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Elements of the synthesis method for the layout of a front-line aircraft

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Abstract. The article describes a technique that allows to form a planned projection of the aircraft using the minimum number of iterations (refinements based on aerodynamic and weight calculations results). The elements of the technique are described via example of the development (synthesis) of the layout scheme of a front-line aircraft with two engines and internal cargo compartments, performed according to the normal balancing scheme. Synthesis is carried out on the basis of certain pre-design parameters, limitations, as well as solutions set by the designer, based on the analysis of parametric ratios and statistical dependencies.

1 Introduction

When synthesizing the aircraft layout scheme the designer must determine the so-called “layout field”. Such “layout field” at the initial stage of the aircraft’s geometric appearance formation is its planned projection which determines its bearing capacity, as well as the squares graph, which determines both the volume itself and the wave increase of the aircraft aerodynamic drag.

2 Problem definition

The formation of the planned aircraft projection is the first thing that traditionally starts the development of a geometric appearance. In order to start the geometric constructions using methods described in [1] and [2], the main (recommended) geometric and weight design parameters are predetermined: S_w – aircraft wetting square; S_{bw} – basic wing square; $C^{\alpha}{}_y$ – lift characteristics, λ_{bw} – basic wing extension.

For the initial formation of the planned projection, it is necessary to determine:

- planned projection square – S_{plan}

- limitations on the transverse size of the planned projection (wing span) (where available).

- limitation on the longitudinal dimension of the planned projection (the length of the aircraft – L_a).

- fuselage width limit (wing chord distance z_b).

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- position of the wing along the length of the aircraft.
- center-of-gravity position (at the first approximation).
- possible location of cargo compartments, air intakes and engines (interconnected with each other).
- engine position along the length of the aircraft.
- air intakes position along the length of the aircraft.
- horizontal tail parameters (square – S_{ho}, and arm –L_{ho}).
- the position of the focus and center of sail of the planned projection.

3 Verification of boundary conditions

Then check the feasibility of the given boundary conditions.

S_{plan} determined by the condition of a predefined S_w and is a limitation precisely on the basis of not exceeding the recommended square of the wetting surface which simultaneously affects the weight and resistance of the aircraft.

Interrelation between S_{plan} and S_w is obvious (Fig. 1).

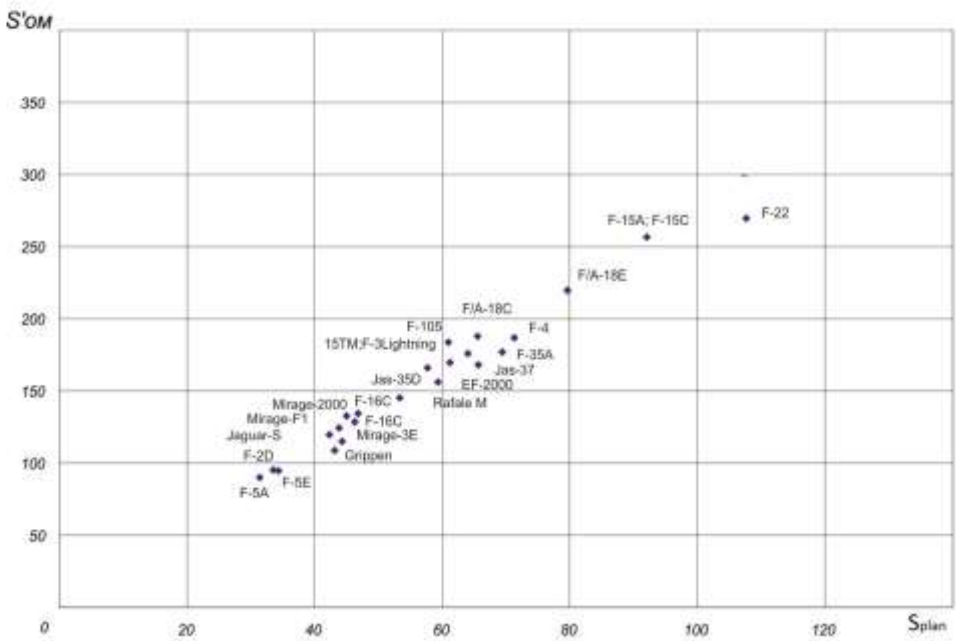

Fig. 1. Dependence of the wetting surface square on the square of the planned projection.

