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МІНІСТЕРСТВО ОСВІТИ І НАУКИ УКРАЇНИ
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General arrangement of airplanes

THE COURSE PROJECT MANUAL

Загальна будова літаків

НАВЧАЛЬНИЙ ПОСІБНИК ДО ВИКОНАННЯ КУРСОВОГО ПРОЕКТУ

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This manual includes requirements to contents and execution of academic project. Technique and dependences for definition of take-off mass and geometrical parameters of units in zero approximation are given. Recommendations of choicing the load-carrying structure of the plane main units are given .

This manual is intended for students studying "Aviation and astronautics".

Fig. 12. Tab. 9. Bibliography: 13 names

The reviewers: ScD., Prof. Sergey A. Bychkov,
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Вказано вимоги, що висуваються до змісту і оформлення курсового проекту. Наведено методику та залежності для визначення злітної маси і геометричних параметрів агрегатів літака в нульовому наближенні. Дано рекомендації щодо вибору конструктивно-силових схем основних агрегатів.

Для студентів, що навчаються за напрямом "Авіація і космонавтика".

Іл. 12. Табл. 9. Бібліогр: 13 назв.

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1. THE EQUATION OF THE AIRPLANE EXISTENCE

For analysis and comparative estimation of various constructive decisions it is possible to use the formula for determination of airplane take-off mass

$$m_0 = m_k + m_{PP} + m_F + m_{EQ} + m_C + m_{CR}, \qquad (1)$$

where m_0 – take-off mass of airplane; m_k – mass of airplane structure; m_{PP} – mass of power-plant; m_F – mass of fuel; m_{EQ} – mass of equipment; m_c – mass of useful load (cargo); m_{CR} – mass of crew (generally mass of service load).

This equation is called ***the equation of mass balance***.

If all members of this equation to divide into m_0, then we receive

$$1 = \bar{m}_k + \bar{m}_{PP} + \bar{m}_E + \bar{m}_{EQ} + \bar{m}_C + \bar{m}_{CR}. \qquad (2)$$

Values $\bar{m}_i$ have the name of mass ratio.

This equation is called ***the equation of airplane existence***. It connects mass of units and parts with the general airplane take-off mass, and through them – all properties of the airplane which are provided with these masses.

At given level of aeronautical engineering development the quantitative increase of any property of the airplane results in increase in mass ratio which provides this property .

But as the sum of mass ratio is equal to unit, then the increase of one of them can be received only due to reduction of any other (provided that take-off mass m_0 = **const**). Hence, if to increase any airplane characteristic it is necessary to reduce another or another ones. If it is not made, then the sum of mass ratio will be more than unit. It testifies that at given level of aerospace science and engineering development the airplane with such set of characteristics cannot be constructed.

If to remove restrictions m_0 = **const**, then change of airplane characteristics can be received by not only redistributions of mass, but also changes of the take-off mass.

Change of one mass ratio of the components will result in changing take-off mass.

Conclusions from given analysis of the airplane existence equation: for given level of aerospace engineering development values of airplane parameters and characteristics cannot be any. Quantitative changes of some parameters and characteristics are certain to occur by changing parameters of others or by changing take-off mass. The complex set of their values should satisfy the equation of airplane existence.

2. DETERMINATION OF THE AIRPLANE PARAMETERS IN ZERO APPROXIMATION

2.1. TACTICAL-TECHNICAL REQUIREMENTS

Now, on the basis of arms though only initial, but nevertheless some knowledge in the field of aerospace engineering. Let's become the Main Designers for a while. First we shall determine, whether it is possible to construct in general on the basis of a modern level of aerospace engineering development new airplane on the basis of preliminary ***tactical-technical requirements*** (TTR) submitted by the customer. If it is possible, we shall determine parameters of the airplane even not in the first approximation, but only in zero one.

Usually for passenger airplanes number of passengers and range of flight are established, also type of the engine (piston, turboprop, turbojet). For transport airplanes mass of a cargo and also range are established. For maneuverable airplanes speed, mass of fighting load, radius of action, ceiling are established. Calculations in zero approximation are based on use ***of the statistical data*** for parameters and characteristics of already constructed airplanes ***of a similar class***.

