Anti-Ship Missiles Carrier Wing In Ground (WIG) Aircraft

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Abstract. A small stealth Wing-In Ground (WIG) aircraft with one crewman carries two anti ship missiles that is designed to fulfil a role of the anti-ship defense system. Because ship defence is based on radar and infrared sensor, it must be capable to avoid both detections. The aircraft has range of 500 km with maximum take-off weight 4500 kg, cruise speed 160 km/hour and payload 750 kg. Its conceptual design follows a procedure from existing design method. It covers wing geometry, power loading, wing loading, sizing, layout, aerodynamic, propulsion, weight, pitching stability, trimmed analysis, and aerodynamic simulation. Besides, by flying low, detection distance between aircraft and ship radars can be reduced. In order to increase anti- ship capability, stealth capability is added and is fulfilled by anti-radar and reduction infared detection. Anti-radar capability is fulfilled by the aircraft configuration designed based on wing-fuselage blended, placement engines and missiles inside the fuselage, and the V-tail. This typical configuration can avoid reflection from radar beam. Capability to reduce infrared detection is fulfilled by concealing the exhaust. Three-dimensional digital based aircraft configuration that is designed by using CAD software is already completed

Keywords: Missile Carrier Aircraft, Stealth, Wing In Ground Effect.

Nomenclature

Bhp = Break horse power
C_{Do} = Coefficient drag at zero lift
C_{bhp} = (TSFC) = Thrust Specfic Fuel consumption
$C_{l\alpha}$ = airfoil lift curve slope
e = Oswald factor
E = Endurance
K_f = empirical pitching moment factor
L_f = fuselage length
L_{VT} = distance of quarter chord of the vertical tail to quater chord of the wing
L_{HT} = distance of quarter chord of the horizontal tail to quater chord of the wing
R = range
V = Velocity W_f = maximum fuselage width
α = angle of attack
η_p = Propeller efficiency

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$\Lambda_{C/4}$ = quarter chird sweep angle
Γ = anhedral angle
σ = density ratio
δ_f = elevator deflection
,$(\frac{\partial\epsilon}{\partial\alpha})$ = downwash angle derivative with respect to angle of attack
Γ = angle of inclination of the V-Tail

1. Introduction

In the context of sovereignty in territorial waters, Indonesia needs to build a defense force capable of deterring against large warships. The ship's defence rely on radar, infrared and audio sensor capabilities in addition to visual. One of the weaknesses of the ship's defenses is against low-flying weapon systems that approach the ship without being visualized and detected by radar, infrared and audio sensors[1,2,3].

To achieve a high level of effectiveness, this deterrence relies on weapon system which is much cheaper than the price of large warships. Cheap means small aircraft and missiles meet these requirements. By comparison, the price of a large ship is $13 billion and an anti-ship missile can cost as little as $100,000 [4].

An aircraft flies at low altitudes has an advantage of reduced drag and increased lift. This happens because off the ground effect. This typical aircraft is a Wing In Ground (WIG) aircraft [5-10]. To avoid detection by ship defense radar, WIG aircraft must fly at the lowest altitude allowed by sea state conditions[11]. In order to increase its effectiveness, the aircraft is made to be stealthy. It is achieved by adding anti-radar and reducing infrared detection. The anti radar capability is achieved by avoiding making the outer surface of the aircraft perpendicular to radar transmission which results in a blended wing-fuselage configuration. Besides, the engines and armament must be placed inside the fuselage and empennage is V-Tail. The reduction of the infrared detection from the exhaust nozzle is achieved by designing the curved duct of the exhaust engine.

The project is a research and development of the design configuration of a single-crew stealth aircraft flying at low altitudes. Its main function is as an antiship missile carrier. In addition, this type of aircraft takes off from the surface of the water which means it does not require a runway on land. With a small, single-crew stealth aircraft, flying low and carrying two anti-ship missiles, this weapon system is highly effective as a formidable defence system.

2. Metodology of Wing In Ground Aircraft Design

2.1 Performance

The aircraft will be operated above water such as high seas waters and will utilize ground effects to reduce drag and increase lift. This type of aircraft takes off from above the surface of the water and is called a Wing In Ground (WIG) aircraft. To meet stealth requirements, aircraft must be anti-radar and reduce infrared detection.

Anti-radar capability is achieved by using a blended wing fuselage configuration, V-Tail, engines and armaments in the fuselage[12,13]. Infrared detection reduction is achieved by making curved exhaust ducts which results in hiding the exhaust engine from the rear view. In the curve duct, the axes of the inlet and outlet nozzles are not in one line. The two axes are horizontally spaced. This configuration can also avoid detection of the fan blades from the front view. The design targets are as follows, range 500 km, cruise speed 160 km/hour, stall speed ≤ 26 m/sec, takeoff distance (from the surface of the water) ≤ 200 m, payload 750 kg.

There are four main flight segments, namely take-off, cruise, loiter and landing as shown in figure -1.

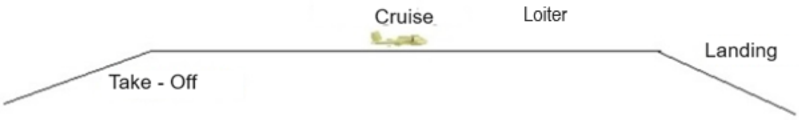

Fig 1. Flying Segment

2.2 Selection of Wing Geometry

For the initial wing layout, the initial Aspect Ratio value can be taken from the results of a statistical analysis of a number of aircraft [14,15] Taper ratio for swept wing is approximated from NACA airfoil wind tunnel test [16]. The wing airfoil is airfoil Y – Clark[17].

2.3 Weight to Power ratio

Power loading is based on historical data.

2.4 Wing Loading

Calculation of other design parameters such as wing loading is done by the concept design method

Stall segment

Wing loading formula

$$\frac{W}{S} = \frac{C_{L,max}\,\rho V_{stall}^2}{2}$$

(1)[15] **Take-Off Segment**

The concept design method uses the Take-Off parameter (TOP) [15].

