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# Design and Optimization of Aircraft Configuration for Minimum Drag

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### Cover Page Footnote

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## Introduction

A complex aircraft design process consisting of numerous disciplines has been developed over many years. These disciplines are integrated and blended together to generate an optimum configuration that satisfies the given requirements (Nguyen, 2011).

There are three phases of aircraft design; conceptual, preliminary, and detail phases. Among them, the conceptual design phase is characterized by the initial definitions that come from requirements established by market needs. Thus, this phase is the most interactive in the whole aircraft design process. The aircraft geometry will change several times driven by optimizations done in order to achieve mission requirements (de Paula, 2017).

Raymer established an aircraft conceptual design process characterized by a large number of design alternatives and trade-off studies, as well as a continuous change in the aircraft concepts under consideration (1999). Howe proposed a systematic and logical approach for several types of aircraft such as two-seat, aerobatic, short- and medium-haul airliners or short take-off landing (STOL) aircraft (2000). Corke proposed an optimization approach to conceptual design of a supersonic business jet (SSBJ; 2003).

Above traditional optimization methods rely on empirical data, which are readily available for common configuration aircraft. Multidisciplinary design optimization is a field of research that studies the application of numerical optimization techniques to the design of engineering systems involving multiple disciplines or components. Since the inception of multidisciplinary design optimization, various methods (architectures) have been developed and applied to solve multidisciplinary design-optimization problems (Martins & Lambe, 2013). In addition, Boone and Striz optimized aircraft configuration for minimum Drag and maximum Range (2010). Nhu Van Nguyen, Daniel Neufeld, Sang Ho Kim, and Jae-Woo Lee optimized Multidisciplinary Configuration Design for Advanced Very Light Aircraft (VLA) by using SQP algorithm (Nguyen, 2011). Ashraf and Abbas presented conceptual design and used genetic algorithm to optimize maximum Range of Supersonic Business Jet (SSBJ) (2014). Jaeger proposed Aircraft Multidisciplinary Design Optimization under both model and design variables uncertainty (2013). Furthermore, many optimization techniques are also implemented during the aircraft conceptual design (Hoburg & Abbeel, 2014; Perez, 2006; Zuo & Chen, 2015).

Drag is wasted energy, the generation of cross flow kinetic energy is an inherent byproduct of the generation of lift over a finite wing span (Takahashi & Donovan, 2011). Optimization for drag results in maximum L/D, which can improve the performance (Boone & Striz, 2010). To get the optimum lift distribution and optimum wing span, Hunsaker, with assistance from Phillips, minimize the induced drag (2017). McGeer designed wing for minimum drag with practical constraints (1984). Rakshith proposed optimal low-drag wing planforms for tractor-configuration propeller-driven aircraft (2015). Pate and German presented wing optimization for minimum induced drag with generalized bending moment constraints (2013).

In this study, conceptual design code of Single Seat Aerobatic Airplane (SSA) was developed and validated. The previous researches study on the aerodynamics of aircraft that are mainly focus on the Drag of aircrafts. The results shown that the reduction of drag can directly improve the performances such as Range. The optimization of SSA focus on higher performance, better stability, therefore, the validated SSA was used as baseline for optimum configuration for minimum total drag to improve the performance.

## Conceptual Design Code Development Steps

The role of the conceptual aircraft design is to propose aircraft configurations that can best meet a set of needs, then to identify several design alternatives. The conceptual design of a Single Seat Aerobatic aircraft (SSA) that present in (Raymer, 1999), will be considered, which allows us to define the main features of the aircraft.

### Requirements

As mention above, the requirements which is the main part of the conceptual design. The SSA need to design cruise Range ≥ 280nm at 115kts, and maximum velocity of 130kts, and a stall velocity of 50kts.

Take-off distance ≤ 1000ft
Rate of climb ≥ 1500 ft/min
Crew weight = 220 lb

The engine (LYCOMING O-320-A2B) having $C_{bhp}$, specific fuel consumption is assumed 0.5 at cruise speed, revolution of 2700RPM, and horse power of 150Hp. The SSA has to fly with above requirements, and the related mission profile is given in Figure 1.