In this case calculations consist of such stages :

- gathering and processing statistical data (flight, mass, geometrical characteristics) and power-plant parameters of airplanes-analogues;
- addition of set TTR;
- choice and substantiation of the airplane aerodynamic configuration;
- determination of take-off mass for the projected airplane;
- determination of engine parameters;
- determination of the basic geometrical sizes for airplane units;
- performance of drawings of general views for the airplane and its units;

- determination of load-carrying structures for the basic airplane units.

2.2. GATHERING AND PROCESSING STATISTICAL DATA

The analysis of statistical material enables to add and specify TTR to the designed airplane, to choose its configuration. In This case it is necessary to use the data of the airplanes being similar to projected one and having close flight perfomance and conditions of operation. It is possible to include only airplanes with the specified type of engines in statistical data . These data are placed in the statistical table typified kind of which is submitted (Tab. 1).

Such parameters and characteristics of airplanes are given in tables:

The flight data:

V_{max} – maximum flight speed;

$H_{V\,MAX}$ – flight altitude at maximum speed;

V_{CR} – cruising flight speed ;

H_{CR} – cruising flight altitude;

V_L – landing speed;

V_{TO} – take-off speed;

V_y – vertical speed;

H_{SC} – absolute ceiling;

L – flight range;

L_p – take-off ground run;

L_{TO} – take-off distance;

L_L – landing ground run.

The mass data:

m_0 – take-off mass of the airplane;

$m_{0\,max}$ – maximum take-off mass of the airplane;

m_L – landing mass of the airplane;

m_{EM} – mass of empty airplane;

m_{EQ} – mass of the equipment;

m_K – mass of a structure;

m_C– mass of payload;

n_{pass} – number of passengers;

m_F – mass of fuel.

The data of a power-plant:

engine ***type***;

n_{EN} – number of engines;

$P_0(N_0)$ – total thrust (power) of engines;

m_{EN} – mass of engine;

C_{p0} – starting value of specific fuel consumption per hour at $H = 0$, $V = 0$;

$C_{p\,H=V=}$ – value of specific specific fuel consumption per hour at altitude H and flight speed V.

The geometrical data of units:

S – wing area;

L – wing span;

χ – wing sweep angle;

λ – wing aspect ratio;

$\overline{c}(\overline{c}_{TR})$ – wing thickness ratio in root (tip) section;

η – wing taper;

D_F – fuselage diameter;

λ_ϕ – fuselage fineness ratio ;

$\bar{S}_{EL} = S_{EL}/S$ – relative area of ailerons;

$\bar{S}_{HS} = S_{HS}/S$ – relative area of horizontal tail unit;

$\bar{S}_{VS} = S_{VS}/S$ – relative area of vertical tail unit.

The column in Tab. 1 allows to judge about derivative parameters:

p_0 – wing loading at take-off; N/m^2 (or daN/m^2);

t_0 – starting thrust-to-weight ratio, N/kg (or daN/kg, H.P./kg or kW/kg);

γ_{EN} – specific mass of the engine, kg/N (or kg/daN);

k_C – factor of useful load.

Derivative parameters can be designed by the following formulas:

$$p_0 = \frac{gm_0}{S_W}, \quad t_0 = \frac{P_0}{gm_0}, \quad \gamma_{EN} = \frac{m_{EN}}{P_{01}}, \quad K_c = \frac{m_c}{m_0},$$

where $\mathbf{m_0}$ – take-off mass of the airplane, kg; $\mathbf{S_W}$ – wing area, m^2, $\mathbf{P_0}$ – total thrust of engines, N (or daN); $\mathbf{m_{EN}}$ – mass of the engine, kg; $\mathbf{P_{01}}$ – engine thrust , N (or daN); $\mathbf{m_C}$ – mass of useful load, kg. (Remark: daN = 10N).

If on the airplane piston or turbo-prop engines are mounted instead of size $\mathbf{P_0}$ it is necessary to take engine power $\mathbf{N_0}$ (in horsepowers or watts).

