/W = thrust loading, thrust to weight ratio (N/N)
V = velocity (cruise) (m/s)
W_0 = maximum take-off weight (kg)
W_i = weight at mission segment *I* (kg)
W/S = wing loading, weight to wing area ratio (N/m^2)
X_{CG} = longitudinal location of aircraft CG from nose (m)

I. Introduction and Market Survey

SEAPLANES were one of the first planes to be designed, and were very important during the early days of aviation. This was mainly because of lack of sufficiently large airports. Seaplanes used an already established

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harbor for operations which saved land space and costs. After WW II, the number of airports increased, and their runway lengths became longer, due to the increase in the performance of land planes. However, seaplanes still find their use in areas where it is difficult to build runways or for purposes like firefighting.

India sits astride the major sea routes of the world with a coastline of 7,516 km with a total of 1,197 island territories in the Bay of Bengal and the Arabian Sea. India has 12 major ports, and nearly 200 minor and intermediate ports spread all along the country's vast coastline[1]. India has an Exclusive Economic Zone (EEZ) of 2.01 million sq. km., and 90 per cent by volume and 77 % of total value of India's trade comes from the seas. The resource rich EEZ provides 68 per cent of its oil production and fish production of 2.82 million tons. In addition, the entire import of oil and gas comes by the sea. India's economy, and therefore its development is crucially dependent on the sea on account of the critical role of maritime trade as well as oil and gas, fisheries and other mineral resources.

India has a very large potential to develop a market for an amphibian aircraft due to its large coastline. There is a recent trend to promote tourism between its coastal regions and various islands. Currently, ships are the only reliable form of transport to many Indian islands like Lakshadweep and Andaman and Nicobar. The transport times on such journeys can be anywhere between two hours to two days ! Furthermore, most of these islands are incapable of having a proper runway for land aircraft operations. An amphibian aircraft would improve accessibility and travel times to such areas by a huge margin and thus be an ideal choice for transport as well as tourism purposes.

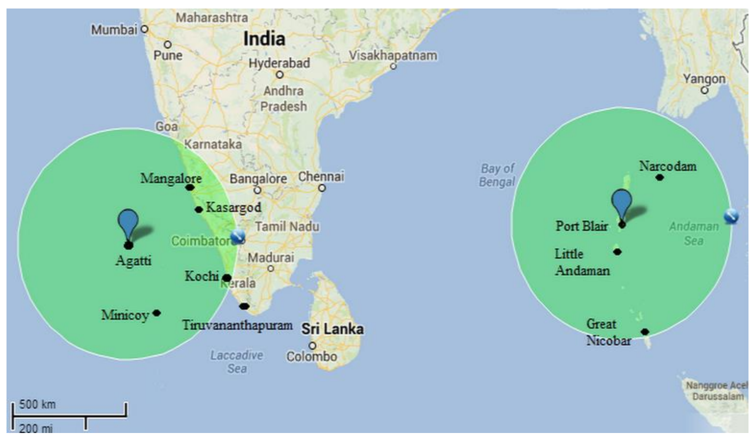

Figure 1. The two main cities, and other points to be connected

The only form of transport in Lakshadweep and Andaman and Nicobar islands currently is by ferries which typically travel at 20-30 knots. They take anywhere from 5-10 hours to travel from one island to another given in Fig 1. Operating an amphibian aircraft on these routes will reduce the same travel time to maximum of 1 hour while also eliminating the inconvenience of sea travel.

This kind of tourism service has been successfully implemented in the Maldives by two companies Maldivian Air Taxi and Trans Maldivian using about 50 Twin-Otters along routes up to 320 km. A recent survey on feasibility of such a service in Kerela[2] suggested a few locations where seaports can be established. The data for Andaman and Nicobar and Lakshadweep islands has been modeled along the lines of Maldives with distances to popular tourist locations greater than 100 km from the nearest airport considered. As shown in Fig. 1, a range of 500 km can be considered sufficient for the present requirements since it covers all major tourist locations in Kerala, Lakshadweep and Andaman and Nicobar islands.