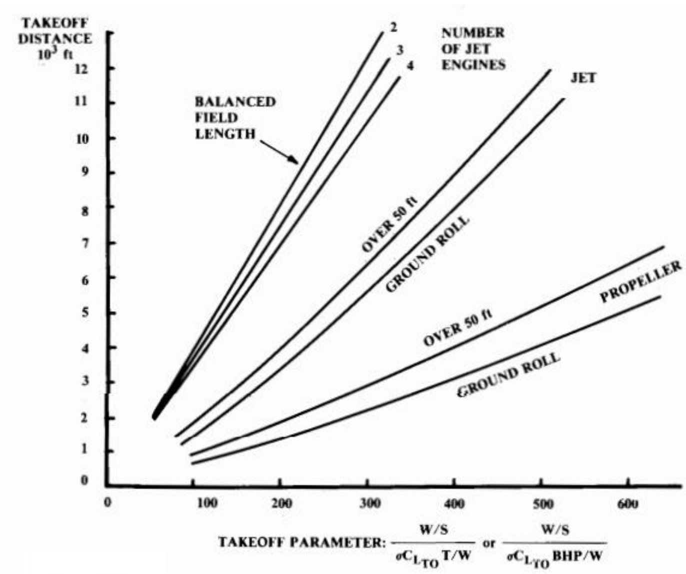

Fig 2. Take-off Parameter

Cruise Flying Segment

The wing loading is calculated by the formula,

$$(W/S)_{Cruise} = \frac{1}{2}\rho V^2\sqrt{\pi A e C_{D0}} \tag{1}$$

Aircraft wing loading is the lowest wing loading of the three flight segments above.

2.5 Initial Sizing

Empty Weight

The empty weight fraction is estimated using a statistical equation that includes the impact of main design variables such as Aspect Ratio (A), take-off weight (W_{TO}), power to weight ratio (hp/W_{TO}), wing loading (W_{TO}/S) and maximum speed (V_{max}). Among types of the aircraft in table A-1 (appendix – 1), the flying boat is the closest resemble to the WIG aircraft.

The equation is applied to flying boat

$$\frac{W_e}{W_0} = 0.41\ (W_0)^{-0.01}\ A^{0.1}\left(\frac{h_p}{W_0}\right)^{0.05}\left(\frac{W}{S}\right)^{-0.12} V_{max}^{0.18} \tag{2}$$

Fraction weight of the flight segments

The drone is flown through seven flight segments. For each flight segment, the weight fraction formula is $W_i/W_{(i-1)}$ where W_i = initial weight of the segment, $W_{(i-1)}$ = final weight of the segment

Engine is turned on, Taxi (segment-0) and Take – Off (segment-1)

When the engine is started, and continues to taxi and take-off, the weight fraction of this flying segment is estimated from historical data.

Climb and acceleration (segment-2)

The weight fraction of the climb flight segment and acceleration is estimated using historical data.

Cruise (segment-3)

To calculate the weight fraction in the cruise segment, the Breguet distance formula is used and for propeller engine,

$$\frac{W_3}{W_2} = \exp\left[\frac{-RC_{bhp}}{550\ \eta_p\ (L/D)}\right] \tag{3}$$

During cruise and loiter L/D can be calculated with the equation

$$\frac{L}{D} = \frac{1}{\frac{qC_{D0}}{W/S} + \frac{W}{S}\frac{1}{q\pi Ae}} \tag{4}$$

The wing loading used in this formula is the wing loading on the cruise or loiter flight segment

Loiter (segment-4)

Fraction weight on the loiter flying segment

$$\frac{W_4}{W_3} = \exp\left[\frac{-EVC_{bhp}}{550\,\eta_p\,(L/D)}\right] \tag{5}$$

Cruise-return (segment-3a)

The same as the cruise segment – 3. Therefore, $\frac{W_{3a}}{W_4}$ = 0.94

Descent (segment-5)

For initial sizing, weight reduction is taken from the data base or historical data, $\frac{W_5}{W_{3a}}$ = 0.995

Landing (segment-6)

Weight reduction after landing and taxing back to the parking area taken from historical data, $\frac{W_6}{W_5}$ = 0.996

Total Mission Weight Fraction

$W_6/W_0 = W_6/W_5$ X W_5/W_{3a} X W_{3a}/W_4 X W_4/W_3 X W_3/W_2 (Cruise) X W_2/W_1 X W_1/W_0 = 0.853

Fuel fraction

With 3 % fuel reserve, total fuel fraction is

$$\frac{W_{fuel}}{W_0} = 1.03\left(1 - \frac{W_6}{W_0}\right) \tag{6}$$

(6)[15]

2.6 Layout Data

From historical data, aspect ratio and taper ratio, the wing area is calculated and the result is, $S_W = 54\ m^2$

Wings

Mean Aerodynamic Chord (MAC) is calculated from formula in [15].

Fuselage

Retrieval of fuselage length data from WIG aircraft in its class provides the length of the fuselage

$$L_{fuselage} = L_{statistic} \tag{7}$$

V-Tails

To meet anti-radar requirements, the aircraft must use the V-Tail.The explanation will be discussed in sub-chapter 2.10.1 (Anti Radar). The V-Tail configuration comes from configuration of horizontal tail and vertical tail.

Calculation of Vertical Tail and Horizontal Tail

The calculation of the layout parameters is calculated based on the concept design method and data historical[15]. Parameters used are the tail volume coefficient, ie.

$$C_{VT} = \frac{L_{VT}S_{VT}}{b_W S_W} \tag{8a}$$

$$C_{HT} = \frac{L_{HT}S_{HT}}{\bar{C}S_W} \tag{8b}$$

where C_{VT} and C_{HT} are the vertical and horizontal tail volume coefficients, respectively The value is taken from statistical data[15]. S_{VT} = vertical tail area, S_{HT} = horizontal tail area,b_W = wing span, S_W = wing area,

V-Tail Calculation

The total area of the V-Tail is equal to the sum of the areas of the S_{VT} and S_{HT}. Aspect ratio (A) ,tip ratio ($\lambda_{V\text{-}Tail}$) , sweep angle is obtained from statistical data.

2.7 Aircraft Configuration

Configuration is drawn by a Computer Aided Design (CAD) software. The 3D digital models are shown in figure – 3. This is an original design from the design team. The configuration aircraft is drawn based upon data in sub chapter 2.6 Layout Data**.**

Fig 3. Aircraft Configuration

2.8 Aerodynamic

Aerodynamic calculation at concept level uses analytical formulas analytical and empirical data[15].