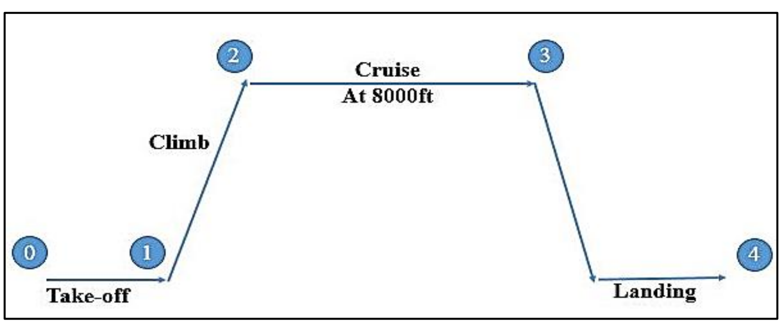



*Figure 1.* Mission Profile

## Geometry Section

The selection of wing thickness ratio, aspect ratio, taper ratio, and sweep angle plays a vital role in determining the performance of a conceptual aircraft design (Raymer, 1999). Low wing configuration and aspect ratio of 6, taper ratio of 0.4, quarter chord sweep of 0, NACA $63_2015$ as tip and $63_2012$ as root were used to prevent tip stall. Horizontal and vertical tails were used with aspect ratios of 4, NACA 0012 and taper ratios of 0.4.

## Power to Weight Ratio and Wing Loading

Hp/W of 1/8 was chosen by engine type and W/S was calculated for each mission segment: take-off, stall, cruise, and landing conditions. The lowest value was selected to ensure that the wing is large enough for all flight conditions.

## Initial Sizing

Gross take-off had been calculated (1) using an iterative process and by using the fuel fraction for each mission segment along with an estimated weight of the same type of aerobatic aircraft from historical data. Gross take-off weight is the sum of Payload weight, Crew weight, Fuel weight, Empty weight of the aircraft. The SSA was designed one crew member.

$$W_0 = (W_{crew}+W_{pay}) / [1- (W_f/W_0)-(W_e/W_0) ] \quad (1)$$

$$(W_f/W_0) = 1.06\,(1 - W_4/W_0) \quad (2)$$

$$(W_4/W_0) = W_1/W_0 * W_2/W_1 * W_3/W_2 * W_4/W_3 \quad (3)$$

$$(W_e/W_0) = 1.495\, W_0^{-0.1} \quad (4)$$

Take-off weight fraction, climb weight fraction and landing weight fraction are selected 0.97, 0.985 and 0.995 respectively (Raymer, 1999).

## Layout Design

Next, the actual sizes of wing, fuselage, tails, fuel tank, tire size, and propeller diameter were defined based upon the estimated gross take-off weight. Fuselage wetted area was defined by Sears-Haack (5), a symmetric revolution that also has relatively low wave Drag compared to other shapes (Howe, 2000). The wave drag is only concern with supersonic and transonic flight but in this paper, Sears-Haack's fuselage wetted area equation was used because it can easily and quickly define the fuselage wetted area.

$$S_{wetf} = 0.8083\, \pi\, l\, r(0) \quad (5)$$

r(0) is the maximum radius of fuselage and 'l' which is the quarter of the fuselage length. Fuselage fineness ratio of 6.38 was chosen.

## Aerodynamics

### Lift curve slope.

The lift curve is needed during the conceptual design for the following reasons. First, it is used to properly set the wing incidence angle. Secondly, it is important for longitudinal stability analysis (Raymer, 1999). The lift curve slope of SSA was done by (6).

$$C_{L\alpha} = \frac{2\,\pi\,AR}{2+\sqrt{4+\frac{AR^2\,\beta^2}{\eta^2}\left(1+\frac{\tan\ \Lambda maxt)}{\beta^2}\right)}}\left(\frac{Sexp}{S}\right) F \quad (6)$$

$$\beta^2 = 1 - M^2 \quad (7)$$

Airfoil efficiency η is approximately as about in the 0.95 if the airfoil lift curve slope is as a function of Mach number (Raymer, 1999). Sexp is the exposed wing planform, i.e., the wing reference area less the part of the wing covered by the fuselage. F is the fuselage lift factor which is done by (8)

$$F = 1.07\ (\ 1+ d/b\ )^2 \quad (8)$$

### Total drag.

The total drag highly effects on the performance parameters and minimization of total Drag directly improves the Range (Nguyen, 2011). Total drag is the sum of parasite Drag ($C_{D0}$) and lift induced Drag ($K\ C_L^2$).

$$D = 1/2\ \rho_{cr}\ V_{cr}^2\ S\ C_D \quad (9)$$

$$C_D = C_{D0} + K\ C_L^2 \quad (10)$$

$$C_{D0} = \Sigma\ [C_f\ F\ Q\ (S_{wet}/S_{ref})] + C_{Dmisc} + C_{DL\&P} \quad (11)$$