A. Initial Requirements

Data from European markets was also used as a starting point to generate initial requirements. According to a survey by FUSETRA[3], most operational seaplanes have 7-9 or 17-19 passenger seats. It was also found that the DHC-6 Twin Otter contributes about 30-35% of the total aircraft operational as a seaplane by attaching floats to it. Hence this aircraft was chosen as the baseline aircraft for the present study, to meet the design requirements listed in Table 1.

Table 1. Design requirements.

Requirements	**Current Design**	**DHC-6-300[4]**
Regulatory Requirements	FAR 25	FAR 23
Harmonic Range (km)	500	180
No. of Passengers	19	20
Max. Payload (kg)	1805	1945
Operating Altitude (m)	3048	3048
Takeoff and Landing Distance (m)	950	500
Cruise Speed (m/s)	100	77
Max. Speed (m/s)	120	90

The requirement of takeoff and landing distance is dictated by the runway length available at Agatti aerodrome in Lakshadweep islands on a standard day. The aircraft was also required to possess certain other capabilities and features, such as ability to operate from a gravel runway, a pressurized fuselage for passenger comfort, and possibility of a quick-change between all cargo, all passenger and mixed configuration. Another key design requirement was to ensure that the aircraft is unsinkable in the desired operating condition.

B. Design Choices

Seaplanes can be categorized into two categories; those having floats and those having hulls. Amphibians are seaplanes with a landing gear, which enables them to take off and land on sea or land. Floats are easy to manufacture and are the easiest option when converting a landplane to a seaplane. Generally, a land based plane is converted into a seaplane version by attaching floats, as in the case of Twin-Otter. However, in the present study, the objective was to design an amphibian aircraft right from the start. Amphibian aircraft with hulls have better takeoff, landing and drag performance, and they also provide better sea state capability.

II. Initial Design and Sizing

The first estimate of design gross weight was obtained using the methodology for initial sizing described in this section. The optimal lift to drag ratio estimated after one design cycle was used to calculate all the values for the next cycle. This procedure is iterated until all the values converge. This design methodology was also applied to the Beriev 112, which is a 27 seater amphibian aircraft. The values of critical parameters obtained by this methodology were within 5% of the actual values[5].

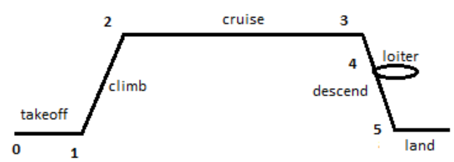

Figure 2. Design Mission Profile

A. Takeoff Weight Estimation

Initial take-off weight estimation was carried out as per the methods given by Raymer[6]. The maximum take-off weight W_0 of any aircraft can be estimated using Eq. 1 as:

$$W_0 = W_{crew} + W_{payload} + W_{fuel} + W_{empty} = \frac{W_{crew} + W_{payload}}{1 - (W_{fuel}/W_0) - (W_{empty}/W_0)} \tag{1}$$

The Empty weight fraction for a seaplane can be estimated using Eq. 2 as:

$$W_e / W_0 = 1.05 W_0^{-0.05} \tag{2}$$

For a commercial transport operating under VFR (Visual Flight Rules), FAA has mandated 30 minutes of cruise equivalent additional fuel for daytime operations.

The fuel fraction estimation was carried out by assuming a mission profile as shown in Fig. 2. The weight fractions for each segment were calculated in order to obtain the fuel weight fraction. The take-off, climb and landing weight fractions are approximately obtained from historical data order to obtain the fuel weight fraction. The remaining weight fractions are given as:

Cruise fraction:

$$W_3 / W_2 = \exp\left(\frac{-R \cdot sfc}{V(L/D)}\right) \tag{3}$$

Loiter fraction:

$$W_4 / W_3 = \exp\left(\frac{-E \cdot sfc}{(L/D)}\right) \tag{4}$$

Table 2. lists the assumptions made during estimation of Take-off weight. (L/D) during cruise converged to a value of 16.9 after design iterations and was used in the final weight breakdown shown in Fig. 3.