In this case, the square is shown without taking into account the contribution of propelling nozzles ($S`_w$ from $S`_{plan}$). Such a comparison is considered to be more correct, since the nozzle is an engine accessory and is not involved in the design parameters of the aircraft.

According to [3] and [4], depending on the coefficient of integrality: $K_{int} = 2 \cdot \frac{S_{plan}}{S_w}$, limitation on the square of the planned projection is “direct”. According to the analysis of statistical results for aircrafts generations 4, 4+ and 5, K_{int} varies in the range 0.7-0.8 depending on the degree of “flatness” of the fuselage and consequently the degree of its units “isolation”. I.e. when the designer imagines the scheme to be implemented the designer can choose K_{int}, which means define S_{plan} knowing S_w.

The limitation on the wing span is taken into account when the aircraft consist a need for its placement in the reinforced concrete shelter (RCS). Wherein two assumptions are applied:

1. The height of the end edge is taken at a height of 2-2.5 m from the ground.

The assumption is fair when the aircraft involves the use of the internal cargo compartment therefore the lower wing placement is unlikely because “low-winged aircraft” scheme is not rational from the point of view of constructive-power linkage.

2. RCS profile is considered to be a given circle with a diameter equal to a given width (B_{RCS}) with center at ground level.

The assumption is true because the results of a comparative analysis of the given in the proposed way RCS profile with real results showed minimal differences lying within the limits of engineering error.

In this case the equation of the RCS internal contour can be written down as:

$$\frac{Z^2}{\left(B_{\mathrm{RCS}}/2\right)}+\frac{Y^2}{\left(B_{\mathrm{RCS}}/2\right)}=1 \tag{1}$$

Presenting the control “point” of the airframe unit (in this case the tip) with the coordinates Y_{gr} (height of the tip from the ground) and Z_{gr} (half-span of the wing l_w / 2), we can formulate the dependence:

$$\Delta \approx 1.1 \times \frac{\left[\sqrt{\left(B_{\mathrm{RCS}}/2\right)^2-y_{\mathrm{gr}}^2}-z_{\mathrm{gr}}\right]\times\left[\sqrt{\left(B_{\mathrm{RCS}}/2\right)^2-z_{A\Gamma P}^2}-y_{\mathrm{gr}}\right]}{\sqrt{\left[\sqrt{\left(B_{\mathrm{RCS}}/2\right)^2-y_{\mathrm{gr}}^2}-z_{\mathrm{gr}}\right]^2+\left[\sqrt{\left(B_{\mathrm{RCS}}/2\right)^2-z_{A\Gamma P}^2}-y_{A\mathrm{gr}}\right]^2}} \geq \Delta_{\mathrm{GS}} \tag{2}$$

Where: Δ_{GS} – the minimum allowable clearance from the airframe unit to the RCS inner surface regulated by Air Force general specifications.

If the wingspan 1_w defined for the start of the construction satisfies the conditions of placement in the RCS, then the work continues, if not the designer must either remove the limitation condition (RCS) or choose a span that satisfies the condition of placement in the RCS.

The basis for determining the position of the wing along the length of the aircraft is the idea that the designed aircraft should have the smallest (possible) supersonic resistance. Thus, it is necessary to strive to ensure that the position of the mid-section along the length of the aircraft would be the most rational. In accordance with [4] from the point of view of the minimum wave increase the preferred positions of the mid-section along the length of the aircraft will be a range of 60-65% of the length of the aircraft respectively for airplanes constructed according to the normal balancing scheme.

4 Determining the position of the wing along the length of the aircraft

To determine the position of the wing along the length of the aircraft, it is necessary to “bind” the position of the mid-section of the aircraft and the position of the wing. To do so an assumption is made that the position of the mid-section of the wing and the plane coincide [5].

