Project 2 – Conceptual Design

Second Project, entitled „Conceptual Design" includes two parts:

1. Estimation of selected (basic) geometric and weight parameters of an aircraft,
2. Drawing (in three views/projections) of the designed aircraft (glider, helicopter).

The Project should have a form described in Requirements, the drawing should be made on A3 or bigger format. Method of estimating basic geometric parameters depends on the aircraft's type (airplane, glider, helicopter), it's destination, size etc. A typical method for different types of aircrafts will be presented below. One should take notice, that methods presented below are much simplified and allow taking into account only selected geometric and mass related characteristics.

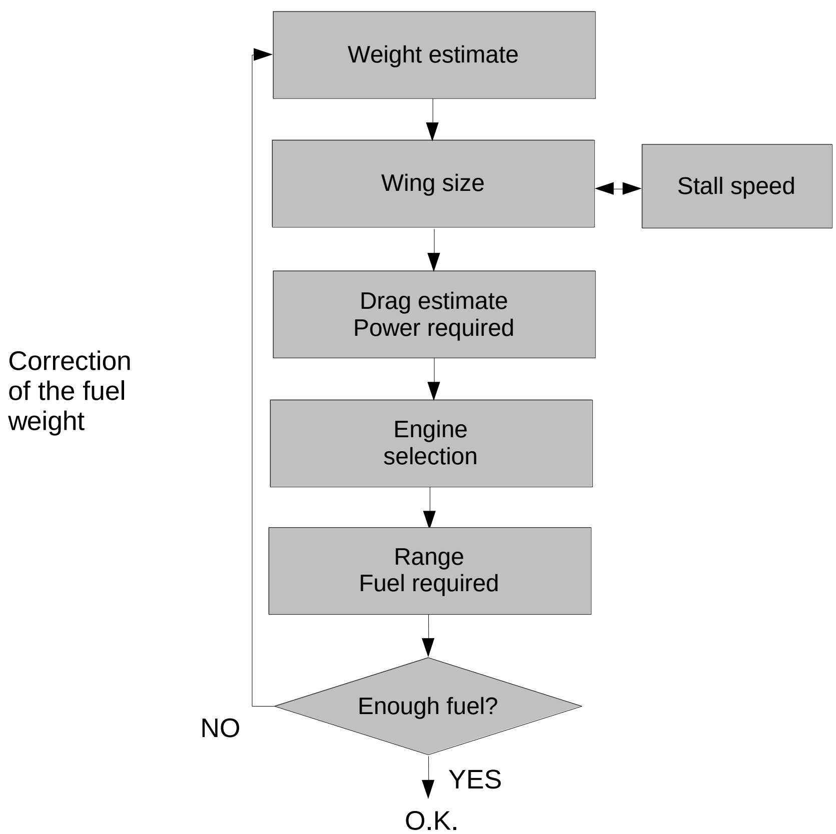

Fig. 1 - Scheme of initial estimation of weight

1 Aircrafts

When estimating basic geometric characteristics we assume, that the following requirements are to be met [1]:

- payload,
- cruise airspeed,
- operational ceiling,

- range or endurance
- type of regulations used for the designed aircraft (e.g. CS-23).

Moreover, based on statistic data we find some of dimensionless weight, geometric and aerodynamic data (e.g. empty weight ratio, aspect ratio, Oswald factor, etc.) The result of this stage should be estimated total weight, weight of fuel and basic dimensions of the aircraft.

The process is shown on scheme (Fig. 1). It shows an example of iterative estimation of fuel weight and in consequence also other parameters. This scheme can be divided into the following stages:

1.1 Weight estimation

In order to do that, we need to gather data on dimensionless contribution of the structure (empty aircraft)

$$\overline{W_e} = \frac{W_e}{W_{TO}} \tag{1}$$

and the fuel ratio:

$$\overline{W_f} = \frac{W_f}{W_{TO}} \tag{2}$$

Dimensionless weight contributions should be found from statistic analysis (trend analysis). Graphs shown on Fig. 2-3 can also be helpful.