Parasite Drag ($C_{D0}$) is the total sum of wing, tail, fuselage Drag ($\Sigma\ [C_f\ F\ Q\ (S_{wet}/S_{ref})]$), leakage and protuberance Drag ($C_{DL\&P}$), engine cooling Drag, landing gear Drag and miscellaneous Drag ($C_{Dmisc}$).

$$C_f = f\,(M, R_e) \quad (12)$$

$C_f$, skin friction coefficient. Assuming SSA has fully turbulent flow at sea level, Mach number of 0.15 and viscosity of $0.37\times10^{-6}$ are used. To get the actual Oswald span efficiency, equation (13) was used for straight wing aircraft that described in (Raymer, 1999).

$$K = 1\ /\ (\pi\ e\ AR) \quad (13)$$

$$e = 1.78\,[\,1- 0.045(AR)^{0.68}] - 0.64 \tag{14}$$

$$C_L = (W/S) / (1/2\ \rho_{cr}\, V_{cr}^2\ ) \tag{15}$$

## Stability and Control

### Longitudinal static stability.

The longitudinal stability is the measure of response of the aircraft due to a changing pitch angel condition (Howe, 2000). The coefficient of longitudinal stability is done by (16)

$$C_{m\alpha} = -\,(SM)\ C_{La} \tag{16}$$

$$\overline{xnp} = \frac{(C_{La}\ \overline{x_{acw}}) - C_{m\alpha f} + \eta_{ht}\frac{S_{ht}}{S_w}\ C_{Lah}\frac{\partial\, a_h}{\partial\, a}\ \overline{x_{ach}}}{C_{La} + \eta_{ht}\frac{S_{ht}}{S_w}\ C_{Lah}\frac{\partial\, a_h}{\partial\, a}} \tag{17}$$

$$\overline{x_{acw}} = {x_{acw}}/{Cmac} = {x_{c/4w}}/{Cmac} \tag{18}$$

$$\overline{x_{ach}} = {x_{ach}}/{Cmac} = {x_{c/4h}}/{Cmac} \tag{19}$$

$$\frac{\partial\, a_h}{\partial\, a} = f\,(\,I_t,\,Z_t\,) = 0.62 \tag{20}$$

The horizontal distance between $x_{acw}$ and $x_{ach}$ is $I_t$ and the vertical distance between the horizontal reference line and horizontal tail position is $Z_t$. The downwash, $\frac{\partial\, a_h}{\partial\, a}$ of 0.62 is set from the downwash estimation that described in (Raymer, 1999).

$$xnp = \frac{\overline{xnp}}{Cmac} \tag{21}$$

$$SM = (xnp - xcg) / Cmac \tag{22}$$

xcg is done by calculating each components' weights and using Statical Group Weights Method (Raymer, 1999). An appropriate range of $C_{m\alpha}$ is between -1.5 and -0.16 (Howe, 2000).

**Lateral-directional static stability.**

Directional stability is the stability about vertical axis. The most important factor in directional stability is the vertical stabilizer.

$$C_{n\beta} = (C_{n\beta})_F + (C_{n\beta})_W + (C_{n\beta})_{VS} \tag{23}$$

A directional stability reasonable range is between 0.05 and 0.1(Nguyen, 2011).

## Performance

A performance parameter is a quantitative indicator representing how a vehicle operates in a specific flight condition. Typical performance parameters are weights, speeds, aerodynamic loads, engine thrust and power, range and endurance, accelerations, emission indexes (noise, exhaust gases) and many more. At least 60 different parameters can be taken into account in a full aircraft performance analysis (Filippone, 2006). The performance main parameters of Range, Take-off distance, climb was selected because the baseline aircraft SSA was only considered that three parameters. For the validation of design code, the authors also selected these parameters to know the accuracy of design code.

**Range.**

The Range is maximized by Breguet Range equation for propeller-power aircraft (24) at a cruise speed of 115kts. The maximum range is that the aircraft is flying at Lift to Drag Ratio Maximum while the specific fuel consumption and engine efficiency are constant.

$$\text{Range} = 550\ (\eta_p / C_{bhp})\ (L/D)\ \ln (W_3/W_2) \tag{24}$$

The maximum Range for $L/D_{max}$ in (26)

$$C_{L\max\text{ Range}} = (C_{D0}/K)^{1/2} \tag{25}$$

$$C_{D\max\text{ Range}} = 2C_{D0} \tag{26}$$

$$L/D_{max} = (C_L/C_D)_{\max\text{ Range}} \tag{27}$$

**Take-off distance.**

Take-off distance was done by calculating the take-off parameter (TOP).

$$TOP = \frac{W/S}{\sigma\ CL_{TO}\ T/W} \tag{28}$$

$\sigma$, density ratio is assumed 1. The aircraft take-off at 1.1 times the stall speed so the $CL_{TO}$, take-off lift coefficient is the maximum lift coefficient divided by 1.21(Raymer, 1999).