Note: this assumption is acceptable because analysis of a number of graphs for front-line supersonic airplanes squares made in [4] showed that the true distances of the positions of the mid-section of the aircraft and wing (outer wing - OW) unmatched not more than 3 ... 4% of the Mean Aerodynamic Chord (MAC).

Based on the approach considered in [5] it was proposed to consider the position of the OW mid-section in the middle of the line 50% of the chords. Thus to determine the position of the wing (Fig. 2) along the length of the aircraft, it is necessary to determine the mid-section of the line for 50% of the OW chord of the. To determine the OW geometry (which in our case is different from the basic trapezium by the size of the ventral part) it is necessary to determine the “width of the body” of the fuselage, i.e. the transverse coordinate Z_b which defines the geometric (but not technological) boundary of the OW and the fuselage.

To determine Zb in current paper we suggest to use the design indicator proposed in [3] – S_{cons} (the ratio of the wing consoles square to the base trapezium square).

The relative square of consoles on the one hand determines the degree of “aerodynamic perfection” of the planned projection of the aircraft, i.e. what proportion of the planned (and hence the wetted) surface is aimed at creating a lifting force; on the other hand – the degree of “weight perfection” of the planned projection (the small value of S_{cons} / S_{plan} means that, with all else being equal, the relative and absolute share of the fuselage is large). With a large value (share) of the fuselage in the planned projection the aircraft structure weight will be more because The fuselage (in terms of specific mass) is the heaviest unit of the aircraft [6, 7, 8].

Note: in the present work the wing on the planned projection is formed taking into account the minimization of “nonlinearities” in the m_z^{pp} characteristic, i.e. a negative sweep of the wing trailing edge is realized according to [2].

Based on the analysis of Scons / Splan values for a number of aircrafts (Table 1), we can formulate recommendations for this criterion (to determine at a first approximation the parameters of the planned projection and limitations on Z_b) for aircrafts constructed according to the normal balancing scheme – S_{cons} / S_{plan} >0.5;

Then in order to define Z_b the formula is proposed:

$$Z_b = \frac{-B \pm \sqrt{B^2 - 4AC}}{2A} \tag{3}$$

where the coefficients for quadratic equation solution:

$$A = 1$$

$$B = -1\left(l_{вк}\left(1 + \frac{b_k}{b_0 - b_k}\right)\right)$$

$$C = \frac{l_{вк}^2}{4} + \frac{l_{вк}^2 \times b_k}{2(b_0 - b_k)} - \frac{S_{cons} \times l}{2(b_0 - b_k)}$$

Taking into account the definition of Z_b as well as the assumptions made above, a formula to define determining the position of a wing along the length of the aircraft has been developed:

$$x_{кр} = L_c \times \overline{Lb} - \frac{(b_0 + b_k)}{4} - \frac{z_б}{l_{вк}}(b_0 + b_k) - tg\chi_{пк}\left(\frac{l_{вк}}{4} + \frac{z_б}{2}\right) \tag{4}$$

Consequently it is possible to determine the coordinate of the beginning of the MAC (X_a) to aircraft length

$$x_a = \frac{(b_0 + 2b_k) \times l_{bw} \times tg\chi_{wp}}{4(b_0 + b_k)} \tag{5}$$

Then the position of the center of mass (X_t) is determined (at a first approximation). This value depends on the degree of static instability which is determined by the shape of the planned projection – the advanced shifts the aerodynamic focus forward along the flight; wide tail (lateral tail beams (LTB)) shifts the aerodynamic focus back.

Based on the analysis of the aerodynamic focus positions for a number of aircraft analogues (table 1) the position of the center of mass of an empty aircraft at the first approximation is proposed to determine the distance (0.25-0.4) b_a from the beginning of MAC depending on the specified degree of longitudinal static instability (m_z^{Cy} in practice no more than 6%) and aircraft balancing scheme [9, 10].