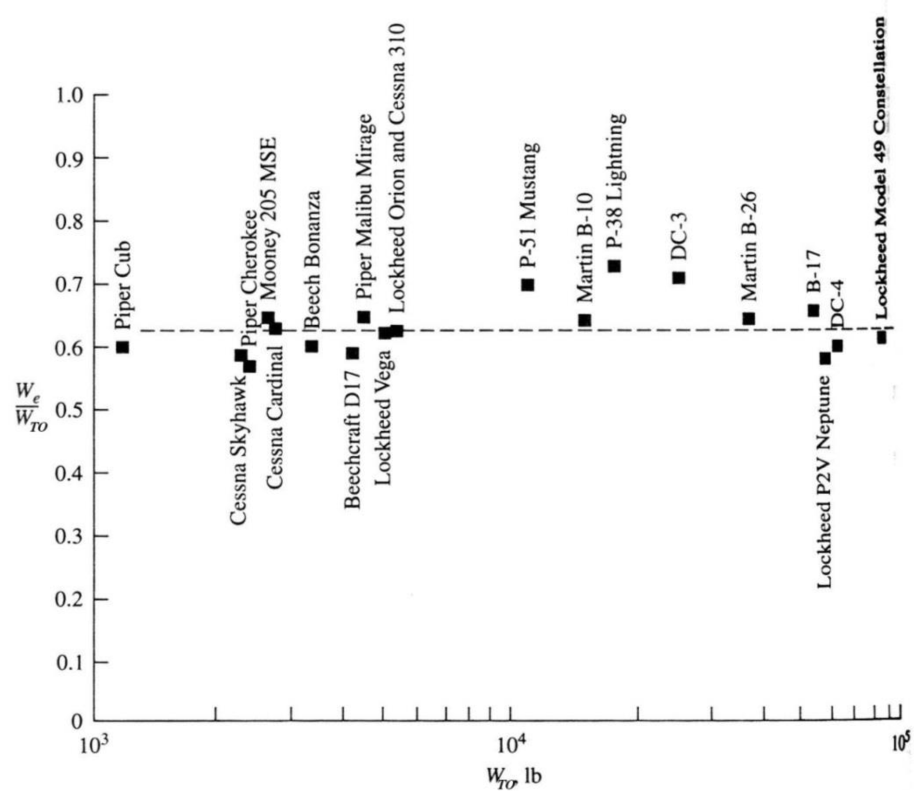

Fig. 2 - Contribution of empty weight in the total mass of piston propeller aircraft (*by Anderson* [2])

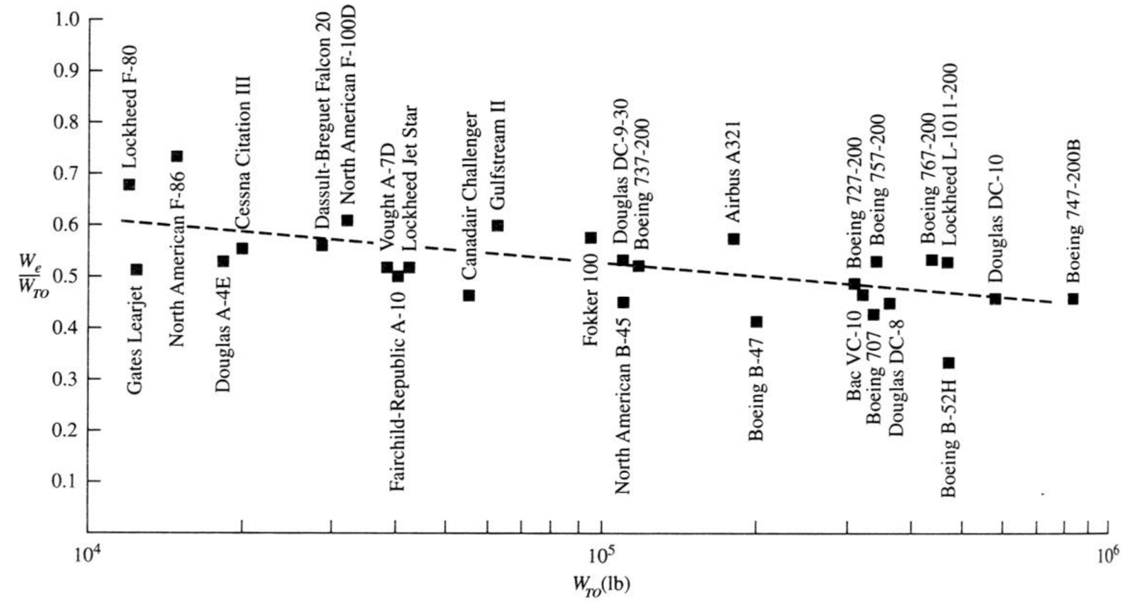

Fig. 3 - Contribution of empty weight in the total mass of jet aircraft (*by Anderson* [2])