Lift Curve Slope

$$\textit{Lift Curve Slope} = C_{L_\alpha} = \frac{2\pi A}{2+\sqrt{4+\frac{A^2\beta^2}{\eta^2}\left(1-\frac{tan^2\Lambda_{\max t}}{\beta}\right)}}\left(\frac{S_{exposed}}{S_{ref}}\right) \tag{9}$$

$$\beta^2 = 1 - M^2 \quad \text{and} \quad \eta = \frac{C_{l\alpha}}{2\pi/\beta}$$

Parasite Drag

Component buildup Method[15]

$$(C_{D_0})_{subsonic} = \frac{\Sigma(C_{f_c}FF_cQ_cS_{wet_c})}{S_{ref}} + C_{D_{misc}} + C_{D_{L\&P}} \tag{10}$$

The skin friction coeffecient is, as follows,

$$C_f = \frac{0,455}{(Log_{10}\,R)^{2,58}\,(1+0,144M^2)^{0,65}} \tag{11}$$

Fuselage

$$\text{Form factor, } FF_{fuselage} = \left(1 + \frac{60}{f^3} = \frac{f}{400} \right) \qquad (12)$$

Wing

$$\text{Form factor, } FF = \left[1 + \frac{0.6}{(x/c)_m}\left(\frac{t}{c}\right) + 100\left(\frac{t}{c}\right)^4\right][1.34\, M^{0.18}\,(cos\Lambda_m)^{0.28}] \qquad (13)$$

V-Tail

Form factor formula is the same as the one from wing

Cooling Drag

$$(D/q)_{cooling} = (4.9 \text{ X } 10^{-7})\,\frac{bhp.T^2}{oV}\text{, ft}^2 \qquad (14)$$

Miscellaneous drag

$$(D/q)_{misc} = (2 \text{ X } 10^{-4}) \text{ bhp, ft}^2 \qquad (15)$$

Total parasite and engine drag

It is calculated by equation (10)

Induced Drag, $C_{D,i}$

$C_{D,I} = KC_L^2$ Where $K = \frac{1}{\pi Ae}$

Oswald factor for straight-wing aircraft,

$$e = 1.78(1 - 0.045\, A^{0.68}) - 0.64 \qquad (16)$$

(16)[(15)]

Total aircraft drag

For a moderate camber wing, total drag is, as follows,

$$C_{D\text{-}aircraft} = C_{D0} + KC_L^2 \qquad (17)$$

2.9 Propulsion

For cruise condition

$T_{cruise} = D_{cruise}$	(18)

$D_{cruise} = C_{D\text{-}aircraft}\,\frac{1}{2}\,\rho V^2\, S_{ref}$

$$T_{cruise} = C_{D\text{-}aircraft}\,\frac{1}{2}\,\rho V^2\, S_{ref} \qquad (19)$$

This is thrust required that is provided by the engines. Power required for cruise is 25 % of the $P_{max.}$ which requires rate flow mass that can be fulfilled by two inlet engine ducts. The flow velocity at the inlet (V_i) is the same as flight speed. The aircraft flies close surface sea. The formula for the increase in kinetic energy rate for the required power is

$$P_{max} = \frac{1}{2}\,\dot{m}\,(V_e^2 - V_i^2) \qquad (20)$$

With the required power (P), inlet speed (V_i), and mass flow rate mass $\dot{m}$,velocity exhaust velocity (V_e) can be calculated from equation (20). Two turboprop engines will power the aircraft. System propulsion is ducted fan. The engine and the ducted fan are positioned around the middle engine duct located inside the fuselage of the aircraf (figure-4). Calculation of the ducted fan discussed in reference -18.

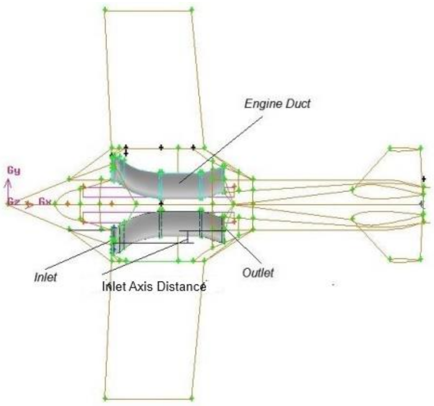

Fig 4. Horizontal distance from the Inlet Axis to the Outlet Axis

2.10 Weight

Calculation of the component weight of the aircraft at conceptual level uses formula analytical and empirical data[15].

$$W_{wing} = 0.036 S_w^{0.758} W_{wf}^{0.0035} \left(\frac{A}{cos^2\Lambda}\right)^{0.6} Q^{0.006} \lambda^{0.04} \left(\frac{100\, t/c}{cos\Lambda}\right)^{-0.3} \left(N_z W_{dg}\right)^{0.49} \tag{21a}$$

$$W_{Horizontal\, Tail} = 0.016 \left(N_z W_{dg}\right)^{0.414} Q^{0.168} S_{ht}^{0.896} \left(\frac{100\, t/c}{cos\Lambda_{ht}}\right)^{-0.12} x \left(\frac{A}{cos^2\Lambda_{ht}}\right)^{0.043} \lambda_h^{-0.02} \tag{21b}$$

$$W_{Vertical\, Tail} = 0.073 \left(1 + 0.2\frac{H_t}{H_v}\right) \left(N_z W_{dg}\right)^{0.376} Q^{0.122} S_{vt}^{0.873} \left(\frac{100\, t/c}{cos\Lambda_{vt}}\right)^{-0.49} x \left(\frac{A}{cos^2\Lambda_{vt}}\right)^{0.375} \lambda_{vt}^{0.039} \tag{21c}$$

$$W_{Fuselage} = 0.052\, S_f^{1.086} \left(N_z W_{dg}\right)^{0.177} L_t^{-0.051} (L/D)^{-0.072} q^{0.241} + W_{press} \tag{21d}$$

$$W_{Flight\, Controls} = 0.053\, L^{1.536} B_w^{0.371} \left(N_z W_{dg} x\, 10^{-4}\right)^{0.80} \tag{21e}$$

$$W_{Avionics} = 2.117\, W_{uav}^{0.933} \tag{21f}$$

$$W_{Furnishings} = 0.0582\, W_{dg} - 65 \tag{21g}$$

$$W_{installed\ engine} = 2.575\, W_{en}^{0.922}\, N_{en} \tag{21h}$$

$$W_{\text{fuel system}} = 2.49\, V_t^{0.726}\left(\frac{1}{1 + V_i/V_t}\right)^{0.363} N_t^{0.242} N_{en}^{0.157} \tag{21i}$$

2.11 Aircraft Stability

A WIG aircraft has stability of the changing *pitching* moment because change of the angle of the attack and flight altitude. An aircraft that flies outside influence ground effect, flying altitude variation has no influence the pitching moment. Change lift coeficient (C_L) is caused by change of angle of attack (α) and flight altitude or height[22]. Then, the parameters that measure stability are $\frac{\partial C_M}{\partial \alpha}$ and $\frac{\partial C_M}{\partial H}$ or variation of the coefficient pitching moment (C_M) with respect to the variation of the angle of attack (α) and height (H). These stability criteria is called Iridov criteria[23] . The formula are, as follows :

$$X_\alpha - X_H > 0 \tag{22}$$

$$X_\alpha = \frac{\partial C_M}{\partial \alpha} \Big/ \frac{\partial C_L}{\partial \alpha} \tag{22a}$$

$$X_H = \frac{\partial C_M}{\partial H} \Big/ \frac{\partial C_L}{\partial H} \tag{22b}$$