Table 2. Assumptions in Initial Sizing

Parameter	Value
Specific Fuel Consumption (sec^{-1})	0.45
(L/D) cruise	16
Loiter Time (mins)	30
Reserve Fuel (cruise mins.)	30
Unusable fuel (% of total)	06

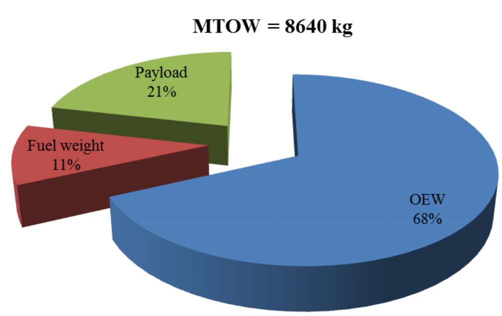

Figure 3. Weight breakdown

B. Wing and Power Loading

Wing loading is determined by the takeoff parameter plot given in Raymer[6]. A first estimate of the power loading (in Watt/gm.) from statistical data for seaplanes was obtained using Eq. 5:

$$P/W_0 = 0.043 V_{max}^{0.23} \quad (5)$$

Another method to find initial estimates of power loading is to find the cruise power loading and match it to takeoff conditions. For this, the equivalent thrust loading is first calculated and then converted to power loading using Eq. 6:

$$(T/W) = (\eta_p / V)(P/W) \quad (6)$$

The propeller efficiency in cruise was assumed to be 0.85. Cruise thrust loading (in N/kg) can be calculated using Eq. 7 as:

$$(T/W)_{cruise} = g/(L/D)_{cruise} \quad (7)$$

It is then adjusted for Take-off condition using Eq. 8:

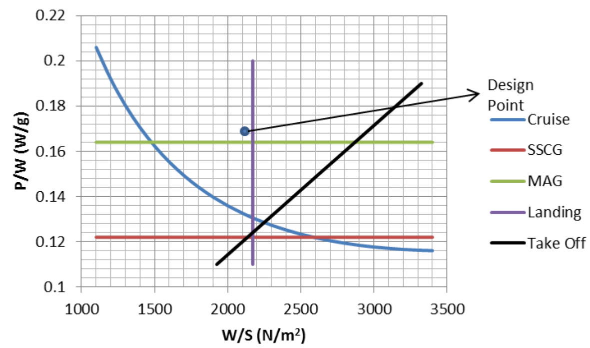

Figure 4. Constraint Analysis diagram

$$(T/W)_{takeoff} = (T/W)_{cruise}(W_2/W_1)(T_{takeoff}/T_{cruise}) \quad (8)$$

The value of $T_{cruise}/T_{takeoff}$ was chosen as 0.6 based on historical data for propeller aircraft.

Another constraint on power loading for an amphibian aircraft arises from the requirements on missed approach gradient and second segment climb gradients specified by FAR 25 during land operation. The values can be estimated using Eq. 9 as:

$$(T/W) = \frac{n_{eng}}{n_{eng}-1}\left(\frac{1}{(L/D)} + \gamma\right) \quad (9)$$

Where $\gamma = \gamma_{MA} = 0.021$ during missed approach and $\gamma = \gamma_{SSGC} = 0.024$ during 2nd stage climb for twin engine aircraft. (L/D) during missed approach and second segment climb were approximated at 80% and 70% of the cruise (L/D) value, respectively. The velocity is required to be $\leq 1.5V_{stall}$ during missed approach and $\geq 1.2V_{stall}$ during second segment climb. The highest P/W is chosen in the calculations that follow.

The wing loading required to meet the take-off requirement was estimated using Eq. 10:

$$(W/S) = (TOP)\sigma C_{L_{TO}}(P/W) \quad (10)$$

Another upper limit on wing loading is placed by the landing constraints using Eq. 11:

$$s_{landing} = 1.67\frac{(W/S)}{\sigma C_{L_{max}}} + s_{approach} \quad (11)$$

Where approach distance was chosen as 1000 feet for a 3 degree glideslope. The constraint diagram indicating the permissible values of P/W and W/S meeting all the above mentioned requirements is shown in Fig. 4. The design point indicates the values of these parameters that were finally decided which are listed in Table 3, along with the values of other parameters.