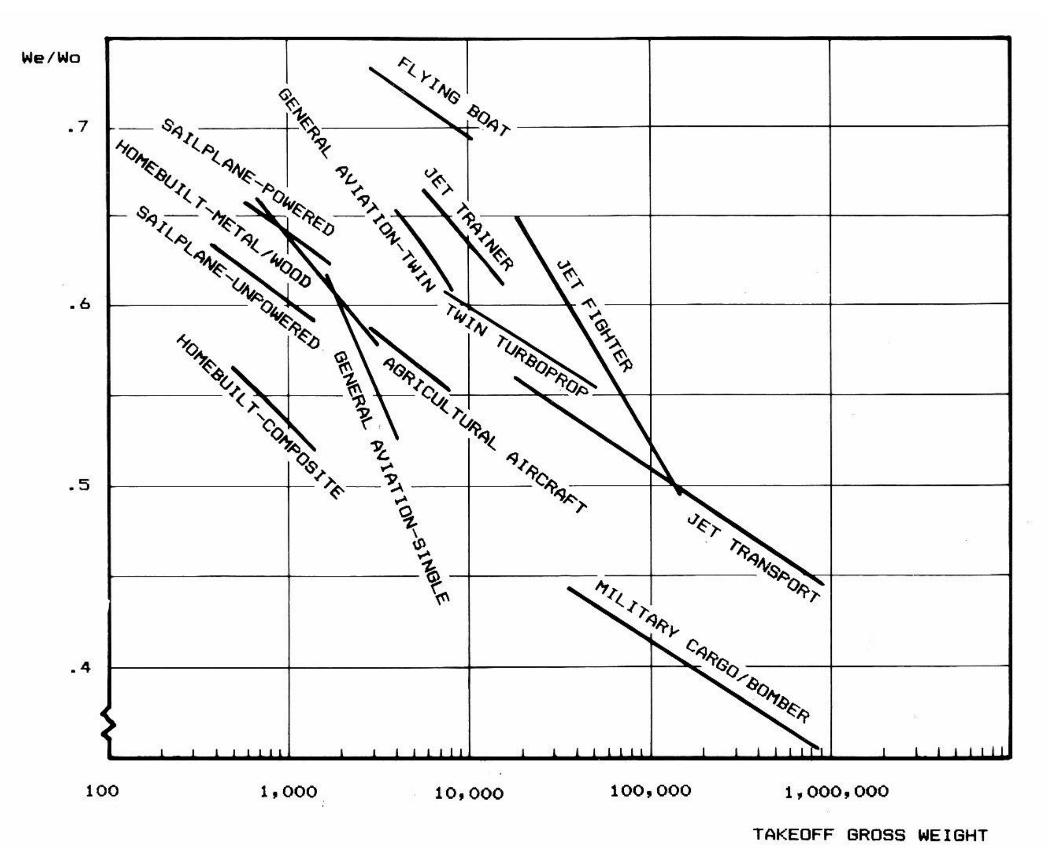

Fig. 4 - Contribution of empty weight in the total mass for different types of aircrafts (*by Raymer* [5])

Total weight can be obtained from the formula:

$$W_{TO} = \frac{W_p}{1 - \overline{W}_e - \overline{W}_f} \tag{3}$$

where:

W_{TO} – total weight (max. takeoff weight),
W_p – payload,
W_e – weight of an empty aircraft,
W_f – fuel weight.

Thus fuel weight, derived from (2), can be calculated as follows:

$$W_f = \overline{W}_f \cdot W_{TO} \tag{3a}$$

1.2 Wing surface

Wing surface can be calculated using the minimum airspeed condition (stall). The minimum airspeed is obtained from trend analysis and additionally we have to check regulations requirements concerning stall airspeed (CS 22.49(b), CS 23.49(c), CS LSA.5(b), CS VLA.49(b)). While at that, we should decide according to which regulations the designed aircraft will be certified, based on information included in chapter “A”.

Wing surface is obtained using the balance equation for the straight level flight at minimum airspeed:

$$P_z(V_{\min}) = Q \tag{4}$$

where:

P_z ($V_{\min}$) – lift force at minimum airspeed,
Q – aircraft weight.

Equation (4) after expanding has form:

$$\tfrac{1}{2}\rho S V_{\min}^2 C_{z,\max} = Q \tag{5}$$

Thus the wing surface:

$$S = \frac{2Q}{\rho V_{\min}^2 C_{z,\max}} \tag{6}$$

where:

ρ - air density at sea level,
$C_{z,\max}$ – max. lift coefficient.

To obtain wing surface, maximum lift force coefficient needs to be estimated. It is important to check for which configuration (clean, takeoff, landing, etc.) the stall airspeed condition is defined, because $C_{z,\max}$ strongly depends on super-lift devices (flaps, slots). Increments of wing section lift force coefficient for different types of flaps are shown in Table 1, and types of flaps are presented on Fig. 5.

Table 1 – Effectiveness of flaps (*by Cymerkiewicz* [3])

Type of super-lift device	$\frac{\Delta C_z}{C_{z,\max}}$ 100%
Leading edge slat	55…65
Leading edge flap	50…60
Plain flap	65…75
Split flap	75…85
Slotted flap	85…95
Split-external flap	85…95
Fowler flap	110…130
Double slotted flap	130…150

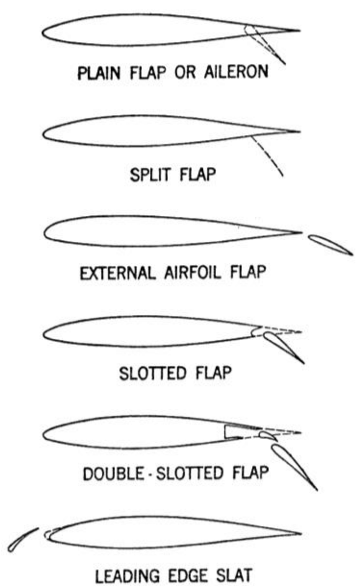

Fig. 5 - Types of flaps (by *Abbot* [4])

Lift force increment refers to a situation, when flap (or other super-lift device) includes the whole wing span. Such solution is used quite rarely. With partial flap span it can be assumed, that lift coefficient increment due to flaps is proportional to the part of the span contained by the flap (Fig. 6)