For aircrafts constructed according to the normal scheme with a fuselage of constant (in length) width X_T=0.28-0,3b_a because horizontal tail (HT) shifts the focus relative to the wing by 5%; with the fuselage expanding in the tail section and wide lateral tail beams (ΔZ_{LTB}>400 mm) X_T=0.3-0,35b_a because HT and LTB shift the focus position by about 10%.

Consequently

$$X_m^0 = X + X_A + (0.25...0.4)b_a \tag{6}$$

depending on assigned m_z^{pp} and aircraft balancing scheme.

Table 1. The values of the front-line aircraft geometric and aerodynamic parameters.

Aircraft	Beginning of MAC , m	Length of MAC , m	Center of gravity position , m	Focus position % MAC	Degree of longitudinal static stability %	Centre of wind profile position % MAC	Wing square , m2	Arm of HT , m /relative arm	HT square /relative square	Static moment of HT	Planned projection square , m^2	HT Planned projection square / relative square	S_{cons}/S_b
F-15	10.1	5.37	11.57	32.9	-5.5	33	56.5	6.4 / **1.28**	11.6 / **0.205**	**0.262**	92	11.6/ **0.126**	0,56
F-16	7.1	3.45	8.24	30	+3.1	52.75	27.9	4.27 / **1.237**	4.6 / **0.165**	**0.204**	46,6	4.6 / **0.1**	0,646
F-18E	9.72	3.45	10.84	21.73	+6.87	51.5	46.45	5.42 / **1.57**	10.7 / **0.23**	**0.36**	79,6	10.7/	0,689
F-22	8.85	6.765	10.86	27.5	+2.2	44.78	77.5	5.75 / **0.85**	12.6 / **0.163**	**0.14**	107,6	12.6/ **0.117**	0,542
F-35A	7.62	4.53	9.27	29.1	+7.3	44.15	42.7	4.6 / **1.015**	8.94 / **0.21**	**0.21**	69,6	8.94/ **0.128**	0,527
MiG-29	8.45	3.8	9.41	28.6	-3	51.3	38	5.3 / **1.39**	7.16 / **0.188**	**0.262**	64	7.16/ **0.112**	0,579
Su-27	11.35	4.66	13	35.4	0	53.86	62	5.23 / **1.122**	13.34/ **0.215**	**0.241**	107	13.34 / **0.125**	0,577

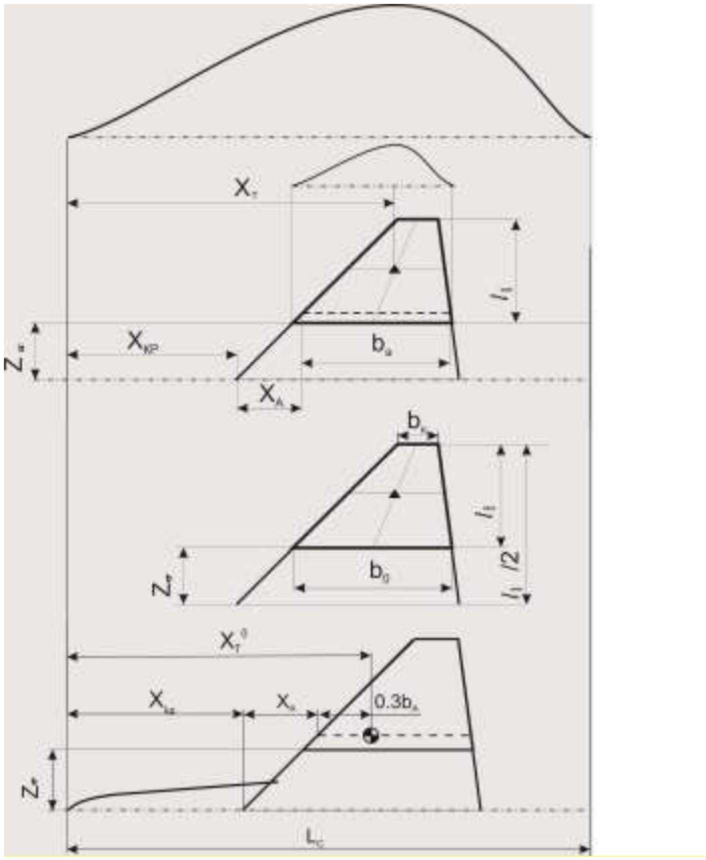

Fig. 2. Algorithm to find the wing position by means of an aircraft square chart.