Table 1. Statistic data of airplanes-analogues

№	Name airplane, state, year manuf.	Flight data													Massa data								
		V_{max}, km/h		$H_{y\,max}$, km	V_{CR}, km/h	H_{CR}, km/h	V_L, km/h	V_{TO}, km/h	V_y, m/s	H_{SC}, m	L, km		L_{TO}, m	L_L, m	$m_o(m_{TO})$, kg	$m_{o\,max}$, kg	m_L, kg	m_{EM}, kg	m_{EQ}, kg	m_K, kg	m_c, kg	n_{PASS},.	M_F, kg

Data of a power-plant				Geometrical data														Derivative parameters					
Type and number of engines	$P_o(N_o)$, daN(kW)	m_{EN}, kg	C_{p0}, g/daN·h	$C_{PH\,V}$ g/daN·h	S, m^2	L, m	χ	λ	$\bar{C}(\bar{C}_{TR})$	η	L_F, m	D_F, m	λ_F	ΣS_m, m^2	S_{EL}, m^2	S_{HS}, m^2	S_{VS}, m^2	p_o, daN/m^2	t_o daN/kg	γ_{EN}, kg/daN	K_c	k_m	S_F

Statistical materials are taken from the literature, descriptions of airplanes, reference books, handbooks of airplanes, etc.

For each airplane given in statistical table it is necessary to have the circuit of its general view in three projections.

According to circuits of general views of airplanes it is also necessary to determine such sizes (average values): $\bar{S}_{WF} = S_{WF}/S$, where S_{WF} – wing area is occupied by fuselage; $\bar{S}_{WLD} = S_{WLD}/S$, where S_{WLD} – wing area is occupied by lift devices.

The development cycle of the tactical-technical requirements is carried out on the basis of the analysis for statistical materials consists in additions given in TTR for the projected airplane.

For passenger airplanes it is necessary to appoint number of passangers n_{pass}, altitude of cruising flight H_{CR} and cruising speed V_{CR}, for cargo airplane – mass of payload m_c, for maneuverable and high-speed airplanes – maximum speed of flight V_{max} and vertical speed V_y. For all types of airplanes distance or range L and take-off distance L_{TO} must be established.

For all types of airplanes, resulting from their purpose it is necessary to choose number of crew members n_{SCR}. For passenger airplanes stewards must be included into crew members. The number of stewards is determined from conditions: two stewards for compartment of the first class with number of passengers up to 30, one steward for 50 passengers in cabins of the second and third classes.

The established TTR are put down in Tab. 2.

Table 2. Tactical-technical data

$V_{H=11}(M_{H=11})$	$V_{max}(M_{max})$	V_{CR}, km/h	H_{CR}, m	L_{TO}, m

L, km	n_{pass} (m_c)	$Vy_{H=0}$, km/s	n_{SCR}, per.

2.3. CHOICE AND SUBSTANTIATION OF THE AIRPLANE CONFIGURATION

This stage of activity provides a choice of the form and a relative position of wing, fuselage, tail unit, type and number of engines; their arrangement for the projected airplane, type of the landing gear, determination of some geometrical parameters of wing, fuselage, tail unit by results of processing the collected statistical data of given airplanes.

As criterion of estimating (criterion function) majority of airplanes their take-off mass is taken, and as restriction – the performences which are set in TTR. In this case it is necessary to achieve extreme value of criterion of estimating the airplane, i.e. the best version of the airplane configuration will be the version with the least take-off mass, all other things being equal.

At this stage it is necessary to determine:

- wing planform (trapezoidal, swept, delta);
- wing and fuselage relative position (highwing monoplane, midwing monoplane, lowwing monoplane);
- type of tail unit (conventional, T-shaped, all-moving);

- type of landing gear (with rear auxiliary support, with nose support, bicycle);

- type and number of engines and their arrangement on the airplane (in fuselage, on fuselage, on wing, under wing, etc.).

Then according to statistical data it is necessary to determine and write down the basic geometrical parameters of the airplane in Tab. 3.