Table 3. Converged Wing and Power loading

Parameter	Value
Take off parameter	400
Take-off lift coefficient	1.50
Maximum lift coefficient	1.75
$(P/W)_{design}$ (W/g)	0.164
$(W/S)_{design}$ (N/m^2)	2171

C. Airfoil Selection and Wing Sizing

For initial airfoil selection, the wing lift coefficient was assumed to be equal to the airfoil lift coefficient, using Eq. 12:

$$C_l = \frac{1}{q}\left(\frac{W}{S}\right) \quad (12)$$

Wing area was found out based on the take-off gross weight and the design wing loading. Assuming an initial

Table 4. Wing geometry

Parameter	Value
Dynamic pressure (q) (N/m^2)	4545
Design lift coefficient	0.48
Area (m^2)	39.0
Aspect Ratio	10
Span (m)	19.75
Chord (m)	1.975

aspect ratio for the rectangular wing, the span and chord was calculated. The results are summarized in the Table 4.

Based on airfoil drag polar and the historical trend for thickness ratio with Mach number, the *NACA 4415* airfoil was chosen. It has a wide drag bucket allowing low drag operations from C_l values from 0 to 1.2 which will allow for low drag high lift operations during takeoff.

A rectangular unswept wing planform was chosen for design and structural simplicity. The wing was unswept since cruise will be at around Mach 0.3. A high wing configuration was chosen since it gives better protection from water spray as well as better roll stability and eliminates the need for a dihedral angle. This also helps to reduce the landing gear weight by allowing for a shorter strut.

D. Fuselage Sizing

Fuselage cross section was decided based on two abreast seating arrangement, keeping in mind that the primary aim of the aircraft was for tourism. A seat pitch of 32 inches and a seat width of 19 inches were considered, as per the usual norm for comfortable cabin spacing, which resulted in a passenger cabin length of 8.5 m. This, along with aisle width and other corrections resulted in a cabin width of 1.5m based on DHC-6-300 values[4].

L_{fus} was estimated to be 16.5 m using Eq. 13 for flying boats[6]:

$$L_{fus} = 0.439 W_0^{0.40} \tag{13}$$

The luggage area has been kept at the rear of the cabin to utilize the excess length available.

A hull of an amphibian aircraft is inherently unstable since the center of buoyancy is below the center of gravity, resulting in a destabilizing rolling moment during water operations. To stabilize this, sponsens were used, which are claimed to be more effective than wing tip floats[7]. The cross-sectional view of the aircraft is shown in Fig. 5.

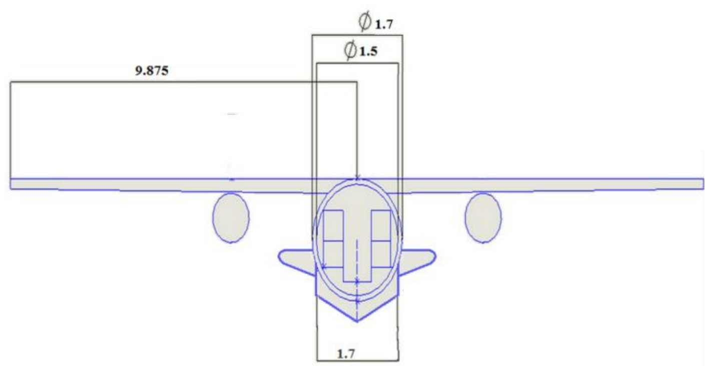

Figure 5. Cross-Sectional view

E. Tail Sizing

The tail arm was approximated at 50% of L_{fus} as an initial estimate for an aircraft with engines mounted on the wings. The equations for horizontal and vertical tail sizing given the volume coefficients are:

$$S_{VT} = c_{VT} b_W S_W / L_{VT} \tag{14}$$

$$S_{HT} = c_{HT} \bar{C}_W S_W / L_{HT} \tag{15}$$

The estimates for tail volume parameters and aspect ratios were chosen based on historical values listed in Raymer[6]. The results of tail sizing are listed in Table 5.