To determine at the first approximation the position of the engines in length of the aircraft the assumption that the airframe is balanced is used (on the mass of the empty aircraft), and the equipment and systems installed in the fuselage front section (FFS) (shifts forward) have the main influence on the position of the center of gravity (at this stage) and power plant (shifts back). The task of theoretical contours of the FFS, the determination of the mass of the FFS equipment and the position of its center of mass are described in [6].

Contingent on above [7] remark the condition of an aircraft "centeredness" by "mass of empty" relative to the center of gravity aircraft can be written down as:

$$A \cdot G_{\mathrm{FFS}} = B \cdot G_{PP} \tag{7}$$

In accordance with [8]said above the expression for determining the position of the engine along the aircraft length at the first approximation can be written down as:

$$X^0_{\mathrm{eng}} = \left(X^0_T - X^{\mathrm{FFS}}_T\right)\frac{G^{\mathrm{FFS}}}{G^{PP}} - \left(X^{\mathrm{eng}}_T - X^0_T\right) \tag{8}$$

where X^0_T – the position of the center of an empty aircraft mass at the first approximation; X^{FFS}_T – the position of the FFS equipment mass center; G^{FFS} – FFS equipment mass; G^{PP} – power plant mass; X^{eng}_T – distance from the plane of entry into the engine to the position of the engine mass center.

In its turn $G_{PP} = G_{eng} \cdot k_{edmi}$ depending on the engine (the presence of a controlled nozzle to rotate the thrust vector) falls in the range 1.15…1.25, where k_{edmi} is the engine delivered mass index.

If after determining the position of the engine along the length of the aircraft, the situation $X^0_{eng}+L_{LD}>L^0_C$ is realized it is necessary to change X^0_T by decreasing X_W up to feasibility of $X^0_{eng}+L_{LD}\leq L^0_C$ observing the conditions of all dependences described earlier.

The determination of the engine position in width [10] is carried out in several stages.

The scheme of layout features (LF) dependencies is used; existing layout parameters and boundary conditions are described in [6].

For each version of LF E (the location of the cargo compartment) are determined by the corresponding variants of the LF B and LF C (the location of air intakes and the engine), respectively (Fig. 3)

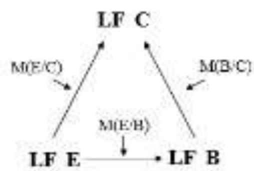

Fig. 3. Scheme to determine LF C.

Herewith the matching options described in table 2 are suggested:

Table 2. Matching options of LF C and Z_{eng}.

Layout feature	LF value	LF value description	Z_{eng} value
LF C (engine layout scheme)	1	Packaged	$< 0.75D_{nozzle}$*
	2	Spread	$\geq 0.75D_{nozzle}$*

* - the condition is accepted that the location of the engines with the distance between the nozzles less than $0.5D_{nozzle}$ is considered packaged, more than $0.5D_{nozzle}$ – spread.

The dependence of the engine position on existing layout parameters can be reflected in the form:

$$Z_{eng} = \begin{cases} B_{cc} + \dfrac{D_{nozzle}}{2} + \Delta & \text{variation 1 = LF E 1a} \\ \dfrac{Z_b}{2} \geq D_{nozzle} & \text{variation 2 = LF E 1b, 3} \\ \dfrac{D_{nozzle}}{2} & \text{variation 3 = LF E 1c} \end{cases}$$

The boundary conditions for the engine position is the expression of the form:

$$\frac{D\text{nozzle}}{2} \leq Z_{eng} \leq Z_b - \frac{D\text{nozzle}}{2} \tag{9}$$

Taking into account the described dependencies and observing the condition of "non-contradiction" between them using the expression (8) we determine the aircraft engines position at the first approximation.

To determine the position of the air intake the present work suggests using at the initial stage statistical data on the minimum value of the channel length [11] depending on the air intake and engine layout.

