Task FlyMi

Dimitri Prencipe

September 2024

1 Engine and batteries

Before diving into engines, some assumptions and preliminary calculations are needed.

Given the data about the weight distribution between structure and payload and the limitations and goals of the competition, I assumed the structure's weight as 40 per cent of the maximum weight and, since we have the goal of maximising the payload weight, the total weight of the aircraft is expected to be 9.5 kg, to allow an extra margin, eventually for additional adjustments.

Also, for a small UA, these assumptions are made:

- wing load $\frac{W}{S}$: 10 $\frac{kg}{m^2}$
- $C_{D_{cruise}}$: 0.05

Now we can find the value for S: $S = \frac{W}{\frac{W}{S}} = \frac{W_{STRUCT}+W_{PAY}}{\frac{W}{S}} = 0.95\ m^2$

Assuming that the conditions of maximum thrust required take place while cruising on horizontal and unaccelerated flight, we find the value for T: $T_{MAX} = \frac{1}{2}\rho v_{MAX}^2 SC_{L_{cruise}} = 19.22\ N = 1960.05\ g$ (since the thrust data on the website is given in grams and 1 N = 102 g).

This value of T_{MAX} could be reached through many available configurations. I would go for the AT2814 KV900 long shaft brushless engine with the APC 11*5.5 propeller which, at maximum regime, gives 2216 grams of thrust. This choice satisfies the needed amount, giving also a good compromise between costs, thrust, consumptions and weight.

Some useful data, that will be used soon, about this configuration for each engine, are:

- Maximum thrust $T_{MAX} = 2216\ g$
- Power needed at full regime: $W_{MAX} = 503.64\ W$
- Weight : 108 g

At this point, we can consider the battery. Given the expected flight time of 15 minutes and the depth of discharge of 80 percent, assuming a 98 percent efficiency (η) of the distribution system , considering an average use of the propeller at 70 percent (425.55 W), the capacity is found using this formula:

$C_{BAT} = \frac{W_M \times T_{FLY}}{DOD \times \eta} = 137\ Wh = 9251\ mAh$

A possible choice to satisfy this requirement is 2 units of the Pink Performance Battery Li-Po Bashing (5000 mAh capacity, 590 g weight, 74.90 euros), with a total capacity of 10000 mAh (148) and weight $W_{BAT} = 1.18\ kg$

This battery is compatible with the engine since the battery has 4 cells in series and the engines requires 3-4 cells in series.

The remaining weight available for the payload is defined by this relation:

$W_{TOT} = W_{PAY} + W_{STRUCT} + W_{ENG} + W_{BAT}$

The result is 5.29 kg spare for the payload, considering a safe limit of the total weight at 9.5 kg.

2 Center of gravity

Even if high manouverability is requested, the project of an unstable aircraft would probably bring a big series of complications, so I would choose a stable profile. We will only consider one dimensional and relative positions with the AC as there are no data about the plane length or MAC, so the hypothesis of an ideal mono-dimensional profile is introduced where $x_{AC} = MAC/4$, so $\xi_{AC} = 1/4$ and $\xi'_{AC} = 0$ ($\xi = \frac{x}{c}$). Another two hypothesis are introduced: the AC is also the CG of the shear structure and the weight of the payload can be applied there too.

Under these hypothesis, the relative position of the CG is found from the definition of x_{CG}:

$x_{CG} = \frac{\Sigma m_i x_i}{M_{TOT}} \Longrightarrow \xi'_{CG} = \frac{\Sigma m_i \xi'_i}{M_{TOT}}$

Using the AC as reference and considering the weight of the motors is applied in $\xi = -1/4$ and the batteries' in $\xi = -1/6$: (the direction is defined positive heading towards the tail)

$\xi'_{CG_{empty}} = \frac{(-1/4) \times M_{ENG} + M_{STRUCT} \times 0 + M_{BAT} \times (-1/6)}{W_{TOT} - W_{PAY}} = -0.0538$

$\xi'_{CG_{full}} = \frac{(-1/4) \times M_{ENG} + M_{STRUCT} \times 0 + M_{PAY} \times 0 + M_{BAT} \times (-1/6)}{W_{TOT}} = -0.0239$

Those two values confirm that the aircraft is stable. The two values of ξ'_{CG} correspond to the stability margin: in fact, under the previous hypothesis $x_{AC} = x_N$, so $\xi_{AC} = \xi_N$, and the stability margin is defined as $sm = \xi_G - \xi_N = \xi_G - \xi_{AC} = \xi'_G$.