Thus the lift force gain can be obtained from formula:

$$\Delta C_z = \Delta C_{z,\,profilu} \frac{S_F}{S} \qquad (7).$$

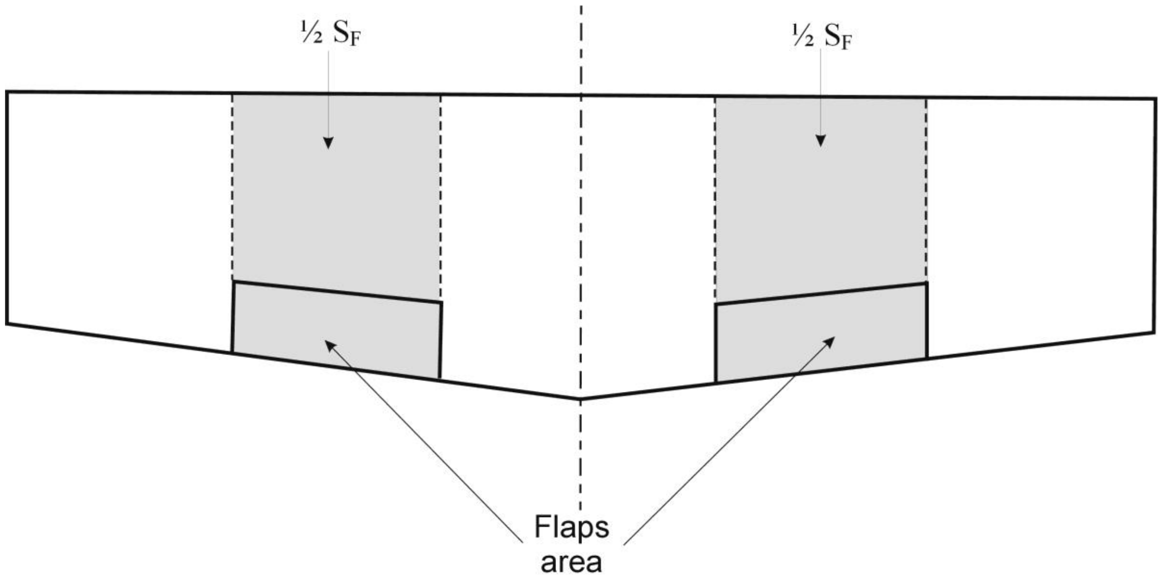

Fig. 6 - Definition of area contained by flaps

Example:

Data: $\frac{S_F}{S}$ = 40 %, plain flap ($\Delta C_z / C_{z,max\ profilu}$ = 0.7) , $C_{z,max\ wing\ section}$ = 1.6

$$C_{z,\max} = 0.9 \cdot 1.6 \cdot (1 + 0.7 \cdot 0.4) = 1.84$$

Notice:

coefficient 0.9 is an average decrease of maximum lift force coefficient, while calculating from 2D (wing section) into 3D (wing) flow; usually we can assume, that $C_{z,\ max\ wing\ section}$ = 1.5…1.6.

Aircraft's drag in cruise conditions can be obtained assuming, that total drag is a sum of minimum drag and induced drag:

$$C_x = C_{x,\min} + C_{x\,i} \tag{8}$$

Induced drag coefficient can be obtained from formula:

$$C_{xi} = \frac{C_z^2}{\pi \Lambda e} \tag{9}$$

where:

Λ - aspect ratio (b^2 / S),

e – Oswald's factor (for project's purpose we can assume values 0.6-0.8).

Formulas (8-9) allow to formulate, so called, analytical polar, which is simplified drag polar $C_x(C_z)$ and can be written as follows:

$$C_x = C_{x0} + \frac{C_z^2}{\pi \Lambda e} \tag{10}$$

Minimum drag coefficient can be estimated based on data included in Tables 2-3 and Fig.7.

Table 2 – Typical values of minimum drag coefficient (friction) referenced to the wetted area (*by Raymer* [5])

$C_{D_0} = C_{fe} \frac{S_{wet}}{S_{ref}}$	C_{fe} – subsonic
Bomber and civil transport	0.0030
Military cargo (high upsweep fuselage)	0.0035
Navy Force fighter	0.0035
Navy fighter	0.0040
Clean supersonic cruise aircraft	0.0025
Light aircraft - single engine	0.0055
Light aircraft - twin engine	0.0045
Prop seaplane	0.0065
Jet seaplane	0.0040