Table 3. Parameters of the airplane

λ	χ	η	$\overline{c}$	$\overline{b_{FL}}$	δ_{FL}	$\overline{S}_{AL}$	λ_F	D_F, m

$\overline{S}_{HS}$	$\overline{S}_{VS}$	λ_{HS}	λ_{VS}	χ_{HS}	χ_{VS}	$\overline{c}_{HS}$	$\overline{c}_{VS}$	η_{HS}	η_{VS}

They are :

- **wing**: aspect ratio λ, sweep angle χ, taper ratio η, airfoil thickness ratio $\overline{c}$, flap chord ratio $\overline{b}_{FL} = b_{FL}/b$, flap angles (take-off and landing positions) δ_{FL}, ailerons relative area $\overline{S}_{AL} = S_{AL}/S$;

- **fuselage**: aspect ratio λ_F , diameter D_F;

- **tail unit**: the relative area of horizontal surface $\overline{S}_{HS}$, the relative area of vertical surface $\overline{S}_{VS}$, aspect ratio of horizontal surface λ_{HS}, aspect ratio of vertical surface λ_{VS}, sweep angle of horizontal surface χ_{HS}, sweep angle of vertical surface χ_{VS}, airfoil thickness ratio of horizontal surface $\bar{c}_{HS}$, airfoil thickness ratio of vertical surface $\bar{c}_{VS}$, taper ratio of horizontal surface η_{HS}, taper ratio of vertical surface η_{VS}.

Then it is possible to start determination of the airplane take-off mass in zero approximation.

2.4. DETERMINATION OF THE AIRPLANE TAKE-OFF MASS

Take-off mass of the airplane for zero approximation is determined by the formula (2) received from the equation of mass ratio with using statistical data which we shall cite in somewhat different kind:

$$m_0 = \frac{m_C + m_{CR}}{1-(\bar{m}_K + \bar{m}_{PP} + \bar{m}_{EQ} + \bar{m}_F)}\,, \tag{3}$$

where m_0 – take-off mass of the airplane in zero approximation; m_C – mass of cargo (payload); m_{CR} – mass of crew; $\bar{m}_K$ – mass ratio of the airplane structure; $\bar{m}_{PP}$ – mass ratio of power-plant; $\bar{m}_{EQ}$ – mass ratio of the equipment; $\bar{m}_F$ – mass ratio of fuel.

Mass of cargo m_C for transport and military airplanes in the tactical-technical requirements (TTR) is established. Mass of a cargo for passenger airplanes is determined provided that the mass of one passenger with luggage is 90 kg. Therefore mass of a cargo for the passenger airplane is determined by such equation:

$$m_c = 90\, n_{PAS},$$

where n_{pas} – the number of passengers what is established in TTR.

Mass of crew m_{CR} is determined provided that the average mass of each crew member is 80 kg, and is calculated by the equation

$$m_{CR} = 80\ n_{CR},$$

where n_{CR} – number of crew members (pilots and stewards) is established on the basis of processing statistical data or mentioned above recommendations.

Value of mass ratio of fuel $\overline{m}_F$ is determined by the formula

$$\overline{m}_F = a + b\frac{L}{V_{CR}}, \qquad (4)$$

where L – flight range, km; V_{CR} – cruising speed of flight, km/h.

Factors **a** and **b** have values for airplanes with turbojet engines:

- **a** = 0.03...0.04 – for light nonmaneuverable airplanes (m_0<6000 kg);
- **a** = 0.05...0.06 – for all other airplanes;
- **b** = 0.04...0.05 – for subsonic airplanes;
- ***b*** = 0.13...0.14 – for supersonic airplanes.

Smaller values of factors correspond to airplanes with the greater take-off-mass.

Mass ratio of structure $\overline{m}_K$, power-plant $\overline{m}_{PP}$, equipment $\overline{m}_{EQ}$ for various types of airplanes are cite in Tab. 4.