Parameter calculations of $\frac{\partial C_M}{\partial \alpha}$ and $\frac{\partial C_M}{\partial H}$ can be performed using analytical and computational methods. It will be in the following sub-chapter 2.12

Analytical Method

MAC (Mean Aerodynamics Chord) is calculated in sub chapter 2.6 Layout Data

Wing

Wing Aerodynamic center $X_{AC,W}$ = 4.9 m (from nose plane), after dividing by MAC then $\bar{X}_{AC,W}$ = 1.4,

$C_{L,\alpha}$ = 2.37/rad, Wing pitching moment (15),

$$C_{mw} = C_{m0,airfoil}\left(\frac{A\cos^2\Lambda}{A + 2\cos\Lambda}\right) \tag{23}$$

$C_{m0,airfoil}$ = 0.097/rad. With swept angle = Λ = 3 deg, $C_{M,wing}$ = 0.066/rad

Fuselage

$$C_m\alpha_{fuselage} = \frac{K_f W_f^2 L_f}{c S_w}\text{, degree (15)} \tag{24}$$

V – Tail

V-Tail aerodynamic center is from the figure. $C_{L_{\alpha},V-Tail}$ is calculate from equation (9)

Downwash

The effect of the downwash is calculated from equation

$$\frac{\partial \alpha_h}{\partial \alpha} = 1 - \frac{\partial \epsilon}{\partial \alpha} \qquad (25)$$

(25)

where the downwash angle derivative with respect to angle of attack,($\frac{\partial \epsilon}{\partial \alpha}$) is calculated from reference – 15. Since it is a V-Tail, the downwash is reduced and it is approximated to be 0.46. Assume dynamic pressure around the tail q_h is 90 % of dynamic pressure around the wing.

Neutral Point

Power – Off neutral point (stick-fixed) is calculated from this equation,

$$\bar{X}_{np} = \frac{C_{L\alpha}\,\bar{X}_{ac,w} - C_{m\alpha,fus} + \eta_h \frac{S_h}{S_w} C_{L\alpha,h} \frac{\partial \alpha_h}{\partial \alpha}\,\bar{X}_{ac,h} + C_{LH}\,\bar{X}_{ac,H}}{C_{L\alpha} + \eta_h \frac{S_h}{S_w} C_{L\alpha h} \frac{\partial \alpha_h}{\partial \alpha} + C_{LH}} \qquad (25a)$$

(26) (15)

$\bar{X}_{AC,H}$ is aerodynamic center of the height ratio. It is taken from the calculation by Computational Fluid Dynamic (CFD) method. By substituting the parameter values from the above, the location of the power-off neutral point, $\bar{X}_{np}$ is calculated.

$$\text{Static margin } \frac{X_{np} - X_{cg,no-fuel}}{MAC} \qquad (25b)$$

Coefficient moment derivative with respect to angle of attack $C_{M\alpha}$

$$C_{M\alpha} = - C_{L\alpha}\,(\,\overline{X_{np}} - \overline{X_{cg}}\,) \qquad (25c)$$

Variation of Height

Variation of lift with respect to height is obtained from CFD simulation of the complete aircraft configuration.

The flow simulation, grid and pressure contours are discussed in the following sub-chapter.

Trim analysis

Moment equation that includes ground effect[15] is

$$C_{m,cg} = C_L\left(\bar{X}_{cg} - \bar{X}_{ac,w}\right) + C_{m,fus} - \eta_h \frac{S_h}{S_w} C_{L,h}\left(\bar{X}_{ac,h} - \bar{X}_{cg}\right) + (C_{L,H})H\left(\bar{X}_{cg} - \bar{X}_{ac,H}\right) \qquad (26)$$

The aircraft is trimmed when moment (C_M) in equation-26 is zero. The aircraft is trimmed by elevator deflection, δ_f which results in moment pitch down from the tail. This moment is generated by lift coefficient of the tail (C_{Lh}).

$$C_{Lh} = C_{L}\alpha_{\text{-V-Tail}}\left[(\alpha + i_w)\left(1 - \frac{\partial \epsilon}{\partial \alpha}\right) + (i_{V-Tail} - i_w) - \alpha_{0-V-Tail}\right] \qquad (27)$$

Change in angle of zero lift, $\Delta\alpha_{0L}$(15)

$$\Delta\alpha_{0L-V-Tail} = -\frac{1}{C_{L\alpha}}\frac{\partial C_L}{\partial \delta_f}\delta_f \qquad (28)$$

where lift increment with flap deflection $\left(\frac{\partial C_L}{\partial \delta_f}\right)$

$$\frac{\partial C_L}{\partial \delta_f} = 0.9\ K_f \left(\frac{\partial C_l}{\partial \delta_f}\right)\frac{S_{flapped}}{S_{ref}}\cos \Lambda_{H.L} \qquad (29)$$

At cruise, the aircraft is trimmed at $C_L = \frac{W}{Sq}$

2.12 Aerodynamic Simulation

The calculation for flow computation of the aircraft configuration was carried out by using CAD/CAE software. It computes Reynold – Averaged Navier-Stokes equations that are capable to simulate aerodynamic flows of incompressible or compressible viscous flow[24,25]. Because the aircraft is designed to fly at low subsonic speeds, namely V = 160 km/h, the air density can be assumed constant. So the flow is incompressible

2.13 Anti – Ship Trajectory

Aircraft is flying at 5 m above sea level and ship defense radar 30 meters above sea level. Aircraft can be seen by the radar at a distance of 27.5 km (figure – 5). In this distance, the reaction time of the ship defense system is reduced. By making WIG aircraft stealthy, time reaction can be further reduced.