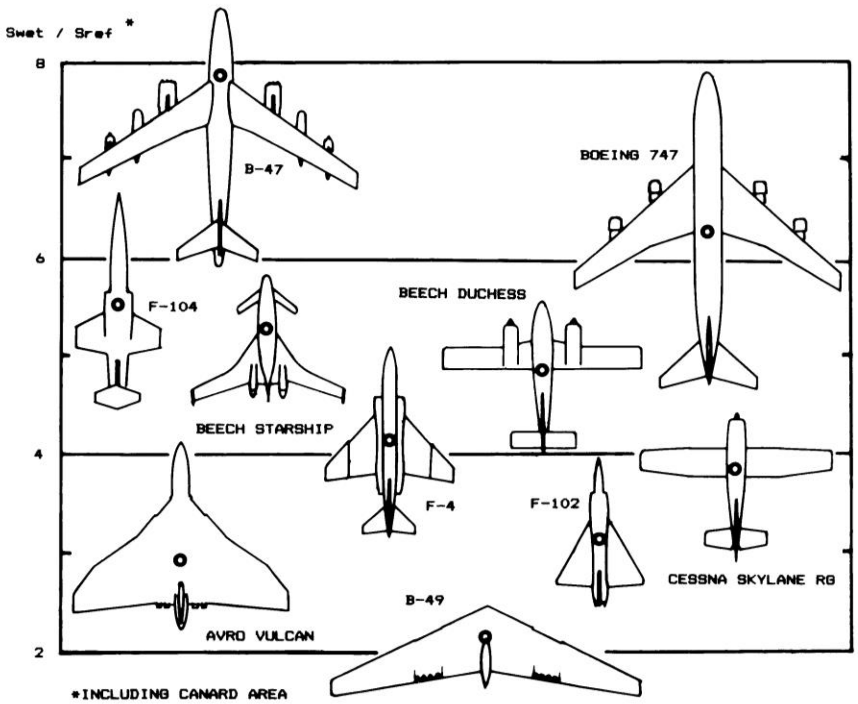

Fig. 7 - Wetted area to wing area (reference area) ratio for different types of aircrafts (b*y Raymer* [5])

Table 3 – Data on aircraft's resistances (by [6])

Type of an aircraft	$C_{x,\,\mathrm{min}}$
Monoplane and biplane with un-retracted landing gear, with struts, not well aerodynamically designed	0,10…0,05
Biplanes with un-retracted landing gear but better aerodynamically designed	0,06…0,04
Monoplanes with un-retracted landing gear but better aerodynamically designed	0,04…0,03
Biplanes with retracted landing gear, better aerodynamically designed	0,04…0,03
Modern monoplanes with retracted landing gear, better aerodynamically designed:	
Single engine, small, touristic and training	0,03…0,025 (even 0,020)
Big single engine – fighters (II WW)	0,025…0,020
Twin- and more engine – transport	0,030…0,025 (big aircraft even up to 0,020)
Twin- and more engine – fighter-bomber	0,025…0,020
Jets, very well aerodynamically designed	0,012…0,015
Flying wing	0,010

To obtain induced drag, lift force coefficient required for flight in cruise conditions needs to be calculated first. We use formula (5) to do that, but adapted to defined initially: airspeed and cruise level.

$$C_z = \frac{2Q}{\rho(h) V_{cruise}^2 S} \tag{11}$$

where:

$\rho(h)$ – air density at cruise level,
V_{cruise} – cruise airspeed.

Total drag force is obtained from formula:

$$P_x = \tfrac{1}{2}\rho(h) S V_{cruise}^2 C_x \tag{12}$$

We assume, that drag force is balanced by thrust of the engines, the aircraft is equipped with, so thrust required for level flight is equal to drag force. To derive the available thrust/power value and following this propulsion system characteristics, we have to take into account the thrust supply (and propeller efficiency). Required thrust obtained for cruise case is up to 75% of the available thrust (for light, single-engine aircrafts). Multi-engine aircrafts, especially passenger and battle aircrafts have a much bigger thrust excess. In airliners' case half of the engines has to be enough for climb flight. It means, that available thrust has to be over two times bigger than required thrust. In battle aircrafts thrust is often defined in relation to takeoff weight (Table 4).

Table 4 – Typical values of thrust to weight ratio and $C_{z,\max}$ for different types of aircrafts

Mission Requirement	$(C_{L_{\max}})_{TO}$	$(T/W)_{TO}$
Long Range	1.6–2.2	0.20–0.35
Short/Medium Range	1.6–2.2	0.30–0.45
Short TO & L	3.0–7.0	0.40–0.60
Light Civil	1.2–1.8	0.25–0.34
Combat Fighter	1.4–2.0	0.60–1.30

In propeller propulsion case we should additionally take into account propeller efficiency, which can be assumed as equal to 80% as the first approximation. So propulsion system power can be obtained from:

$$N = \frac{T \cdot V}{0.8} \tag{13}$$

where:

T – propulsion system thrust,
N – propulsion system power,
V – flight airspeed.

1.3 Wing geometry

The reference wing geometrical data must be established within preliminary project – wing area, wing span and mean aerodynamic chord (MAC). The wing area was computed in previous step, from formula (6), wing span b can be computed taking into account the assumed aspect ratio. To compute mean aerodynamic chord and it's position, the root chord C_R and taper ratio $\lambda = C_T/C_R$ must be assumed. Fig. 8 presents geometrical method to derive mean aerodynamic chord value and position.