Table 4. Mass ratio of airplanes

Types of airplanes	$\overline{m}_K$	$\overline{m}_{PP}$	$\overline{m}_{EQ}$
Subsonic passenger			
light	0.30...0.32	0.12...0.14	0.12...0.14
middle	0.28...0.30	0.10...0.12	0.10...0.12
heavy	0.25...0.27	0.08...0.10	0.09...0.11
Cargo			
light	0.30...0.32	0.12...0.14	0.16...0.18
middle	0.26...0.28	0.10...0.12	0.12...0.14
heavy	0.28...0.32	0.08...0.10	0.06...0.08
Other AV			
light	0.26...0.28	0.10...0.12	0.10...0.12
middle	0.22...0.24	0.08...0.10	0.07...0.10
heavy	0.18...0.20	0.06...0.08	0.06...0.08
Supersonic passanger	0.20...0.24	0.08...0.10	0.07...0.09
Maneuverable	0.28...0.32	0.18...0.22	0.12...0.14
Multipurpose short range	0.29...0.31	0.14...0.16	0.12...0.14
Sports	0.32...0.34	0.26...0.30	0.06...0.07
Agricultural	0.24...0.30	0.12...0.15	0.12...0.15
Light seaplane	0.34...0.38	0.12...0.15	0.12...0.15
Powered gliders	0.48...0.58	0.08...0.10	0.06...0.08

After calculating mass of passengers and crew, mass ratio of fuel on the basis of data of Tab. 5 mass ratios of structure, power-plant and equipment are chosen.

Then ***take-off mass of the airplane*** in zero approximation is calculated by formula (3).

The received result is to be compared to value of take-off mass for airplanes-analogues. If it is in a reasonable range, designing can be continued.

Then it is necessary to determine:

- mass of airplane structure m_K and its componemts (mass of wing m_W, fuselage m_F, tail unit m_{TU}, landing gear m_{LG});
- mass of fuel m_F;
- mass of power-plant m_{PP};
- mass of equipment m_{EQ}.

Mass of wing, fuselage, tail unit, landing gear is determined with these statistical data cited in Tab. 5. In this table mass ratio of the airplane units is shown in shares of total structural mass m_K. It should be stressed once more, that mass ratio of units is determined in shares of structural mass m_K, but not in shares of the airplane take-off mass m_0.

Hence, values of units mass are found by the formula

$$m_i = \overline{m}_i m_k . \quad (7)$$

The total sum of units mass ratio $\overline{m}_i$ for designed airplane must equal 1, that is $\sum \overline{m}_i = 1$.

Table 5. Mass ratio of airplanes units

m_0,T	Passanger and nonmaneuverable	10	50	100	50	200
	Maneuverable	-	5	10	15	20
$\overline{m_W}$	Passanger	0.393	0.396	0.391	0.384	0.377
	Nonmaneuverable	0.389	0.397	0.400	0.402	0.398
	Maneuverable	-	0.345	0.333	0.335	0.333
$\overline{m_F}$	Passanger	0.357	0.351	0.357	0.358	0.367
	Nonmaneuverable	0.346	0.342	0.332	0.328	0.332
	Maneuverable	-	0.410	0.408	0.403	0.400
$\overline{m_{TU}}$	Passanger	0.066	0.069	0.071	0.076	0.073
	Nonmaneuverable	0.083	0.081	0.083	0.079	0.077
	Maneuverable	-	0.064	0.086	0.082	0.080
$\overline{m_{LG}}$	Passanger	0.184	0.184	0.181	0.182	0.183
	Nonmaneuverable	0.102	0.182	0.185	0.191	0.193
	Maneuverable	-	0.161	0.173	0.180	0.187

Masses of parts and units are given in Tab. 6.

Table 6. Masses of the airplane parts and units

m_0 kg	m_C kg	m_{CR} kg	m_F kg	m_{PP} kg	m_{EQ} kg	m_K kg			
						$m_W$kg	$m_F$kg	m_{TU}kg	m_{LG}kg

2.5. DETERMINATION OF ENGINES PARAMETERS

Further it is necessary to determine ***starting thrust of the engine*** P_0 (or starting power N_0). It is determined on the basis of the collected statistical values of starting thrust-to-weight ratio t_0 (see Tab. 1). For this purpose it is necessary to establish value t_0 for the projected airplane.