3 Wing surface, wing length, inertia tensor, stability margin

The surface of the wing was calculated at point 1.

To find b, some hypothesis are needed.

Since high manouverability is requested, the AR should be limited to values no bigger than 6/6.5, so my choice 5.5 as a good compromise between aerodynamic and structure efficiency.

So, $b = \sqrt{\lambda S} = 2.29\ m$

The drone inertia tensor is defined taking the CG as a reference, the x axis heading towards the front, the y axis in the same direction as the right semiwing and z axis as the vector product between x and y. These assumptions make these axes principal inertia axes, so the inertia tensor will be:

$$\begin{vmatrix} \mathrm{I}_{xx} & 0 & 0 \\ 0 & \mathrm{I}_{yy} & 0 \\ 0 & 0 & \mathrm{I}_{zz} \end{vmatrix}$$

In a first approach the aircraft cold be described as a system of a cylinder and a bar, one representing the fuselage and one representing the wing, jointed in their CGs. The engine is jointed to the wings in $(c/4 - c\xi_G)$ on the x axis.

Let's analyze every single term of the tensor:

- $I_{xx} = I_{fuselage_{xx}} + I_{wing_{xx}}$

  The fuselage is assumed to be a cylinder, which, seen from its top, gives an inertia contribution $I_{fuselage_{xx}} = \frac{M_{fus} d^2}{2}$ with d indicating the fuselage diameter.

  The wing is assumed to be a bar, so its contribution will be $I_{wing_{xx}} = \frac{M_{wing} b^2}{12}$

- $I_{yy} = I_{fuselage_{yy}} + I_{engines_{yy}} + I_{pay_{yy}} + I_{bat_{yy}}$

  The fuselage, seen from its side and considering Huygens' theorem ($I_G = I_x - md^2_{xG}$) for parallel axis and using L as the fuselage length and l as distance between the CG and the center of the fuselage gives a contribution of $I_{fuselage_{yy}} = \frac{M_{fus}(L^2+d^2)}{12} - M_{fus}(l^2 - c\xi'_{CG_{fus}})$

  Since, unlike in section 2, we are now considering the structure not as a single unit, its CG will not coincide with the CG of the single elements $(wing + fuselage)$, so $\xi'_{CG_{fus}} \neq \xi'_{CG}$

  The engines' contribution is : $I_{engines_{yy}} = M_{eng}(\frac{c}{4} - c\xi'_G)^2$

  The payload contributes with $I_{pay_{yy}} = M_{pay}(\frac{c}{4} - c\xi_G)^2$

  The batteries: $I_{bat_{yy}} = 2 \times M_{BAT} \times (l - \frac{5}{12}c)^2$

- $I_{zz} = I_{fuselage_{zz}} + I_{wing_{zz}} + I_{engines_{zz}} + I_{pay_{zz}} + I_{bat_{zz}}$

  The fuselage, engine, payload and batteries have the same contribution as in I_{yy}.

The wing's is: $I_{wing_{zz}} = \frac{M_{wing} b^2}{12} - M_{wing} l^2$

The stability margin sm is included at point 2.