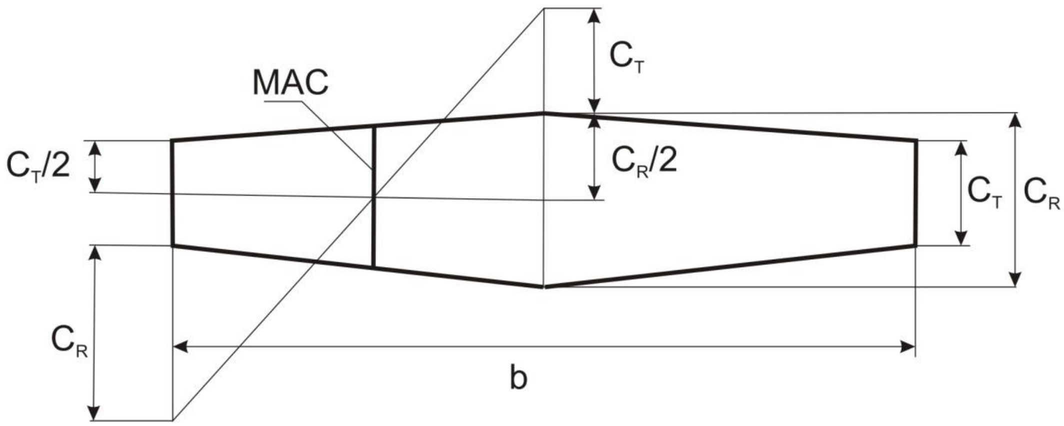

Fig. 8 – Mean aerodynamic chord – tapered wing

If wing planform is more complex, it can be replaced by equivalent tapered wing. C_R and C_T must be then well-fitting to preserve the same wing area and wing span. The MAC value can be computed from:

$$MAC = C_a = \frac{2C_R(1+\lambda+\lambda^2)}{3(1+\lambda)} \quad (14)$$

1.4 Selection of the propulsion system

Choosing propulsion system is done based on catalogue data [7]. Selected propulsion system has to have proper thrust (power), compatible with the values obtained in the previous step. Engines size and their unit fuel consumption should be also taken into account.

1.5 Range

Estimating of the aircraft's range begins with obtaining thrust or power required for level flight in conditions defined as cruise conditions. Required thrust is obtained straight from formula (12):

$$T_N = P_x \quad (15)$$

and power in propeller system case from formula (13):

$$N_N = \frac{T_N \cdot V_{przel}}{0.8} \quad (16)$$

From engines data we get unit fuel consumption q_e (Specific Fuel Consumption – SFC). For propeller engines SFC is measured in [kg/kWh] and for jet engines in [kg/daNh]. Having SFC and required power we can obtain consumption per time unit [kg/h] by use relations:

$$q_T = N_N q_e \qquad \text{lub} \qquad q_T = T_N q_e \tag{17}$$

We should remember about unit compatibility while calculating.

The next step is obtaining the theoretical flight endurance from formula:

$$T = \frac{m_f}{q_T} \tag{18}$$

Obtained value is decreased by 30 minutes reserve and by 10 minutes for each takeoff and landing, and multiplied by cruise airspeed to obtain aircraft's range:

$$L = V_{przel}(T - 50\,\text{min}) \tag{19}$$

Obtained range is compared to the value assumed in the requirements. If the difference is bigger than 10% we calculate fuel weight needed for the required range and we repeat the calculations by scheme from Fig. 1.

Notice: In aircraft's performance calculations formula (18) isn't enough and other relations are used (e.g. Breguet's formula) to reflect weight change during the flight. For first estimation needs, assumed simplifications are acceptable.

2 Gliders

When estimating basic geometric characteristics of a glider, we assume, that the following requirements are to be met:

- Maximum best glide ratio ($\frac{C_z}{C_x}$),
- Optimal airspeed (with maximum best glide ratio),
- Minimum descend,
- Economic airspeed (minimum descend).

Moreover, based on data (trend analysis) we determine glider's weight, including possible water ballast weight. This stage's result should be estimating the wing area of the glider and the aspect ratio necessary for obtaining required performance.

2.1 Theoretical basics

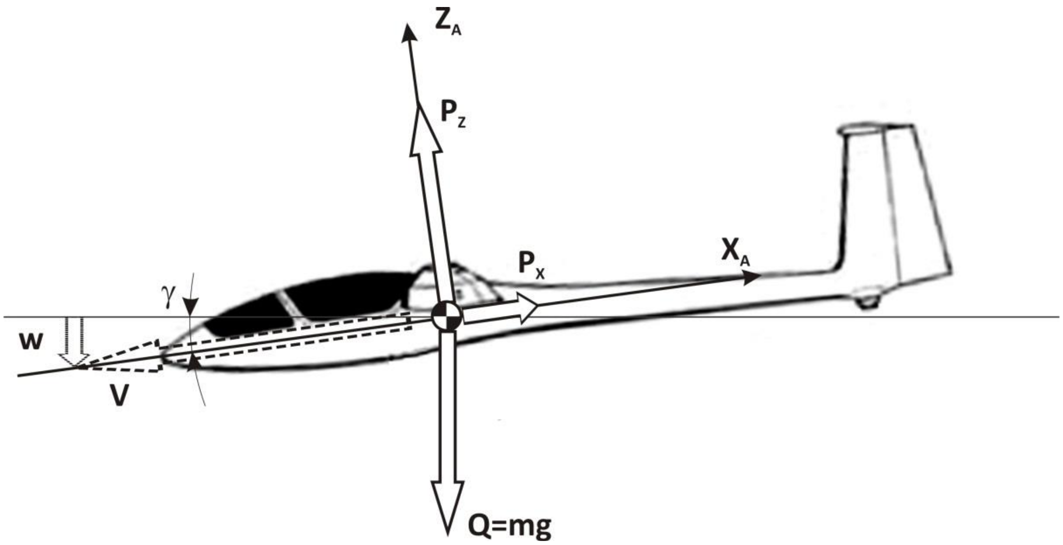

Fig. 9 – The forces acting on the glider during flight

Assume, that the path of flight is a straight line in the vertical plane (Fig. 9), while the velocity vector is sloped by path angle γ. The weight and airspeed are constant. The flight is in equilibrium state and in consequence, the forces acting on the glider are in equilibrium, which can be written as follows:

$$\begin{aligned} P_z - mg\cos\gamma &= 0 \\ P_x - mg\sin\gamma &= 0 \end{aligned} \tag{20}$$