The following expression is suggested to calculate the channel length:

$$L_{\text{chan}}^{MIN} = k_K \cdot \overline{L_{\text{chan}}^{MIN}} \cdot d_{\text{in}} \tag{10}$$

where: is the index depending on the additional channel length, due to the use of adjustable panels (see table 3), statistical data.

Table 3. k_k index values due to the air intake type.

	Regulable air intake	Non-regulable air intake
k_K	1	0.93

$\overline{L_{\text{chan}}^{MIN}}$ is minimum (statistical) relative length of the channel equal to the ratio of the true length of the channel to the diameter of the channel at the engine inlet.

$$\overline{L_{\text{chan}}^{MIN}} = L_{\text{chan}}^{MIN} / d_{\text{in}} \tag{11}$$

d_{in} is the diameter of the channel at the engine inlet.

$\overline{L_{\text{chan}}^{MIN}}$ depends on the engine air intake position. Table 4 is based on the statistical data analysis.

Table 4. $\overline{L_{\text{chan}}^{MIN}}$ value for two-engine aircrafts.

Air intake position scheme / Engines position scheme	«packaged»	«spread»
«packaged»	6.5-7	6
«spread»	-	4.5

Therefore the air intakes position (at the first approximation) can be taken based on the engine entry distance position defined above and the minimum channel length using the expression [12]

$$X_{\text{in}}^{0} = X_{\text{eng}}^{0} - \overline{L_{\text{chan}}^{\min}} \cdot d_{\text{in}} \tag{12}$$

On this basis, observing conditions that the boundaries of the air intake should not exceed the limitation on the width of the fuselage (Z_b), and the fuselage should not end before the distance of the nozzle start, [13] one can determine the minimum square of the fuselage and wing consoles planned projection (Fig. 4).

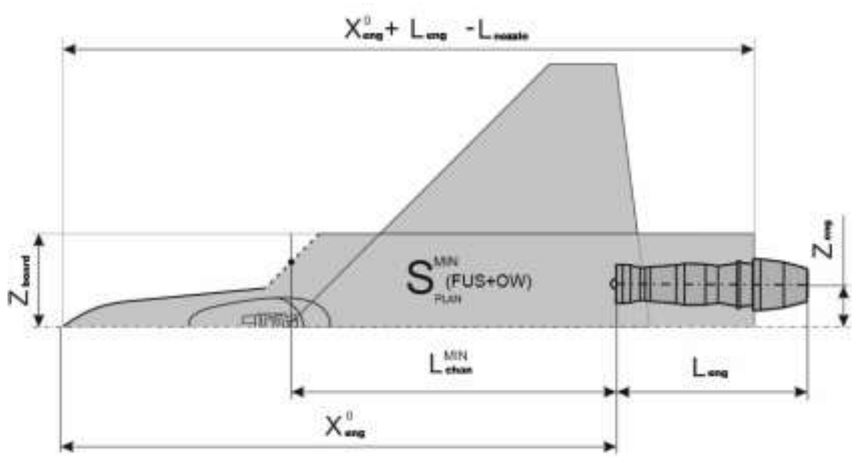

Fig. 4. Determination of the minimum square of the fuselage and wing.

To determine the square of HT it is suggested to use the dependence ($S_{HT}*L_{HT}$) from ($S_{plan}*L_C$), where: S^{HT} – HT square, L_{HT} – HT arm from aircraft mass center.

The physical significance of the proposed dependence is that (with all else being equal), the efficiency of fins which should provide stability and controllability, is determined by the ratio ($S_{HT}*L_{HT}$), while the measure characterizing the aerodynamic and inertial (dynamic) features of the aircraft's behavior in the longitudinal channel is ($S_{plan}*L_C$) [14]. This expression is true because it reflects the typical distribution of square and mass along the length of the aircraft for the typical layout of the aircraft constructed according to the normal balancing scheme.