Table 5. Initial tail sizing parameters

Parameters	**Values**
HT volume- c_{HT}	0.7
VT volume- c_{VT}	0.06
$L_{VT}=L_{HT}$ (in m)	9
S_{VT} (in m^2)	5.14
S_{HT} (in m^2)	6.00
HT Aspect Ratio	4
VT Aspect Ratio	2

F. Weight Breakdown and Center of Gravity Estimation

Raymer[6] and Nicolai and Carichner[8] have suggested methodologies for arriving at component weight breakdown of an aircraft. The methodology by Nicolai and Carichner[8] is more appropriate for conventional passenger transport aircraft of all-metal construction. It does not contain any data that is specifically applicable to seaplanes or amphibians. In the present study, the methodology suggested by Raymer[6] was used, since it relies more on the empty weight fraction, for which formulae for this type of aircraft are listed. The major component weights were then used to calculate the aircraft center of gravity. Wing placement along the fuselage was varied until the obtained center of gravity matched with the desired one at around 30% wing mean aerodynamic chord. The converged value of X_{CG} was 6.56 m from nose.

G. Hull Sizing

Initial hull sizing is required to proceed with drag estimation to obtain the converged design. Most seaplane hulls have a V shape to reduce water impact loads. The angle the V makes is called the deadrise angle ($\alpha_{deadrise}$), which is related to the stall speed as[6]:

$$\alpha_{deadrise} \cong 1.1185 V_{stall} - 10 \quad (16)$$

Eq. 16 gives a high value for the deadrise angle generally found only at the chine. A lower value was selected so as to allow the deadrise angle to gently increase at the chine.

Hull beam was determined by external fuselage width. A step in the hull helps to break the water suction on the after body of the hull and is necessary for takeoff. However, during flight, it serves little purpose while contributing significantly to drag. This can be solved by fairing the step using a flap or any similar mechanism. For initial sizing, step height should be 5% of the hull beam and it should be located at an angle of 10-20 degree behind the center of gravity of the aircraft. The forebody length was determined in this manner.

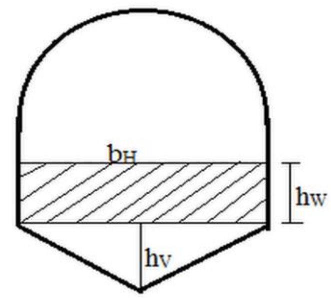

Figure 6. Hull submerged height determination

Detailed hull design was carried out according to procedure given by Smith and Allen[9]. The calculations from Ref. 9 consider the design for the entire fuselage hull geometry. For a forebody to beam ratio of 4.12, the hull total length to beam ratio of 7 was found satisfactory. This gave the hull afterbody length to be 5.24 meter. Thus, the 4.6 meter of the fuselage will not have a hull. The hull fuselage surface area, volume and maximum cross section are given by[9]:

Hull Fuselage total surface area:

$$S_{H+F} = 2l_{H+F}(h_{H+F} + b_H)c_S \quad (17)$$

Hull Fuselage max cross-sectional area:

$$A_{H+F} = h_{H+F} b_H c_A \quad (18)$$

Where c_s=0.65 and c_A=0.9 are constants dependent on forebody length to hull beam ratio[7].