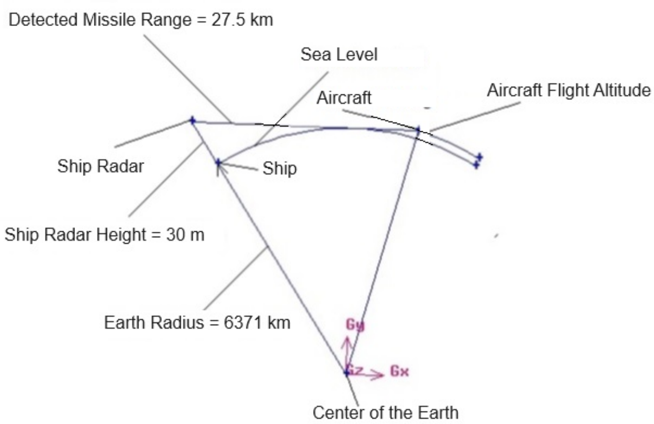

Fig 5. Trajectory WIG Aircraft

2.14 Stealth (AntiRadar)

By locating inlet axis and outlet axis are at a distance horizontally, the fan cross-sections are partially hidden from front and rear directions (see in **fig. 6**). There are no fuselage surfaces that are perpendicular to radar transmission (figure – 21, - 22)

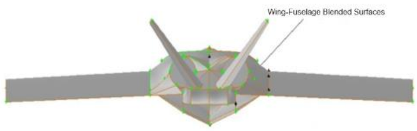

Fig 6. Surface Wing Fuselage Blended (rear view)

2.15 Armament

The aircraft is designed to carry two anti-ship issiles (figure – 7). In order to make make aircraft stealthy, the missiles have to be carried inside the fuselage. The typical missile can be C-705 missile has weight 320 kg, with 4.42 m long and 2.84 m in diameter. Warhead weight is 110 kg. It is high subsonic missile with range of 75 – 140 km[26].

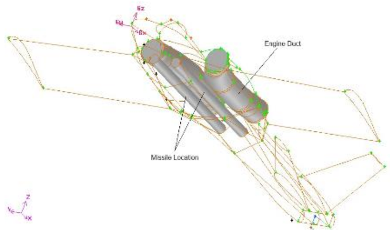

Fig 7. Location of Missiles in the Aircraft Fuselage

3. Calculation

3.1 Geometry Selection

The wing geometry is, as follows, Aspect Ratio = 4.42, taper ratio = 0.9, swept angle = - 3 deg, anhedrals = - 3 deg.

3.2 Power to Weight Ratio

Power to weight ratio was originally chosen to be 6.6 kg/kW (10.87 lb/hp) with the power requirement for take-off is 680 kWatt (910 hp) [14,15].Then, the $W_{TO\ (maximum\ Take\text{-}Off\ weight)}$ = 4500 kg

3.3 Weight to Power ratio

Power loading was originally chosen to be 6.6 kg/kW (10.87 lb/hp) with the power requirement for take-off is 680 kWatt (910 hp) [14,15] .Then, the WTO (maximum Take-Off weight) = 4500 kg

3.4 Wing Loading

Power loading is 6.6 kg/kW.

Stall segment

The lift coefficient at stall $C_{L,stall}$ = 1.8. The stall speed,V_{stall} = 26 m/sec . Air density at sea level, ρ = 1.225 kg/m^3

Therefore, the stall wing loading is 745.3 N/m^2

Take-Off segment

With takeoff distance, $S_{TO} \leq 200$ m, from the graph in figure-2, TOP is 120. $C_{L,TO}$ = 1.5. From the design target hp/W = 0.092. By using the formula $(W/S)_{TO}$ = (TOP) σ $C_{L,TO}$ (hp/W) and σ =1 for sea-level, the wing load at Take-off, $(W/S)_{TO}$ = 794 N/m^2

Cruise segment

The drag (drag) is estimated at $C_{D0} \approx 0.035$[19]. By including the Oswald factor, e = 0.9 and other parameters, then $(W/S)_{Cruise}$ = 800 N/m

The lowest wing load is $(W/S)_{stall}$. Then the wing loading of the plane

$$(W/S) = 745.3\ \text{N/m}^2$$

3.5 Initial Sizing

W_{TO} = 4500 kg, Power Loading (W/P) = 6.6 kg/kW, Wing Loading (W/S) = 745.3 N/m^2

Empty Weight

By substituting the below parameter values to equation (2),

W_{TO} = 9900 lb (4500 kg), A = 4.42, $\left(\frac{h_p}{W_0}\right)$ = 0.092, $\left(\frac{W}{S}\right)$ = 15.56 lb/ft^2, V_{max} = 124,3 mph, the empty weight fraction $\frac{W_e}{W_0}$ = 0.66. Therefore, empty weight, We = 6534 lb or 2970 kg

Engine is turned on, Taxi (segment-0) and Take – Off (segment-1)

The weight fraction of this flying segment is estimated to be W_1/W_0 = 0.985

Climb and acceleration (segment-2)

The weight fraction of the climb flight segment and acceleration to a certain flight height and acceleration to a certain speed is estimated to be W_2/W_1 = 0.995

Cruise (segment-3)

Wing loading cruise $\left(\frac{W}{S}\right)_{cruise}$ = 800 N/m^2

C_{Do}, V, A, e from the above values are substitute to equation (4), then L/D = 9.63. After parameters R = 500 km, Thrust Specfic Fuel consumption (TSFC), C_{bhp} = 0.6 lbm/hp·h[20], V = 160 km/hr, η_p = 0.8 are substituted to equation (3), then $\left(\frac{W_3}{W_2}\right)_{cruise}$ = 0.94

Loiter (segment-4)

With e (endurance) = 1 hour and the same parameter values as the ones in the cruise segment are substituted to equation (5), therefore $\left(\frac{W_4}{W_3}\right)_{loiter}$ = 0.98

Cruise-return (segment-3a)

The same as the cruise segment – 3. Therefore, $\frac{W_{3a}}{W_4} = 0.94$

Descent (segment-5)

The weight reduction is $\frac{W_5}{W_{3a}}$ = 0.995

Landing (segment-6)

The weight reduction is $\frac{W_6}{W_5}$ = 0.996

Total Mission Weight Fraction

W_6/W_0 = 0.853

Fuel fraction

By substitusi, total mission weight fraction to euqation (6), $\frac{W_{fuel}}{W_0}$ = 0.151

Since W_{TO} = 4500 kg, the fuel weight (W_f) is 680 kg. Its volume is 250 gallons. The fuel is enough to cover range of 500 km. From the above, empty weight (W_e) is 2970 kg. The weight of the crew (W_{crew}) is 100 kg.