As figure 6 shows it is possible to build [15] a linear between ($S_{HT}*L_{HT}$) and ($S_{plan}*L_C$), based on the analysis of the geometric parameters of a number of existing aircrafts:

$$S_{\mathrm{HT}} \cdot L_{\mathrm{HT}} = 0.0321 \cdot (S\mathrm{plan} \cdot L_C) + 3.3245 \tag{13}$$

Then in order to determine S_{HT}, it is suggested to use expressions (14) and (15).

$$S^0_{\mathrm{HT}} < S_{\mathrm{plan}} - S_{\mathrm{cons}} - S_{\mathrm{fus}} \tag{14}$$

I.e. the HT square should not exceed the difference between the previously determined S_{plan} and the minimum square of the fuselage and wing planned projection (see Fig. 4) otherwise the condition for ensuring the necessary design parameters will not be realized [16].

$$S_{\mathrm{HT}} = (S_{\mathrm{HT}} \cdot L_{\mathrm{HT}}) / L_{\mathrm{HT}} = 0.0321 \cdot (S_{\mathrm{plan}} \cdot L_C) + 3.3245 / L_{\mathrm{HT}} \tag{15}$$

Expression (15) is a consequence of (13), and L_{HT} is suggested to be defined depending on the presence or absence of aircraft lateral tail beams.

The expression is suggested to be used as a condition determining the possibility of the presence of LTB (16).

$$(Z_{\mathrm{b}} - Z_{\mathrm{eng}} + D_{\mathrm{eng}} / 2) >> 0 \tag{16}$$

When LTB is absent: L_{HT} is defined by the bracing distance of the rear engine mounting (depending on its design) because the last (by flight) fuselage strong frame will be installed at this distance. Therefore it is possible to fix the HT axis at no more than this distance.

HT axis (defining L_{HT}) is recommended (in accordance with the striving to reduce both the probability of HT flutter and minimize the HT hinge moments) to quater of MAC HT [17].

If LTB is present when L_{HT} is being defined it makes sense to be guided by the following factors:

- non-exceedance of aircraft recommended length – L_C;

- The most recommended (according to the terms of rigidity and strength) length of the LTB console section calculated from the last (by flight) fuselage force frame (defined by the bracing distance of the rear engine mounting) to the axis of HT rotation is no more than ~ 1 m (for S_{HT}=10-13 m^2 and G_{emp}=15-20 t).

5 Formation of the planned projection of the aircraft

After determining HT parameters and position an aircraft planned projection [19] is formed (at the first approximation). Therefore, it is possible to test its aerodynamic and dynamic balancing in accordance with a predetermined position of the aircraft mass center (see fig. 5).

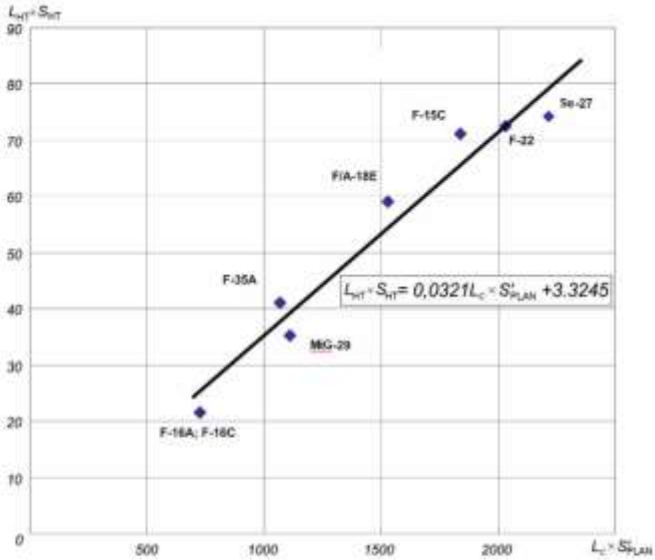

Fig. 5. Dependance of $S_{HT}*L_{HT}$ from $S_{plan}*L_C$.

Aerodynamic balancing is checked by drawing up a design scheme (representing a partition of the planned projection contour into flat panels) and determining (by means of the available software systems) the true position of the aircraft focus on the attack angle. Then a comparison of the focus position and mass center is made to test the implementation of a given degree of mz^{PP}.