Then it is possible to find starting total thrust of engines

$$P_0 = t_0 m_0 g, \quad (5)$$

where g = 9,8 m/c^2.

Further starting thrust of one engine can be determined on the basis of ***engines number n***

$$P_{01}=P_0 / n. \quad (6)$$

The engine of necessary type is chosen on value of starting thrust from ***the catalogue of engines***, and is written out its characteristics including mass. If the mass of the engine is not specified, it is calculated accoding to specific mass of the engine γ_{EN}

$$m_{EN} = \gamma_{EN} P_0 \text{ or } m_{EN} = \gamma_{EN} N_0.$$

Copy of general view for the engine with a designation of overall dimensions must be presented. It should be placed in the place allocated for it (in wing, in fuselage, on fuselage, etc.).

After determination of airplane mass characteristics the calculation of geometrical parameters of its units (wing, tail unit, fuselage, landing gear) must be given.

2.6. DETERMINATION OF GEOMETRICAL PARAMETERS FOR AIRPLANE UNITS

Determination of wing parameters

The wing area is found from an equation

$$S = m_0 g / 10 p_0 \,, \tag{8}$$

where $\boldsymbol{g}$ = 9,8 m/c^2, $\boldsymbol{p_0}$ – specific loading on a wing at take-off, is determined on the statistical data from Tab. 1.

Wing span is equal to

$$L = \sqrt{\lambda S} \,, \tag{9}$$

where value $\boldsymbol{\lambda}$ is taken from Tab. 3 .

Root (on axis of airplane symmetry) $\boldsymbol{b_0}$ and tip $\boldsymbol{b_K}$ wing chords are determined proceeding from values $\boldsymbol{S}$, $\boldsymbol{\eta}$, $\boldsymbol{L}$:

$$b_0 = \frac{S}{L} \cdot \frac{2\eta}{\eta + 1}, \tag{10}$$

$$b_K = b_0 / \eta \,. \tag{11}$$

where value $\boldsymbol{\eta}$ is taken from Tab. 3.

Wing mean aerodynamic chord (MAC) is calculated by the formula

$$b_A = \frac{2}{3} b_0 \frac{\eta^2 + \eta + 1}{\eta(\eta + 1)} \,. \tag{12}$$

Coordinate of MAC along a wing span is determined by relation

$$z_A = \frac{L}{6} \cdot \frac{\eta + 2}{\eta + 1}. \qquad (13)$$

Coordinate of MAC nose along an axis $\boldsymbol{0x}$

$$x_A = \frac{b}{6} \cdot \frac{\eta + 2}{\eta + 1} tg\chi_{LE}, \text{ or } x_A = z_A tg\chi_{LE}, \qquad (14)$$

where χ_{LE} – sweep angle of a wing leading edge,

$$tg\chi_{LE} = tg\chi + \frac{\eta - 1}{\lambda(\eta + 1)}. \qquad (15)$$

where χ – sweep angle of a wing quarter-chord line.

For delta wing with a trailing edge perpendicular to symmetry axis of airplane the formulae for determining $\boldsymbol{b_A}$ and $\boldsymbol{x_A}$ are simplifield:

$$b_A = \frac{2}{3} b_0, \qquad (16)$$

$$x_A = \frac{1}{3} b_0. \qquad (17)$$

Formulae (12) – (17) are correct for wings with straight lines forming along leading and trailing edges and with the tip chord parallel to an axis $\boldsymbol{0x}$. For wing with curvilinear outlines of edges and chords these lines must be replaced with the approximate straight lines.

Value $\boldsymbol{b_A}$ and position ($\boldsymbol{x_A}$, $\boldsymbol{z_A}$) of main aerodynamic chord may be found geometrically (Fig. 1).