Payload Weight

$W_{payload}$ = 750 kg. With this weight, the aircraft can carry two antiship missiles. Each missile weight is 320 kg

3.6 Layout Data

The wing area is calculated to be S_W = 54 m^2

Wings

(Mean Aerodynamic Chord), $\bar{C}$ = 3.55 m. C_{root} = 3.67 m, C_{tip} = 3.3 m, S_W = 54 m^2, wing span, b = 15.46 m, wing incident angle, i_W = 7.1 degree,

Fuselage

The length of the fuselage, $L_{fuselage}$ = 16.46 m

V-Tails

The C_{VT} = 0.13, C_{HT} = 0.5, L_{VT} and L_{HT} are shown in Figure – 3. By using these values then S_{VT} = 6.86 m^2, S_{HT} = 5.5 m^2. Mean Aerodynamic Chords ($\bar{C}$) is 2.23 m

V-Tail Calculation

C_{VT} = 0.13, C_{HT} = 0.5. $S_{V\text{-}Tail\text{-}total}$ = 12.36 m^2, the area of one V-Tail, $S_{V\text{-}Tail}$ = 6.18 m^2, The angle of inclination of the V-Tail , Γ = 38.7 deg. This tilt angle can be seen in figure – 8. Radar transmission from the side will not be reflected back to radar transmitter

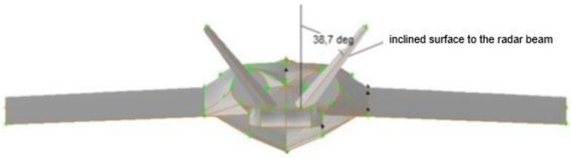

Fig 8. Rear view, Inclined Surface V-Tail

Aspect ratio (A) = 1.4,tip ratio ($\lambda_{V\text{-}Taii}$) = 0.4, sweep angle = 30 deg. Then the span of the V-Tail = 2.94 m, root chord (C_r) = 3 m, tip chord (C_t) = 1.2 m

3.7 Aircraft Configuration

The Y-Clark wing airfoil became the basis for the wings and center section fuselage. The NACA-015 airfoil is the basis for the V-Tail. Sweep angle along the quarter chord (C/4), $\Lambda_{C/4}$is selected to be forward 3 degrees. Small sweep angle suitable for low speed subsonic. The anhedral angle (Γ) is selected 3 degrees for roll stability aircraft. This type of the stability must be fulfilled so that the wing can fly level when it is banked.

Middle and sides of the fuselage are made based on coordinate Clark Y airfoil . There is a step on the middle fuselage from front direction (figures - 9 and - 10). This step made is to reduce hydrodynamics drag from water[17-21]. Step height = 0.1 m. From the step, a straight line is drawn from a middle point of the fuselage until it reaches estimated fuselage length of the aircraft configuration WIG of its class (figure - 8).The fuselage middle section is made wide enough so that it can provide lift force. Besides, the wing fuselage blended configuration for anti-radar requirement, which will discussed in sub- chapter 2.14, provides a flat fuselage configuration. This typical configuration results in wide lower fuselage.

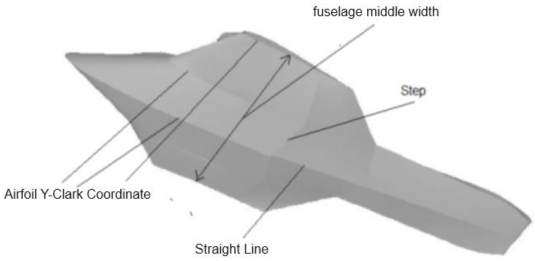

Fig 9. Fuselage bottom view

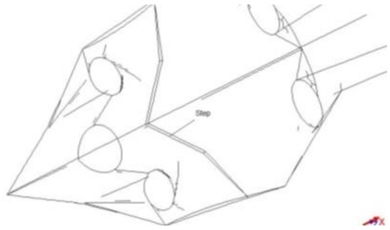

Fig 10. Step (height = 0.1 m) at midship location – lower *fuselage*

Cockpit

Cockpit canopy design of the aircraft is based on figure from reference -15, so that the result can be seen in figure 11.

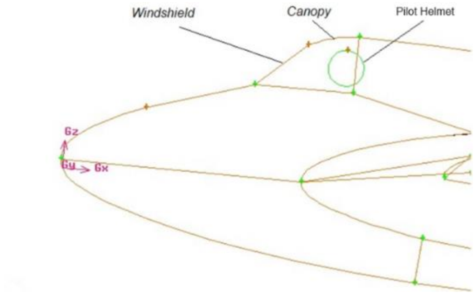

Fig 11. Canopy and Aviator Helmet

3.8 Aerodynamic

M (Mach no.) = 0.13 and β= 0.99

Maximum Lift

Three-dimensional maximum lift coefficient $C_{L_{max}}$ = 0.9$C_{\ell_{max}}$. $C_{\ell_{max}}$= 2 then $C_{L_{max}}$= 1.8

Wing

A = 4.45, η = 0.52, $S_{exposed,wing}$ = 32.4 m^2, $\Lambda_{max,t}$= 3 deg. d = fuselage diamater, F = 1.07 (1 + d/b) ^{2} = 1.2

$C_{L_{\alpha},wing}$= 2.37

V-Tail

$S_{exposed,V\text{-}tail}$= 11.2 m^2, $\Lambda_{max,t,V-Tail}$= 50 deg, $A_{V\text{-}Tail}$ =1.4

$C_{L\alpha,V-Tail}$ = 2.36

Parasite Drag

Reference area, S_{ref} = 54 m^2

Skin friction is calculated at sea level where air viscosity, μ = 1.78 X 10^{-5} kg/m-s . V = 44.44 m/sec.

Fuselage

Length, L_{th} =16.46 m and diamater d = 3 m,Wetted area is obtained from CFD software, $S_{wet\text{-}fuselage}$ = 111.5 m^2, R_e = 50,340,783.14. Interference factor, Q = 1 because the fuselage interference is negligible.

$C_{f-fuselage}$ = 0.0023. f = L /d, thus f = 8.23 and $FF_{fuselage}$ = 1.128

Therefore, $(C_{D0})_{fuselage}$ = 0.0054

Wing

Length is Mean Aerdynamic Chord, MAC = 3.55 m. R_e = 10,857,216.3, C_{f-wing} = 0.003
Wetted area is obtained from CFD software, $S_{wet\text{-}wing}$ = 82.9 m^2. Interference factor, Q = 1 because the wing interference negligible. t/c = airfoil thickness ratio =0.11, $(x/c)_m$ = 0.28 and Λ_m = - 3 deg. Then FF_{wing} = 1.156
Therefore, $(C_{D0})_{wing}$ = 0.006

V-Tail

Length is Mean Aerdynamic Chord, MAC = 2.23 m.The Reynolds number, R_e = 6,820,166.85.
$C_{f-V-Tail}$ = 0.0032. Wetted area is obtained from CFD software, $S_{wet\text{-}V\text{-}tail}$ = 23.23 m^2.Interference factor,
Q = 1.03.Form factor formula is the same as the one from wing, t/c = airfoil thickness ratio =0.15, $(x/c)_m$ = 0.3 and Λ_m = 30 deg. $FF_{V\text{-}tail}$ = 0.976. Therefore, $(C_{D0})_{V\text{-}Tail}$ = 0.006