**Climb.**

Rate of climb, or vertical velocity, is the velocity times the sine of the climb angle (r) (Raymer, 1999).

$$V_v = V \sin(r) \tag{29}$$

$$\sin(r) = (T/W - D/W) \tag{30}$$

T/W was done by the using Hp/W and D/W which is the ratio of total drag and maximum take-off weight.

## Design Code Validation

The design code generates all of the aircraft configurations, weights, Lift, Drag, Static Stabilities, and Performance parameters that are satisfied for design requirements by the conceptual design steps that was shown in each above section. And the design code was validated with existing SSA in (Raymer, 1999) as shown in Table 1 for wing, tails configurations and some specifications. Generally, the results from the code agree well with existing data. Maximum error of 1.6% at wing span and horizontal tail area was found.

Table 1
*Design Code Validation*

| | SSA | Code | Units |
|---|---|---|---|
| $W_0$ | 1200 | 1200 | lb |
| $W_e$ | 883 | 883 | lb |
| W/S | 10.2 | 10.3115 | lb/ft$^2$ |
| S | 116 | 116.3747 | ft$^2$ |
| bw | 26 | 26.4244 | ft |
| AR | 6 | 6 | ft$^2$ |
| $\lambda$ | 0.4 | 0.4 | |
| Sht | 25.3547 | 24.9487 | ft$^2$ |
| bht | 10.0707 | 9.9897 | ft |
| Svt | 11.46481 | 11.2845 | ft$^2$ |
| bvt | 4.1476 | 4.1142 | ft |
| $C_{L_\alpha}$ | 4.8547 | 4.8557 | per radian |
| $C_D$ | 0.0337 | 0.0336 | |
| $C_{m\alpha}$ | -0.58 | -0.5809 | Stick fixed |
| $C_{n\beta}$ | -0.0717 | -0.0717 | |
| Range | 207.204 | 207.204 | nm |
| Max Range | 289.9744 | 289.9744 | nm |
| TOP | 120 | 120 | |
| ROC | 1500 | 1500 | |

## Optimization

Sequential Quadratic Programming (SQP) has become the method for solving nonlinearly constrained optimization problems. In the form of non-linear program (NLP), includes as special cases linear and quadratic programs in which the constraint functions **h** and **g** are affine and **f** is linear or quadratic. While these problems are important and numerous the great strength of the SQP method is its ability to solve problems with nonlinear constraints. For this reason, it is assumed that NLP contains at least one nonlinear constraint function. The SQP is to model at a given approximation solution, say $x^k$, by a quadratic programming subproblem, and then to use the solution to this subproblem to construct a better approximation $x^{k+1}$. This process is iterated to create a sequence of approximations that, it is hoped, will converge to a solution x* (Boggs & Tolle, 1995).

The aircraft configuration is optimized by using SQP algorithm for minimum drag within the desired constraints and design variables. While considering an optimization problem, the number of variables, constraints, challenging objectives and time tend to increase the complexity of design space searching. An optimization tool must be flexible enough to include a high number of design variables to reach better design results. At this point, selecting strong variables, meaningful limits and assigning efficient penalties are crucial to obtain better results on behalf of the consumed time and design effort (Cavus, 2016).

The above SSA airplane was chosen as a baseline model for optimization. The objective function, variables and design constraints are considered as follows. The optimization equations can be mathematically written as below:

$$\text{Minimize:}\quad \textbf{Total Drag Coefficient} \tag{31}$$

$$\left.\begin{array}{lll} \text{Minimize:} & \boldsymbol{f} = C_D\,(x_i) & i = 1 \text{ to } 10 \\ \text{Subject to:} & h\,(x) = 0 & \\ & g_j\,(x) \leq 0 & j = 1 \text{ to } 6 \end{array}\right\} \text{NLP} \qquad (32)\text{–}(34)$$

The ten design variables are listed in Table 2. Wing loading, wing and tails configurations are considered as variables. The objective function is to minimize the total Drag coefficient. The airfoil of wing, horizontal and vertical tail, and thickness ratio are selected, hence the gear Drag, cockpit Drag, engine cooling Drag, miscellaneous Drag are fixed during the optimization.