Table 7. Hull Design Parameters

Parameter	Value
Deadrise angle (deg)	30
Hull beam= external fuselage width (m)	1.7
Hull forebody length (m)	7.0
Hull afterbody length (m)	4.2
Step Height (m)	0.1
Fuselage+Hull max cross section area (m^2)	3.37
Fuselage+Hull Surface Area (m^2)	83.65

FAR 25 states that the reserve buoyance in the main float must be greater than 80% of the takeoff gross weight. The hull was sized to have a buoyancy of 2.5 times the aircraft gross weight without the sponsens, to ensure that the aircraft is unsinkable. The hull volume submerged was conservatively estimated based on the following method:

Assuming length of hull submerged l_W = 9 m, the height to which hull was submerged can be conservatively calculated as:

$$B = b_H h_w l_W \rho_W \quad (19)$$

Where B is the design buoyant force and ρ_W is density of water density, and l_W is the length of hull submerged. This gives the submerged hull height h_W as 1.41m which is less than the hull height suggesting that there is excess buoyancy which, along with sponsens, satisfies the requirement for unsinkable. For 30 degree deadrise the height h_V turns out to be 0.49 m. Initial values of hull parameters are given in Table 7.

H. Drag Estimation

Component buildup method[6] was used to calculate the total parasite drag. The generalized equation can be written as:

$$(C_{D_0})_{subsonic} = \frac{\sum (C_{fi} FF_i Q_i S_{wet_i})}{S_{ref}} + C_{D_{L\&P}} \quad (20)$$

Where *'i'* is the component for which the drag is being computed. FF is the form factor of the components, Q is the interference factor and S_{wet} is the wetted surface area. The turbulent skin friction coefficients for a flat plate are given by:

$$C_f = \frac{0.455}{(\log_{10} \mathrm{Re})^{2.58}(1+0.144M^2)^{0.65}} \quad (21)$$

Table 8. Component Drag Coefficients

Component	Component drag coeff.
Wing	0.0078
Fuselage+hull	0.0071
Step	0.0038
Nacelle	0.0006
Misc. Engine	0.0009
Tail	0.0023
Leakages etc.	0.0011

The second term in the denominator of Eq.21 tends to unity for low subsonic speeds and can be ignored in our case. To take into account increased drag coefficient due to skin roughness, the cutoff Reynolds number is determined using Eq. 22. The lower of actual and the cutoff Reynolds number was used in Eq.19.

$$\mathrm{Re}_{cutoff} = 38.21(l/k)^{1.053} \qquad (22)$$

Where l is the characteristic length and $k=0.634$ x10^{-5}m for smooth paint is used. For the sake of a conservative estimate, a 100% turbulent flow was assumed on the aircraft.

Misc. Engine Drag is obtained using Eq. 23 as:

$$C_{D_0} = 2.5\times10^{-5} P_{net} / S_{ref} \qquad (23)$$

$C_{D\ L\&P}$ is the leakage and protuberance drag which was estimated at 5% of the parasite drag. The fuselage hull drag is based on the surface area obtained using Eq. 17. Hull cleanness ratio is defined as the ratio of hull drag to drag of equivalent body of revolution. For forebody length to beam ratio of 4.12, this value is obtained at 1.5 from Ref 7. Wetted area of the wing (t/c>0.05) is found using the equation:

$$S_{wet} = S_{exposed}(1.977 + 0.52(t/c)) \qquad (24)$$

Where exposed area is the same as wing reference area. Step drag coefficient can be calculated using the equation[10]:

$$C_{D_0} = \frac{0.6\sqrt[3]{h_{step}/l_f}}{S_{ref}} \qquad (25)$$

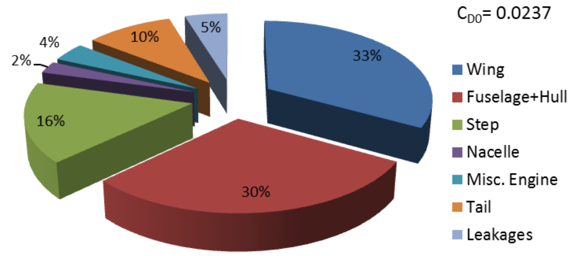

Figure 7. Drag breakdown

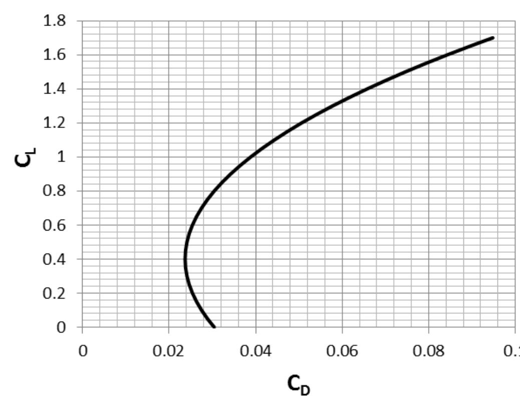

Figure 8. Aircraft Drag Polar

I. Lift Estimation

Lift estimation can be considered to be the final point of initial design. One of the aims here would be to find the lift coefficient to find optimal lift to drag ratio which will then be used to iterate the design from the beginning until it converges.