4 Flaps

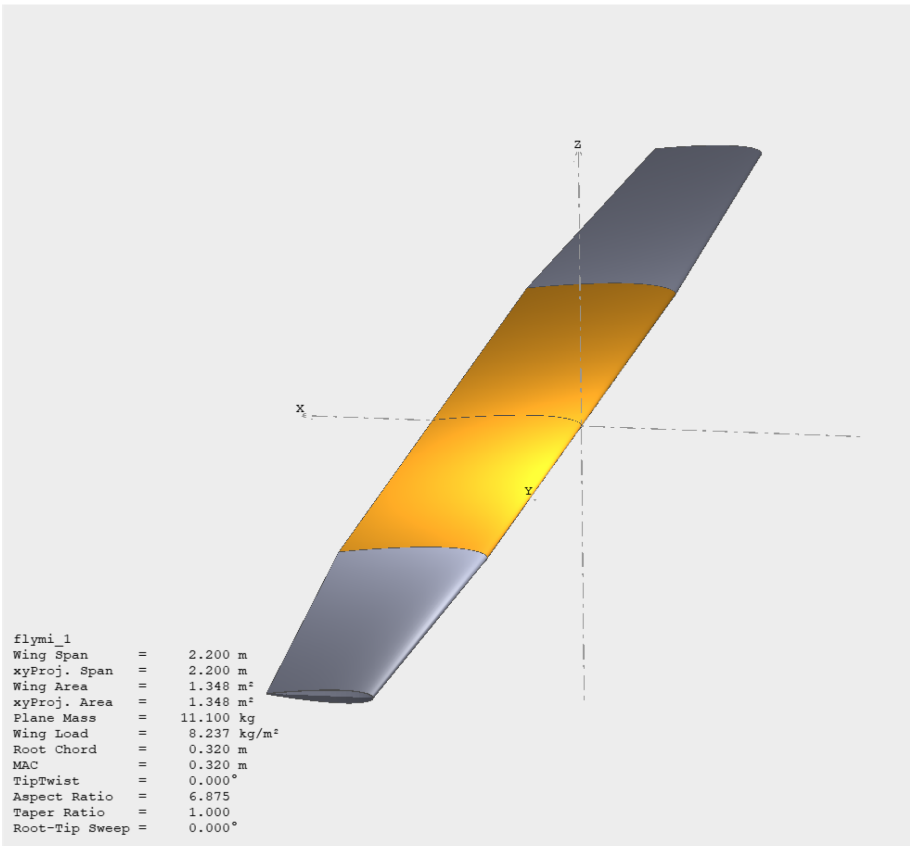

Figure 1: Designed wing

A primary flaps design could be done using the software xflr5.

Using the NACA 0012 design for low speeds (12 m/s), I created a wing with properties close to the ones calculated in the previous points; those datas are in the pictures that I reported. There are four possible flaps solutions: 0°,10°,15°,20°.

The analysis and graphs for these profiles are attached.

These results, especially the C_L/C_D, show how a 10° flaps configuration in the take-off phase would be optimal: in fact, for low values of C_D, so lower speeds, more lift is generated than any other configuration.

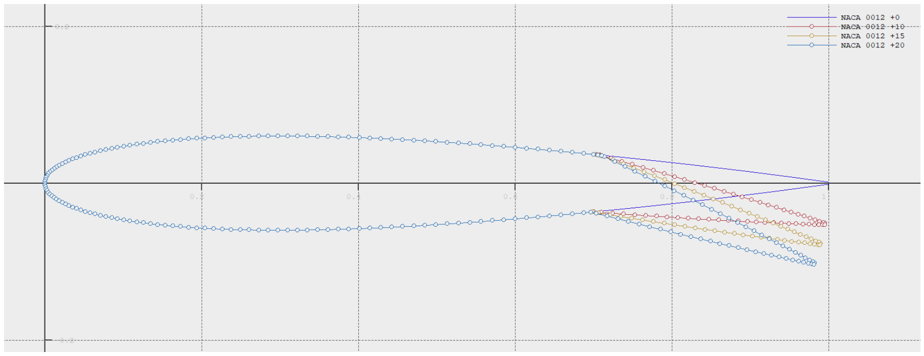

Figure 2: Wing flaps section

If higher lift is generated at lower speeds the take off speed will be lower, so will be the needed acceleration and ground run.

4.1 Considerations on slats

The use of leading edge slats could help to further improve the airplane's take-off performances, increasing the C_L values for small C_D despite giving technical complications to the project. In the figures 6-7 there is a comparison between the before selected 10° configuration and the same configuration with an addition of 5° leading edge slats.