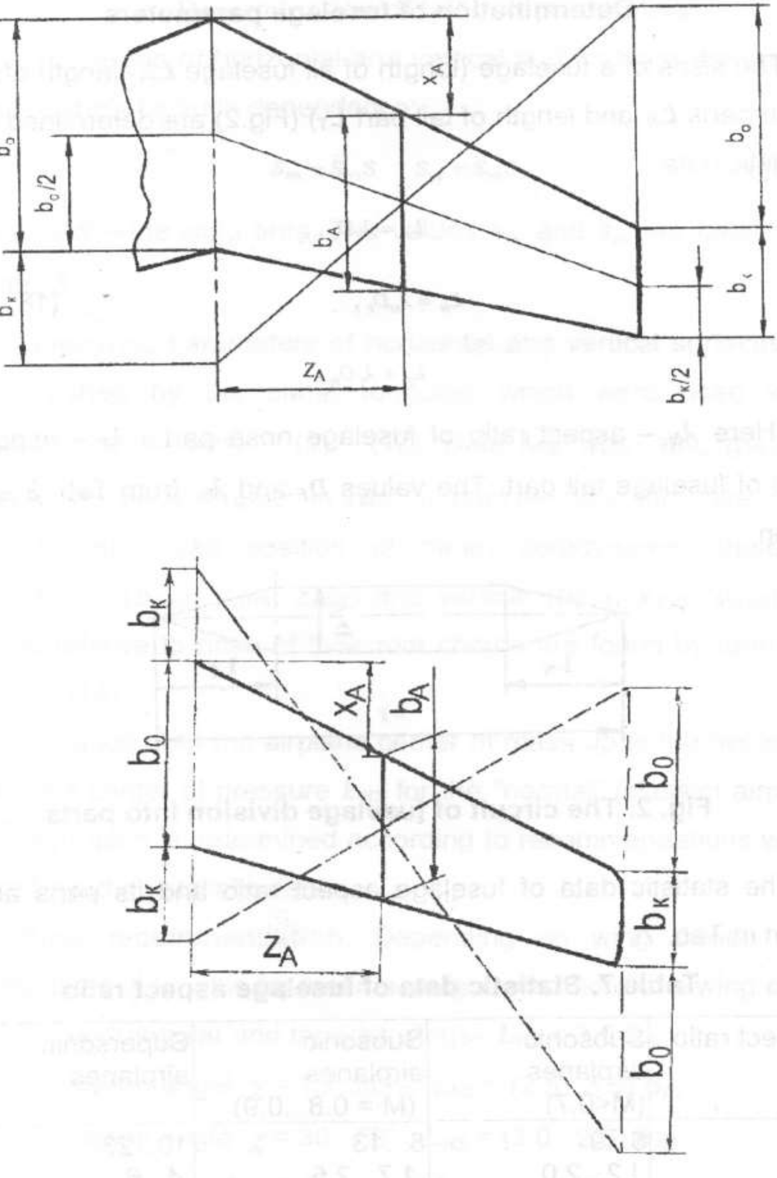

Fig.1. Geometrical ways to determine MAC

Determination of fuselage parameters

The sizes of a fuselage (length of all fuselage L_F, length of its nose parts L_N and length of tail part L_T) (Fig.2) are determined by statistic data

$$L_F = \lambda_F D_F ,$$

$$L_N = \lambda_N D_F , \qquad (18)$$

$$L_T = \lambda_T D_F .$$

Here λ_N – aspect ratio of fuselage nose part , λ_T – aspect ratio of fuselage tail part. The values D_F and λ_F from Tab. 3 are taken.

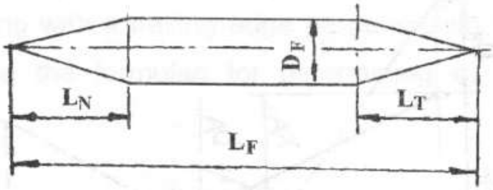

Fig. 2. The circuit of fuselage division into parts

The statistic data of fuselage aspect ratio and its parts are given in Tab. 7.