Table 2
*Design Variables*

| | Baseline | Bounds | | Units |
|---|---|---|---|---|
| $C_D$ | **0.03346** | **lower** | **upper** | |
| W/S | 10.3115 | 8.5 | 23 | lb/ $ft^2$ |
| bw | 26.6 | 23 | 30.8 | ft |
| Cr | 6.25 | 2.4 | 6.25 | ft |
| Ct | 2.533 | 2.4 | 3.18897 | ft |
| bvt | 4.1 | 3.2 | 4.895 | ft |
| Crvt | 4 | 1.6 | 4.9212 | ft |
| Ctvt | 1.6 | 1.1017 | 2.4475 | ft |
| bht | 10.1 | 8.79 | 10.1 | ft |
| Crht | 3.6 | 1.32221 | 3.6 | ft |
| Ctht | 1.4 | 0.88156 | 1.5748 | ft |

Baseline performance is used as constraints for better performance and static stability is also considered in Table 3.

Table 3
*Constraints*

| | | |
|---|---|---|
| Range | ≥ | 280nm |
| Stall Speed | ≤ | 50 kts |
| Take-off Distance | ≤ | 1000ft |
| Rate of Climb | ≥ | 1500 ft/min |
| $C_{m\alpha}$ | ≥ | -1.5 |
| $C_{m\alpha}$ | ≤ | -0.16 |
| $C_{n\beta}$ | ≥ | 0.05 |
| $C_{n\beta}$ | ≤ | 0.1 |

## Optimizer

The overall architecture of the code can be seen in Figure 2 (waterfall diagram). Since the objective function and the constraints require some of the same subroutines, they are nested in another subroutine called 'Physics' which is called by both. The 'Atmosphere' subroutine uses the geometric altitude to determine the static pressure, temperature and density at the operational altitude. The 'Geometries' subroutine calculates the areas of the wing, tails, fuselage and tail location. The 'Weights' function takes the areas and design variables as input. It then uses a Statical Group Weights Method (Raymer, 1999) to calculate each components' weights and to estimate cg location of the aircraft. The 'lifts' function takes its input and calculates first $C_{L\alpha}$ from the wing airfoil lift curve slope and aspect ratio for estimation of stability constraint.