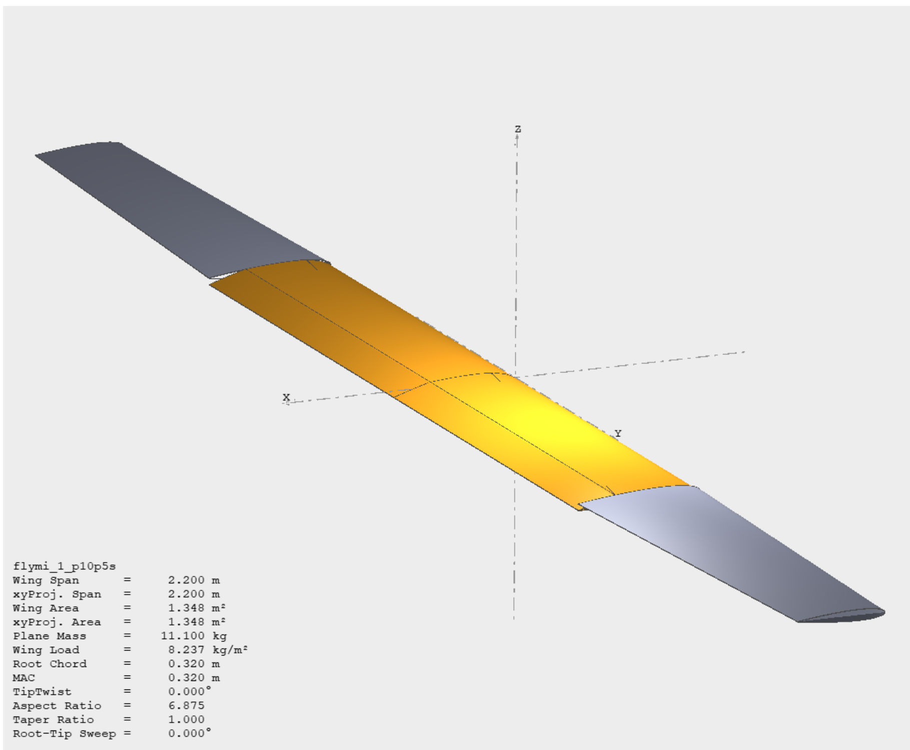

Figure 3: Wing with 10° flaps + 5° slats

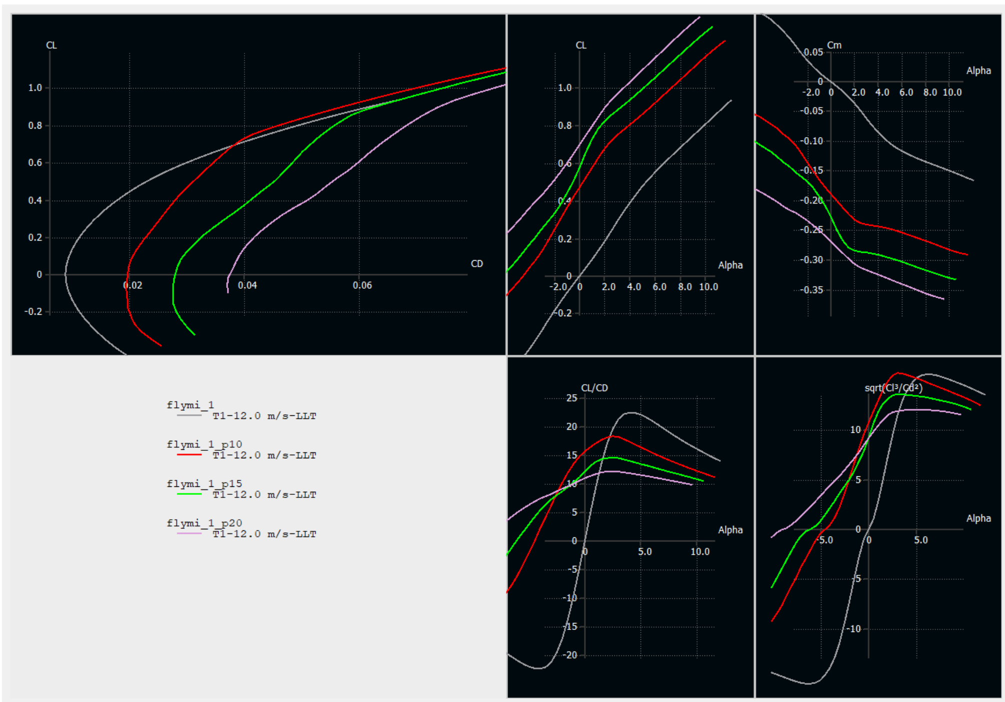

Figure 4: C_L/C_D,C_L/α,C_m/α,$(C_L/C_D)/\alpha$ curves

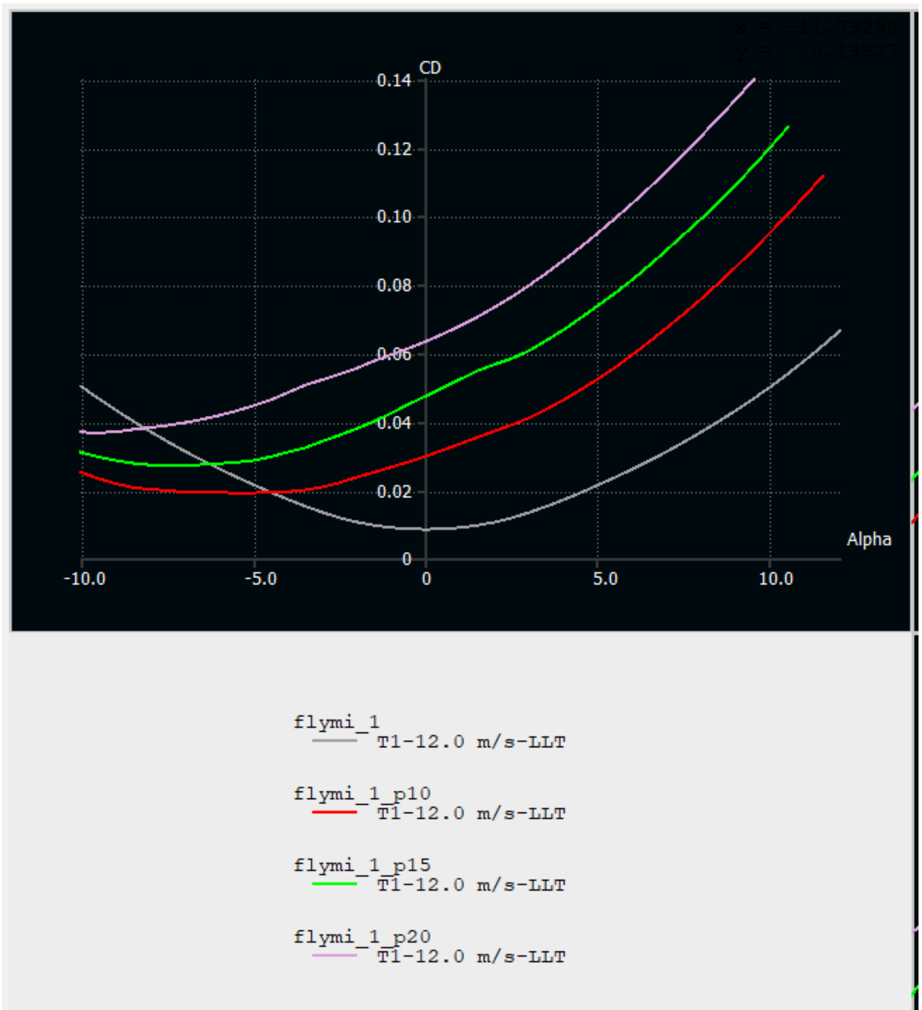

Figure 5: C_D/α

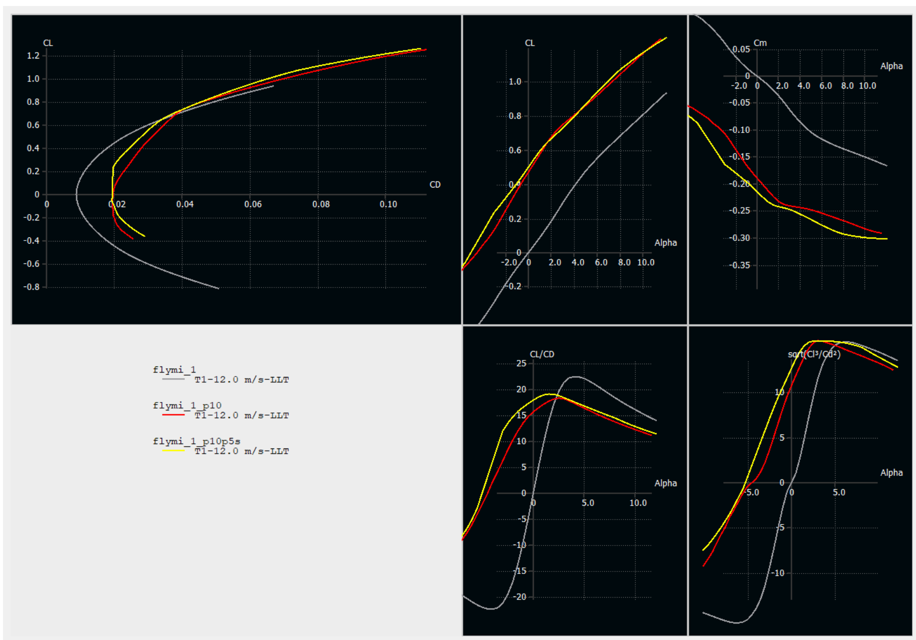

Figure 6: L.E. slat performances