Taking into account formulas for lift and drag:

$$\begin{aligned} P_z &= \tfrac{1}{2}\rho(h)SV^2C_z \\ P_x &= \tfrac{1}{2}\rho(h)SV^2C_x \end{aligned} \tag{21}$$

the airspeed *V*, path angle γ and sink rate *w* can be derived as a function of lift coefficient C_z (drag coefficient C_x is also a function of C_z).

$$V = \sqrt{\frac{2mg}{\rho S}\frac{1}{\sqrt{C_z^2 + C_x^2}}} \tag{22}$$

$$w = \sqrt{\frac{2mg}{\rho S}\frac{C_x^2}{\sqrt{\left(C_z^2 + C_x^2\right)^3}}} \tag{23}$$

$$\gamma = \arctan\left(\frac{C_x}{C_z}\right) \tag{24}$$

These data allow to draw so-called polar curve, which presents sink rate versus airspeed (Fig. 10).

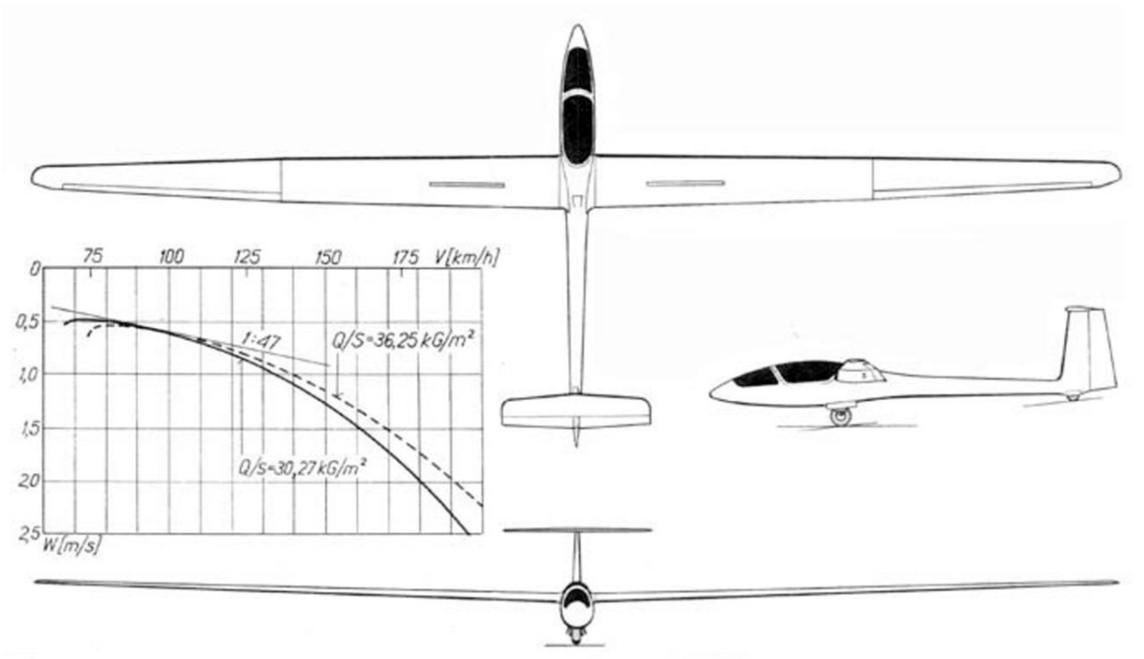

Fig. 10 –SZD-38A Jantar-1 and polar curve ("Skrzydlata Polska" 5/1974)

The values of economical (minimum sink) and optimal (maximum range – best glide) parameters can be derived using analytical polar drag in form (10). Assuming, that path angle is small, formulas (22) and (23) can be simplified, so economical and optimal airspeed and related sink rate can be computed from:

$$V_{ek} = \sqrt{\frac{2mg}{\rho S\sqrt{3\pi\Lambda_e C_{x0}}}} \quad , \quad w_{ek} = \sqrt{\frac{32mg\sqrt{C_{x0}}}{\rho S\left(\sqrt{3\pi\Lambda_e}\right)^3}}$$

$$V_{opt} = \sqrt{\frac{2mg}{\rho S\sqrt{\pi\Lambda_e C_{x0}}}} \quad , \quad w_{opt} = \sqrt{\frac{8mg\sqrt{C_{x0}}}{\rho S\left(\sqrt{\pi\Lambda_e}\right)^3}} \qquad (25)$$

2.2 Scheme of actions

Scheme of actions in this case is as follows:

1. obtaining wing area and aspect ratio,
2. estimating minimum drag from optimal airspeed requirement,
3. estimating aspect ratio based on maximum best glide ratio requirement,
4. comparing obtained aspect ratios from p. 1 and 3,
5. calculating remaining characteristic speeds and comparing them to requirements.