Dynamic balancing is checked on the basis of the position related to the aircraft mass center of sail of the planned projection which in accordance with [2] is determined by the ratio S_1*L_1 and S_2*L_2 (see fig. 6). Dynamic balancing determines the aircraft behavior at large (α~90°) attack angles (17).

In order to guarantee the aircraft stability in these modes the center of sail should be located at the rear (along the flight) aircraft mass center. This condition is provided under the condition defined by the expression (17).

$$S_2 \cdot L_2 > S_1 \cdot L_1 \tag{17}$$

Verification of the feasibility of the boundary conditions is carried out to control the results of the method. The verification algorithm reduces a comparison of the results (geometric parameters) obtained at this stage with the previously defined design parameters. If the boundary conditions are not fulfilled the variation of the solutions determining the scheme synthesis is carried out in order starting with the least "important" (see Table 5) until the boundary conditions are fulfilled. If it is impossible to fulfill one or a number of conditions the variation of the scheme is not considered further.

6 Conclusion

By checking the feasibility of the boundary conditions at this stage (i.e. before the start of calculating the variety of options in terms of geometric and weight indicators, as well as flight technical requirements) the opportunity to reduce the number of "redundant" options and thereby reduce the labor intensity [8] and time to their further analysis is realized.

After checking the feasibility of the boundary conditions the further synthesis of the scheme is carried out - development of longitudinal and cross sections, placement of layout elements, calculation of mass and aerodynamic characteristics.

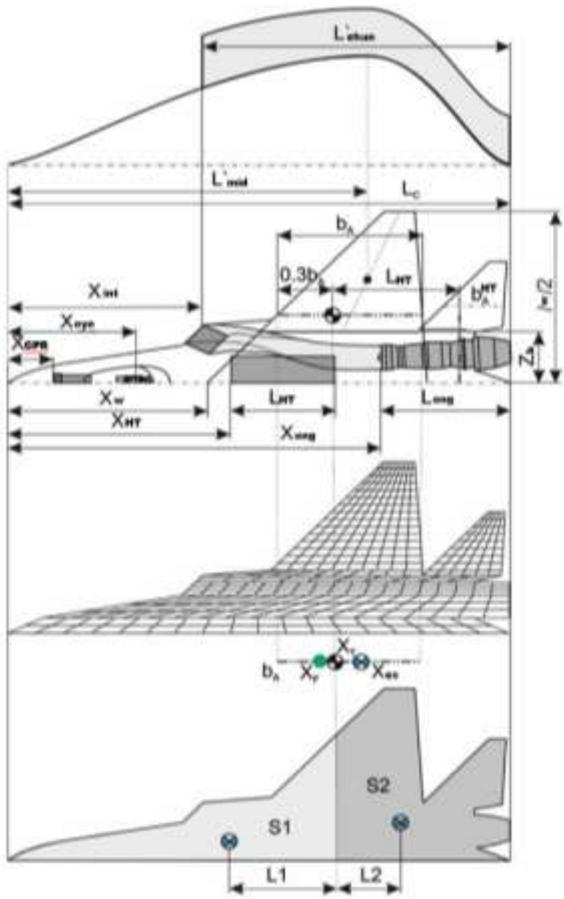

Fig. 6. General planned projection and alignment formation scheme.

Table 5. Layout item categories.

Order of parameter importance	The set of decisions defining constructive layout scheme (CLS)	Parameter (decision) taken during the formation of the CLS
«0»	Design parameters	S_{plan}, S_{W}; V_{C}
1	Geometrical characteristics of the aircraft	Λ_{C}, L_{C}, S_{mid}
2	Geometric parameters of the wing and fuselage	S_{bw}, S_{cons}; Z_{b}, λ_{bw}, $\chi_{bw} \rightarrow V_{bw}$
3	Layout features	Location of air intakes, engines, compartments, etc.
4	Variable parameters of the planned projection	X_{w}, X_{HT}, S_{HT}, X_{eng}, Z_{eng}

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