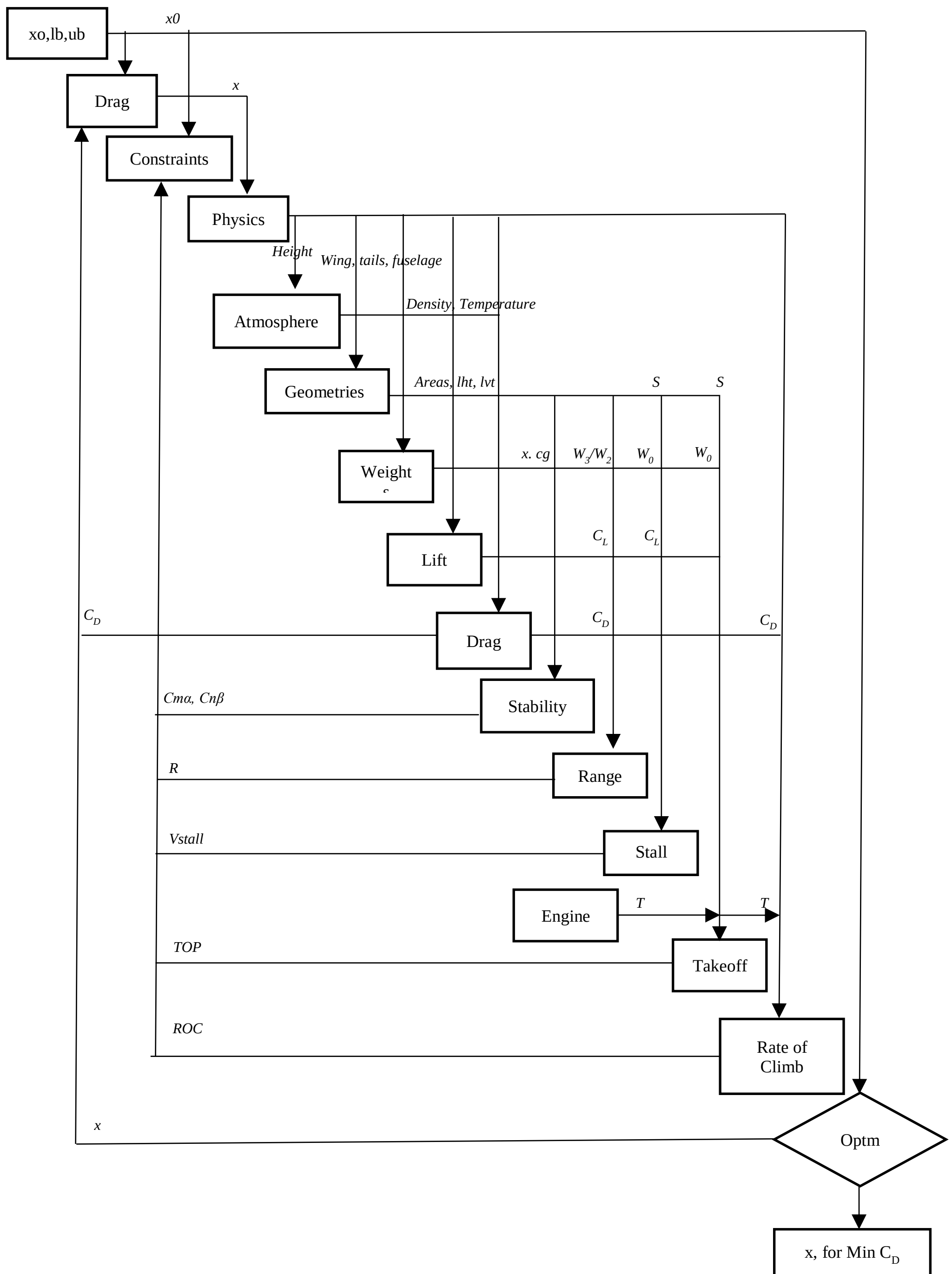



*Figure 2.* SQP optimizer

And the lift calculate by (14). 'Drag' function was estimated from the 'Geometries' function that includes wing areas, fuselage, tails, and landing gear was used as input for Drag coefficient. 'Drag' simply executes the 'Physics' function and passes the $C_D$ value obtained from 'Drag' back to 'Optm'.

The nonlinear constraint function, 'Constraints' simply calls 'Physics', then passes the results to a series of function detailed below.

'Stability' was done using 'Geometries' and 'Weights' as inputs and then the desired constraints of $C_{m\alpha}$, longitudinal stability and $C_{n\beta}$, directional stability are estimated. The range constraint value is obtained from the subroutine 'Range'. This is calculated via an implementation of the Breguet range equation (18). The range value is then subtracted from the desired range to derive a constraint value consisted with the SQP convention that satisfied constraints are negative. 'Stall Speed' was estimated by using wing loading as input and then the desired stall speed (50kts) was subtracted from the Vstall (x) model. The 'Take-off' constraint value was done by the wing loading to get the take-off parameter (TOP). The desired TOP was also subtracted from the TOP which was got from the (x) model. The 'Rate of Climb' constraint was calculated using thrust to weight ratio (T/W), from engine and drag to weight ratio (D/W) as inputs. (ROC) was subtracted from the desired rate of climb of 1500ft/min. The implementation of the optimizer will find the aircraft configuration (x) for minimum Drag coefficient at a given flight condition that satisfied the constraints.

## Optimization Results and Discussion

The convergence history is shown for total Drag coefficient demonstrating that the design optimization formulation is successfully converged by using the Sequential Quadratic Programming (SQP) algorithm in Figure 3.