Obtaining wing area and aspect ratio is done in a similar way as in aircraft case, using formula (6). In order to do that, we need to estimate glider weight, based on trend analysis. Maximum lift force coefficient can be assumed from between (1.5-1.7). Minimum airspeed is estimated based on trend analysis, checking its compatibility with CS 22.49(b) regulations. Wing span is determined based on glider class and trend analysis. Aspect ratio is obtained from definition (b^2/S).

Estimating minimum drag comes from assumed optimal airspeed (25), which equals:

$$C_{x0} = \frac{1}{\pi\Lambda_e}\left(\frac{2mg}{\rho S V_{opt}^2}\right)^2 \tag{26}$$

Estimating aspect ratio is done based on maximum best glide ratio requirement, using optimal airspeed and optimal descend relation:

$$K_{\max} = \frac{V_{opt}}{w_{opt}} = \sqrt{\frac{\pi\Lambda_e}{4C_{x0}}} \tag{27}$$

thus:

$$\Lambda_e = \frac{4C_{x0}K_{\max}^2}{\pi} \tag{28}$$

where: $K_{\max}$ – maximum lift to drag ratio

Comparing aspect ratio obtained from definition and from maximum best glide ratio requirement shouldn't be much different, if assumptions are realistic. If the differences are significant, we should check the assumptions and calculations (focus on used units), if it doesn't solve the problem, consult with the teacher.

Calculating remaining characteristic speeds should be done using formulas (25). Obtained results should be commented.

3 Helicopters

Estimating basic geometric characteristics of a helicopter is much harder than in aircraft's or glider's case. It's connected to the fact, that the propulsion system is strongly coupled with the lifting device – rotor. More information about geometric and weight parameters can be found in [8]. Algorithm used to pick basic geometric parameters of the rotor is shown below, based on [9].

Assumptions:

- Maximum speed – V_{max}
- Maximum weight – W_{TO}
- Rotor disc load ($\omega = W_{TO}/\pi R^2$) – from statistic data (trend analysis)

3.1 Basic definitions:

One of the basic figures defining the rotor is the solidity coefficient, which is a relation between actual wing area and area of a circle formed by the rotor (rotor disc):

$$\sigma = \frac{A_b}{A} = \frac{NcR}{\pi R^2} = \frac{Nc}{\pi R} \tag{29}$$

where:

N – number of rotor blades,
c – chord of rotor blades,
R – rotor radius,
$A = \pi R^2$ – rotor disc area,
A_b – actual lift area of the rotor

The dimensionless thrust coefficient of the rotor is defined in similar way as other dimensionless force coefficients and can be written in form:

$$C_T = \frac{T}{\rho A (\Omega R)^2} \tag{30}$$

where:

T – rotors thrust.

We also define dimensionless blade-loading coefficient:

$$\frac{C_T}{\sigma} = \frac{T}{\rho A (\Omega R)^2} \frac{A}{A_b} = \frac{T}{\rho A_b (\Omega R)^2} \tag{31}$$

This coefficient can be described with a relation between average lift force coefficient on a rotor blade:

$$\frac{C_T}{\sigma} = \frac{\overline{C_L}}{6} \tag{32}$$

Because C_L takes average values (0.3-0.6), $\frac{C_T}{\sigma}$ takes values from range (0.05-0.1) – for this project's needs, we can assume 0.07 value.

3.2 Calculations:

We assume, that helicopter's weight is balanced with rotor thrust, and the blade tips can't exceed the critical speed (V_{KR} = 0.7 Ma). From formula (31) we get then:

$$\frac{C_T}{\sigma} = \frac{W_{TO}}{\rho A_b (\Omega R)^2} = \frac{W_{TO}}{\rho V_{KR}^2} \frac{1}{NcR} \tag{33}$$

Thus rotor lift area:

$$NcR = \frac{W_{TO}}{\rho V_{KR}^2} \frac{\sigma}{C_T} \tag{34}$$

Using data on rotor disc load, we can calculate its radius:

$$R = \sqrt{\frac{W_{TO}}{\pi \omega}} \tag{35}$$

Now we can obtain the aspect ratio:

$$\frac{R}{c} = \frac{NR^2}{NcR} = \frac{R^2}{NcR} N = \frac{R^2}{A_b} N \tag{36}$$

Obtained formula conditions aspect ratio from blades number (N). Of course N has to meet the following conditions: N has to be a natural number and $N \geq 2$. We choose the number of blades by comparing aspect ratio for different N values with aspect ratio of existing helicopters' blades.

4 Conceptual Design

The last stage of this project is drawing of the designed aircraft, glider or helicopter in 3 views. Drawing (designing) should be based on statistic data and materials helpful in deciding fuselage, tail and landing gear geometric.

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