Cooling Drag

bhp = (22 %) bhp_{Max}[15]. bhp = 205 hp, Temperature, T = 519 R, V = 144.36 ft/sec, σ = 1 (sea level)

$C_{Dcooling}$ = 0.003

Miscellaneous drag

C_{Dmisc} = 0.0033

Total parasite and engine drag

C_{D0} = 0.0237

Induced Drag, $C_{D,i}$

With aspect ratio, A = 4.42, therefore e = 0.9 and K = 0.078.C_L is from cruise condition at sea level.
Then C_L = 0.67.Therefore, $C_{D,I}$ = 0.0356

Total aircraft drag

For a moderate camber wing, total drag is $C_{D\text{-}aircraft}$ = 0.06. At the iteration, this drag value is adequate.

3.9 Propulsion

The total drag, $C_{D\text{-}aircraft}$ = 0.06 and V = 44.44 m/sec at sea level. With the values from the above,

T_{cruise} = 3900 N. Each duct has a diameter of 1 m. The flow velocity at the inlet (V_i) is V = 44.44 m/ sec. The air density, ρ = 1.225 kg/m^3. The mass flow rate, $\dot{m}$= 42.75 kg/sec. The velocity exhaust velocity (V_e) can be calculated by using equation (20) and the result is $V_{e,max}$ = 135 m/ sec. Based the aircraft geometry, speed and drag, the engine is two Rolls-Royce RR500. Its maximum power is 354 kW[20] .

3.10 Weight

The component weight of the aircraft are summarized in table -1

Table - 1 Component Aircraft Weight

Component	Weight	Distance to datum	Moment
Fuselage	803.6 kg	6.55 m	5263.6 kg.m
Wing	463.4 kg	5.57 m	2581.14 kg.m
V-tail	90 kg	15.15 m	1363.5 kg.m
Flight Control	102.17 kg	6.55 m	669.2 kg.m
Avionics	491.9 kg	3.36 m	1652.8 kg.m
Furnishings	203.2 kg	3.4 m	690.88 kg.m
Fuel System	70 kg	5.27 m	370 kg.m
Engine	530 kg	5.8 m	3074 kg.m
Electrical	215.7 kg	6.55 m	1412.8 kg.m
Empty Weight	2970 kg		17077.9 kg.m
Crew	100 kg	3.02 m	302 kg.m
Payload	750 kg	5.8 m	4350 kg.m
No-Fuel Weight	3820		21729.9 kg.m
Fuel	680 kg	5 m	4080 kg.m
Total Weight	4500 kg		25129.9 kg.m

c.g. empty weight = 5.75 m, c.g. no fuel = 5.68 m,c.g. total = 5.58 m (datum at nose tip)

3.11 Aircraft Stability

Analytical Method

The stability of the pitching moment locates center of gravity located (most aft) 45% MAC distance X_{CG}= 5.688 m from the nose of the aircraft. After dividing by MAC then $\bar{X}_{CG}$= 1.6. Other parameters, are, as follows, (figure-12)

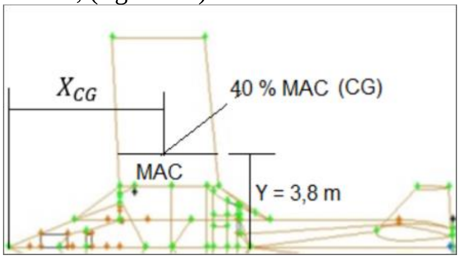

Fig 12. MAC and Location of Center of Gravity

Wing

Wing Aerodynamic center $X_{AC,W}$ = 4.9 m (from nose plane), after dividing by MAC then $\bar{X}_{AC,W}$ = 1.4,

$C_{L,\alpha}$ = 2.37/rad, $C_{m0,airfoil}$ = 0.097/rad. With swept angle = Λ = 3 deg,
$C_{M,wing}$ = 0.066/rad

Fuselage

By substituting W_f = 3.7 m, $L_{,f}$, and K_f to equation (22), $C_{M,\alpha,fuse}$ = 0.157751/radian

V – Tail

V-Tail aerodynamic center, $X_{AC,V-T}$= 14.75 m (from the nose of the aircraft),then $\bar{X}_{AC,V-T}$ = 4.16. From the above, $C_{L_\alpha,V-Tail}$ = 2.36/radian

Downwash

$(\frac{\partial\epsilon}{\partial\alpha})$ = 0.49. Then the downwash $\frac{\partial\alpha_h}{\partial\alpha}$ = 0.51.

Neutral Point

The result is $\bar{X}_{np}$ = 1.65. With MAC = 3.55 m X_{np} = 5.85, center of gravity without fuel, $X_{cg,no-fuel}$ = 5.688, then

$$\text{Static margin } \frac{5.85\ m-5.68\ m}{3.55m} = 0.047\ (4,7\ \%)$$

Coefficient moment derivative with respect to angle of attack $C_{M\alpha}$.With the above values $C_{M\alpha}$ = - 0.1. With this static margin, the aircraft has sufficient pitching stability.

Variation of Height

The results of the lift variation with respect to height are in the following tables

Table – 2 Lift Variation

Height	Lift	C_L
0.9	64377,4	0,985
1.5	61613,3	0,927

Height is from ground level or sea level.

Trim analysis

$\Lambda_{H.L}$ is hinge line angle of the flap which is approximately to be 30 degrees. K_f is a correction factor ≈ 1. $C_{L_\alpha,V-Tail}$ = 2.36. $\left(\frac{\partial C_l}{\partial\delta_f}\right)$ = 4.8, $\frac{S_{flapped}}{S_{ref}}$ ≈ 0.4. Therefore $\left(\frac{\partial C_L}{\partial\delta_f}\right)$ = 1.5 and $\Delta\alpha_{0L-V-Tail}$ = -0.635 δ_f.