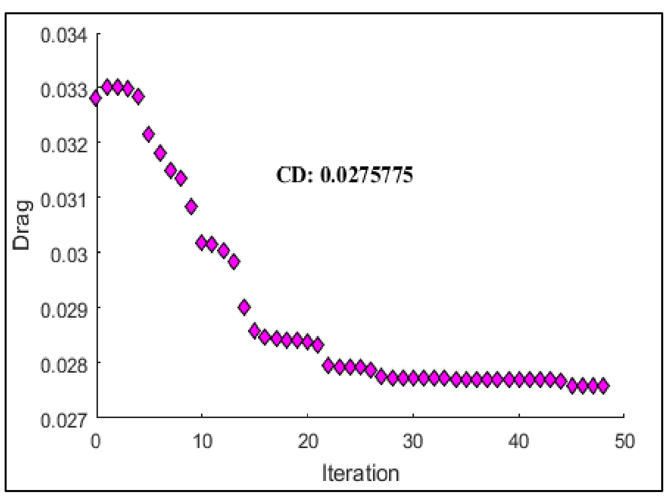



*Figure 3.* Convergence history of Objective Function

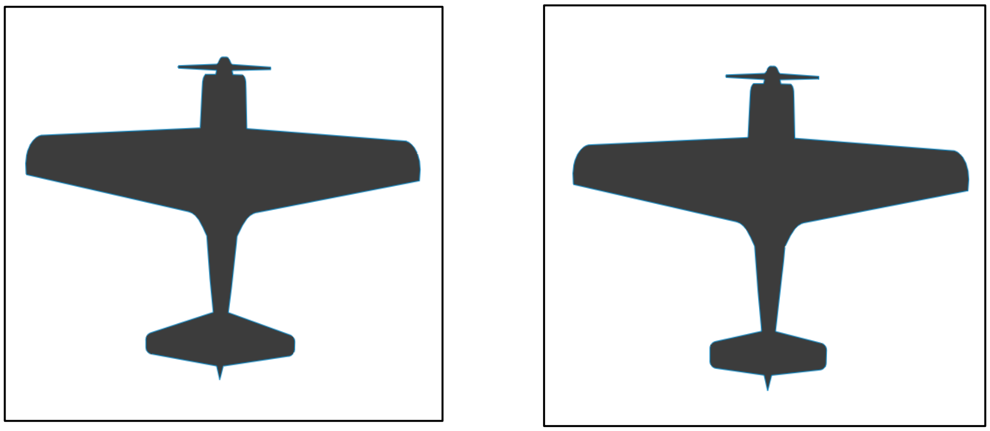
And optimized SSA configuration is shown in Table 4 for wing, tails configuration and some specifications. Also, the optimum configuration is shown in Figure 4 and Figure 5.

Table 4
*Optimum Configuration*

| **OPTIMUM CONFIGURATION** | | | |
|---|---|---|---|
| | **Baseline** | **Optimum** | **Units** |
| $W_0$ | 1200 | 1200 | lb |
| $W_e$ | 883 | 843.373 | lb |
| W/S | 10.2 | 8.5 | $lb/ft^2$ |
| Sw | 116 | 141.1765 | $ft^2$ |
| bw | 26 | 26.817 | ft |
| AR | 6 | 5.0939 | |
| $\lambda$ | 0.4 | 0.5102 | |
| bht | 10.0707 | 8.79 | ft |
| Crvt | 3.9183 | 1.723 | ft |
| bvt | 4.1476 | 4.8 | ft |
| Ctvt | 1.4271 | 1.127 | ft |
| Crht | 3.6 | 1.322 | ft |
| Ctht | 1.4 | 0.882 | ft |
| $C_{L_\alpha}$ | 4.8547 | 4.7385 | per radian |
| **$C_D$** | **0.0336** | **0.02757** | **Obj fun** |
| Range | 207.204 | 280 | nm |
| Max Range | 289.9714 | 395.67 | nm |
| **CONSTRAINTS** | | | |
| Range | 229 | 280 | nm |
| Stall Speed | 50 | 39.16933 | kts |
| Take-off Dist. | 1000 | 500 | ft |
| Rate of Climb | 1500 | 2487.192 | ft/min |
| $C_{m\alpha}$ | -0.58 | -0.3786 | Stick Fixed |
| $C_{n\beta}$ | 0.0717 | 0.06 | |

Optimum results show that the total Drag coefficient is reduced by 17.9% and the lower wing loading give more aerobatic performance and highly reduce lift induced drag. By the reduction of drag coefficient and higher lift from larger wing, the subsonic L/D of SSA rises from 12.1509 to 12.35. The increase in L/D causes Drag reduction and lower wing loading, giving less weight friction at cruise condition and improving maximum Range. Maximum Range is increased about

26.71% by the reduction of Drag coefficient. The minimization of total Drag directly improves the performance parameter such as Range at cruise speed. The stability constraints are also in the stable region and the aircraft empty weight is also reduced by the optimum configuration.



(a) Baseline Configuration (b) Optimum Configuration

*Figure 4.* Comparision for Baseline and Optimum Configuration from Top View.