The max lift coefficient for NACA4415 airfoil is around 1.6. For an unswept wing, Raymer[6] gives a correlation between wing and airfoil max lift coefficient as:

$$C_{L\max} = 0.9C_{l\max} + \Delta C_{L\max} \qquad (26)$$

Where $\Delta C_{L\max} = -0.3$ is obtained graphically from Ref.1. The max lift coefficient for wing thus equals 1.14. The requirement of $C_{L\max}$=1.75 can be fulfilled by a leading edge flap along with a plain flap where a leading edge flap can provide an increase of 0.4 and the remaining 0.21 can be obtained by selecting plain flap area according to:

$$\Delta C_{L\max_{flap}} = 0.9C_{l\max_{flap}}(S_{flap}/S_{ref})\cos\Lambda_{HL} \qquad (27)$$

Where $C_{L\max\ flap}$=0.9 for plain flap, S_{flap}/S_{ref} is the ratio of flap area to wing reference area and $\cos\Lambda_{HL} = 1$ since the flap hinge line is unswept. This gives a flap of around 30% of wing chord if ailerons are used as flaps during landing.

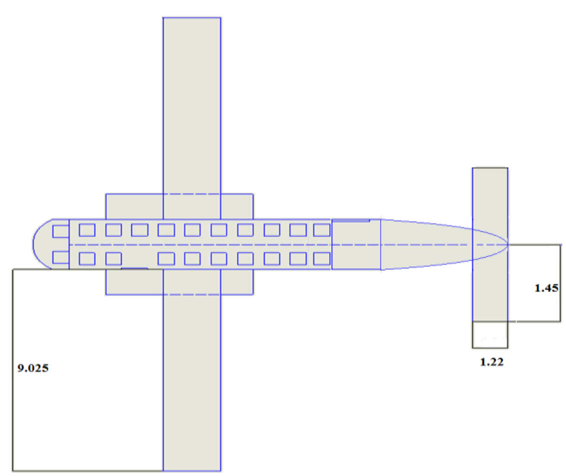

Figure 9. Schematic Top View

The schematic top view of the aircraft in Fig. 9 shows passenger entry door and a cargo door on opposite sides. This is to allow either one to function as an emergency exit in case of the aircraft toppling into the water.

The drag polar equation for a cambered airfoil wing is given by[6]:

$$C_D = C_{D\min} + K(C_L - C_{L\min drag})^2 \tag{28}$$

Where $C_{Dmin} = C_{D0} = 0.0237$ as calculated earlier. $C_{Lmin\ drag}=0.4$ can be safely assumed to be equal to C_l at which angle of attack is zero since NACA 4415 has a large drag bucket.

III. Conclusions and Future Work

In the present study, the key operating requirements for an amphibian aircraft suitable for connecting some islands with each other and /or with some coastal cities in South India were determined from an existing case study[2]. The baseline specifications and general layout of the aircraft were obtained using formulae listed in Raymer[6], recent literature[7] and procedures specific to seaplanes[9]. The results obtained indicate that the aircraft met or surpassed all these requirements.

The current work focuses on obtaining a conceptual design while making various design choices which may not always be the most efficient. This design will now serve as a baseline for the proceeding work regarding multidisciplinary optimization of an amphibian aircraft where every parameter in this design will serve as a variable or an optimization target. For example, wing taper will become a variable to optimize the wing aerodynamically as well as structurally. Multiple hull concepts will be analyzed and considered to obtain the best possible design.

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