Since it is V-Tail with tilt angle 38.7 degree, the effective lift is sin 38.7 of the total tail lift.
The total lift of the aircraft is

$$C_{Ltotal} = 2.565\ \alpha + 0.27\ \delta_f + 0.343H - 0.0284$$

(30)

By substituting all the parameter values, equation-26 becomes

$$C_{mcg} = -0.1317\ \alpha - 0.69\ \delta_f - 0.0472\ H + 0.073$$

(31)

Tables of C_M versus C_L at various α and δ_f at height above sea level, H = 1.5 m, are

Table – 2 C_M at various α and δ_f(degree)

	δ_f			
α	0	2	4	6
6	0,032	0,0072	-0,0162	-0,04
10	0,0227	-0,0138	-0,0255	-0,0495

Table – 3 C_L at various α and δ_f(degree)

	δ_f			
α	0	2	4	6
6	0,4385	0,448	0,4574	0,4668
10	0,6176	0,627	0,6364	0,646

The graphic is in figure 13

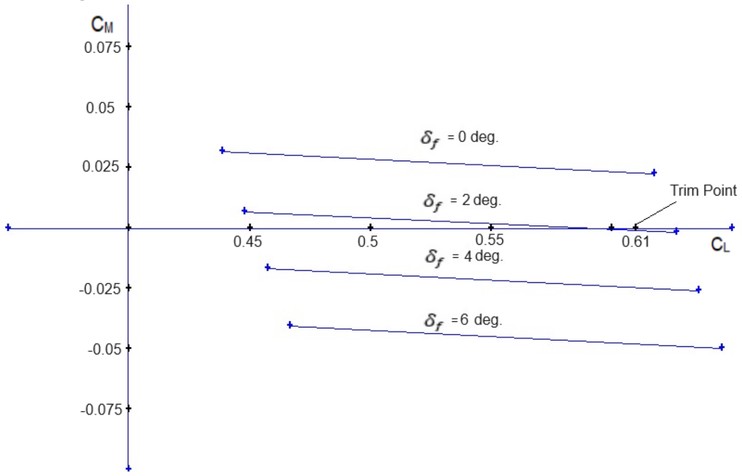

Fig 13. Trim Plot, elevator deflection δ_f in degree

At velocity, V = 160 km/hour, cruise 1,5 m above sea level and wing loading (W/S) = 745.3 N/m^2, q is dynamic pressure. At trim, C_L = 0.61, $\delta_f \approx$ 6.5 degree. With these C_L and δ_f, the aircraft has sufficient lift and torable trim drag

3.12 CFD Simulation Result

The grid consists of 611,497 triangle cells. In order to find out the influence ground effect, simulations on three flying altitudes which are h =10000 m, 1.5 m and 0.9 m are carried out. Pressure on the upper surface can be seen to contribute to the lift. Its lowest value is around maximum thickness of the wing section (figure – 14). At flying altitude of 10,000 m, there is no influence of the ground effect. From figure – 15 b to figure – 15 c, it is clearly seen that pressure on the lower surface increases with reduction in flight altitude from 10000 m to 0.9m. Integration of the pressure results in lift that shows increment as the flight altitude is reduced. The result is in table – 4.

Table - 4 Lift

H(m)	Lift(N)
10000	47782
1.5	54651
0.9	57221

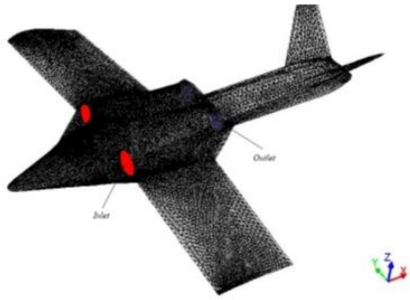

Fig 14. Aircraft Grid (611,497 triangle cells)

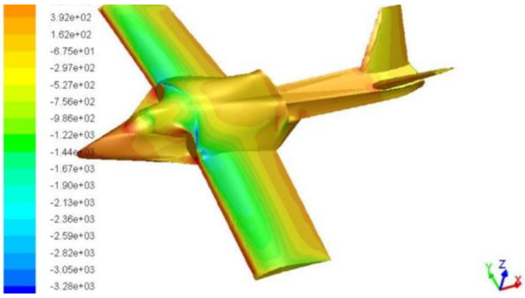

Fig 15. a Contours Pressure of the aircraft upper surface in Pa, flight altitude , h = 0.9 m above surface sea

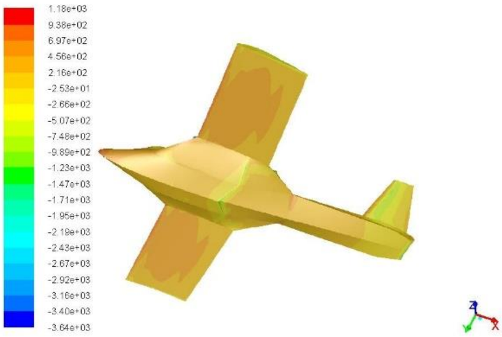

Fig 15. b Contours Pressure of the lower surface in Pa, flight attitude h = 1000 m above surface sea

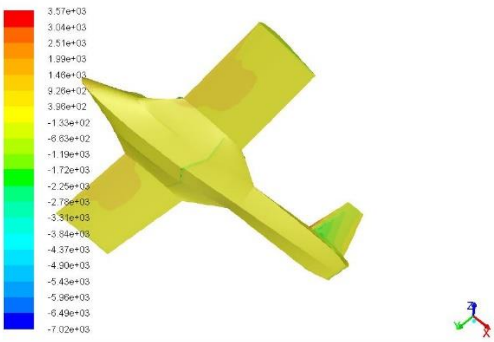

Fig 15 c Contours Pressure of the lower surface in Pa, flight attitude h = 0,9 m above surface sea

4. Conclusion

The aircraft mission profile is to deliver antiship missiles with sea skimming capability at cruise speed 160 km/hour. Y-clark airfoil is selected for wing which is suitable for WIG aircraft. Constraint analysis provides weight power ratio and wing loading which are 6.6 kg/kg and 745.3 N/m2, respectively. The sizing results in maximum take-off weight 4500 kg, fuel weight, paylod weight. The fuel weight of 680 kg fulfilled range profile of 500 km. With payload weight of 750, the aircarft can carry two antiship missiles.

With complete geometry paramaters, 3-dimensional digital model of the configuration WIG aircraft is drawn by using CAD software. This aircraft configuration is the original work of our team. In order to guarantee the the success of the aircraft mission, stealth capability is added which fulfilled by wing-fuselage blended configuration, engines and missiles inside the fuselage, and the V-tail.

The component weight estimation provides center gravity. Stattic margin is 4.7 % which fulfilled the pitching stability. It is calculated after the calculation neutral point. The CFD simulation provides pressure contour which shows the increase in lift as the flight height decreases. The aircraft is trimmed at CL = 0.61 with elevator deflection $\delta f \approx 6.5$ degree. With its relatively small size, this WIG missile-carrier aircraft is a formidable system weaponry for anti-ship defence.

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