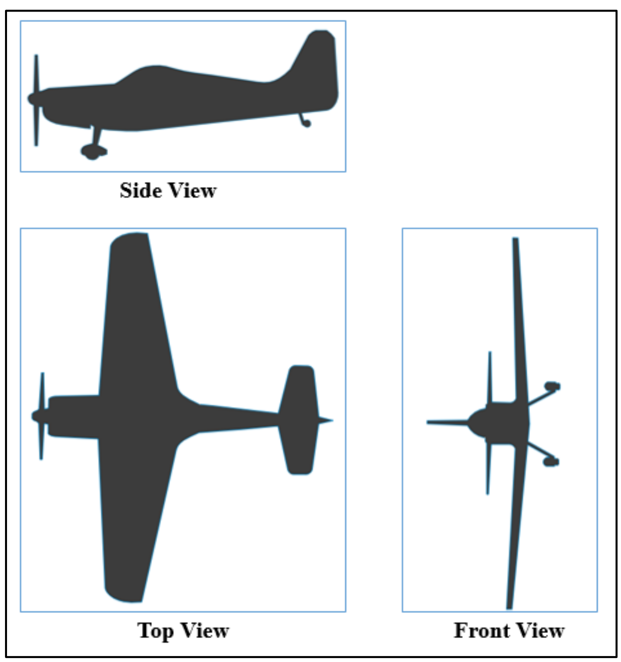



*Figure 5.* Optimum Configuration in Orthographic View.

## Conclusions

The conceptual design code for SSA was successful developed and validated with existing SSA. The validated SSA was used as a baseline model to optimize for minimum total Drag coefficient for better performance. Optimization of SSA was formulated using SQP algorithm to get optimum configuration. The optimum result shows that wing loading is lower than the baseline model's, which gives better aerobatic performance that makes reduction of induced Drag. The wing area is higher from 116 to 141. Although the Drag coefficient was minimized, the wing span and wing area were larger than the existing aircrafts. By the higher wing area, the wing Drag can be larger than the baseline. Also, the higher wing span does not experience as much as a loss of lift and increase of Drag due to tip effects as a low aspect ratio wing. In addition, the longitudinal and directional stability are also in the range of historical data. Overall, the optimization not only minimizes total Drag but also gives shorter take-off distance, increases maximum Range, and also increases Rate of Climb.

## Nomenclature

AR = wing aspect ratio
Aht = horizontal tail aspect ratio
Avt = vertical tail aspect ratio
bw = wing span
Crvt = vertical tail root chord
Ctvt = vertical tail tip chord
$C_{bhp}$ = specific fuel consumption
$C_D$ = total Drag coefficient
$C_{D0}$ = parasite Drag
$C_L$ = lift coefficient
$C_f$ = skin- friction coefficient
$C_{Dmisc}$ = miscellaneous Drag
$C_{DL\&P}$ = leakages and protuberances Drag
$C_{L_\alpha}$ = lift curve slope
$C_{ma}$ = longitudinal static stability
$C_{m\alpha f}$ = fuselage pitching moment
$C_{n\beta}$ = directional static stability
d = fuselage diameter
D = total Drag
e = Oswald efficiency
F = Form Factor, fuselage lift factor
Hp/W = horse power to weight ratio
K = lift induced factor
lb = lower bounds
L/D = lift to Drag ratio
$L/D_{max}$ = maximum lift to Drag ratio
M = Mach number
Q = interference factor =1
Re = Reynolds number
R = Range
ROC = rate of climb
SSA = Single Seat Aerobatic
S = wing area
Sht = horizontal tail area
Svt = vertical tail area
$S_{wet}/S_{ref}$ = wetted area ratio
$S_{exposed}$ = exposed wing area
$S_{wetf}$ = fuselage wetted area
ub = upper bounds
$V_{cr}$ = cruise velocity
W/S = wing loading
$W_0$ = Gross take-off weight
$W_{crew}$ = crew weight
$W_{pay}$ = payload weight
$W_f$ = fuel weight
$W_e$ = empty weight
$W_3/W_2$ = cruise end and start weight ratio
x0 = baseline model
x = optimized model
$\rho_{cr}$ = density at cruise
$\eta_p$ = propeller efficiency
$\eta$ = airfoil efficiency
$\Lambda$maxt = wing sweep angle at maximum airfoil thickness
$\lambda$ = wing taper ratio
$\eta_{ht}$ = dynamic pressure ratio at tail
$x_{acw}$ = wing aerodynamic center location
$x_{ach}$ = horizontal tail aerodynamic center location
xnp = neural point location
xcg = c.g location
$\frac{\partial a_h}{\partial a}$ = downwash effect

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