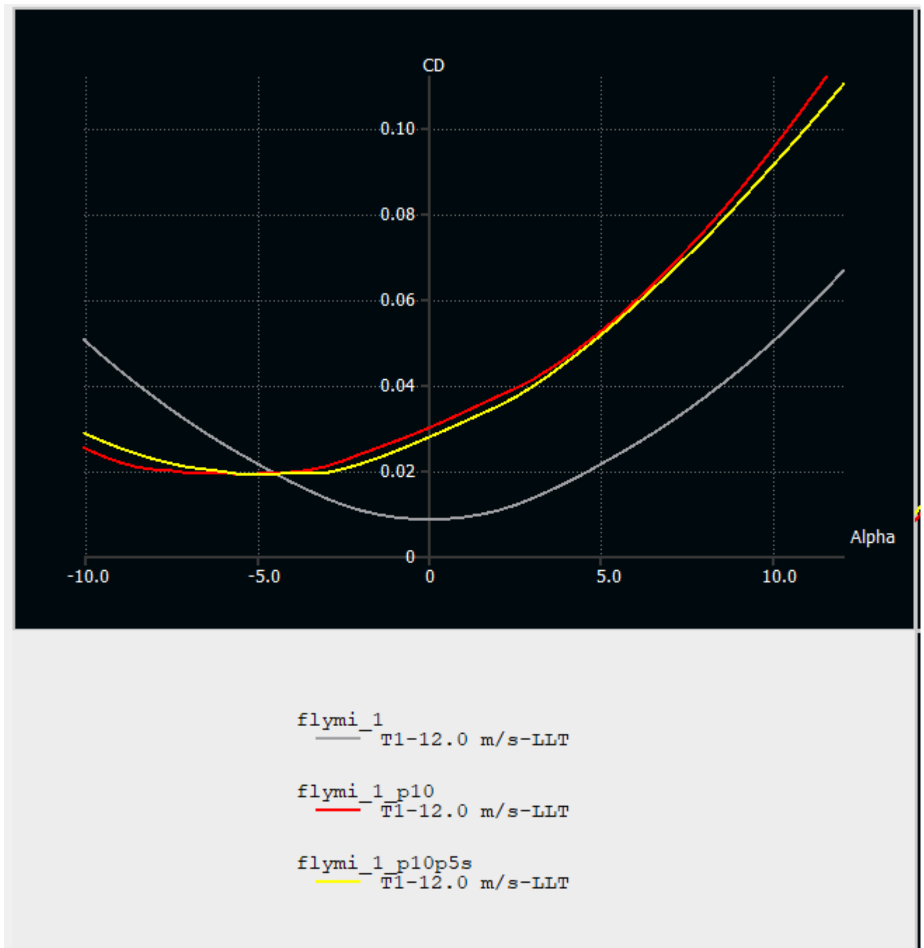

Figure 7: C_D/α

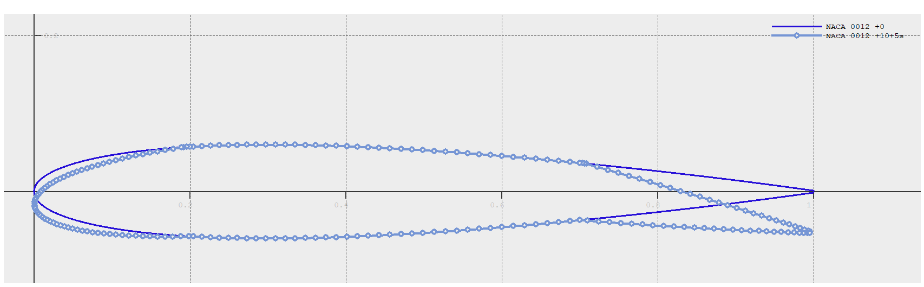

Figure 8: Slat configuration section