Table 7. Statistic data of fuselage aspect ratio

Aspect ratio	Subsonic airplanes (M<0.7)	Subsonic airplanes (M = 0.8...0.9)	Supersonic airplanes
λ_F	6...9	8...13	10...23
λ_N	I.2...2.0	1.7...2.5	4...6
λ_T	2...3	3...4	5...7

Determination of tail unit parameters

The areas of horizontal and vertical surfaces are determined accordingly by such dependence:

$$S_{HS} = \bar{S}_{HS} S \quad S_{VS} = \bar{S}_{VS} S, \tag{19}$$

where S – the wing area. The values $\bar{S}_{HS}$ and $\bar{S}_{VS}$ are taken from Tabl. 3 .

Geometric parameters of horizontal and vertical surfaces are determined by the same formulae which were used when calculating the wing, – (9) – (12). Data λ_{HS}, λ_{VS}, η_{HS}, η_{VS}, χ_{HS}, χ_{VS} for the tail unit (given in Tab. 3) must be put in this case.

The size and position of mean aerodynamic chord of horizontal (b_{AHS}, x_{AHS}, z_{AHS}) and vertical (b_{AVS}, x_{AVS}, y_{AVS}) tail units relative to nose of their root chords are found by formulae (12) – (14).

Distance from the airplane center of mass up to the horizontal tail unit center of pressure L_{HS} for the "normal" (classic) airplane configuration is determined according to recommendations which are based on statistic data.

First recommendation. Depending on wing sweep it is calculated in lengths for mean aerodynamic chord of a wing b_A:

- rectrangular and tapered wings– L_{HS} = 3.5 b_A;
- sweep angle χ = 10...30°– L_{HS} = (2.5...3.6) b_A;
- sweep angle χ = 30...60°– L_{HS} = (2.0...2.5) b_A;
- delta wing– L_{HS} = (1.2...1.5) b_A.

For the "canard" configuration L_{HS} = (1.2...1.5) b_A.

Distance from the vertical tail unit center of pressure up to the airplane center of mass L_{VS} in first approximation may be considered such which is equal to the distance of horizontal tail unit,– $L_{VS} = L_{HS}$.

Second recommendation. In deciding on L_{HS} statistic data given in Tab. 8 may be also taken into account.

Table 8. Distance from the airplane center of mass up to the horizontal tail unit center of pressure

Types of airplanes	L_{HS}
Passanger (with turboprop engines)	(2.0...3.0) b_A
Passanger (with turbojet and bypass engines)	(2.5...3.5) b_A
Heavy nonmaneuverable (with swept wing)	(2.5...3.5) b_A
Heavy nonmaneuverable (with tapered wing)	(2.0...3.0) b_A
Maneuverable	(1.5...2.0) b_A

Determination of position center of mass for the airplane

Position of the airplane center of mass is determined relative to nose part of wing mean aerodynamic chord.

The recommended distance for the center of mass (point *0*) from nose part of mean aerodynamic chord x_m has such values:

- for airplanes with rectrangular and tapered wings – x_m = (0.20...0.25) b_A;

- for airplanes with swept wing (χ = 35...50°) – x_m = (0.26...0.30) b_A;

- for airplanes with swept wing (χ > 50°) – x_m = (0.30...0.34) b_A;

- for airplanes with delta wing (low aspect ratio) – x_m = (0.32...0.36) b_A.

On the basis of the chosen values distance for the airplane center of mass from nose point of mean aerodynamic chord along the axis **0x**– x_m can be determined.

2.7. CONSTRUCTION OF THE AIRPLANE THEORETICAL SCHEME

After calculations of geometrical parameters for the airplane main units the theoretical scheme of plane general view may be constructed.

The theoretical scheme of the airplane is constructed in such sequence (Fig. 3):

1) to construct the scheme for airplane fuselage on the basis of its certain parameters;
2) to construct the scheme for horizontal tail unit so that the end of its root chord approximately coincided with the end of fuselage tail part (point 1);
3) to find value and site MAC of horizontal tail unit ;
4) to find position for the airplane center of mass on an axis **0x** – point **0**. For this purpose draw the arm shoulder of horizontal tail unit L_{HS} from a remote point on:

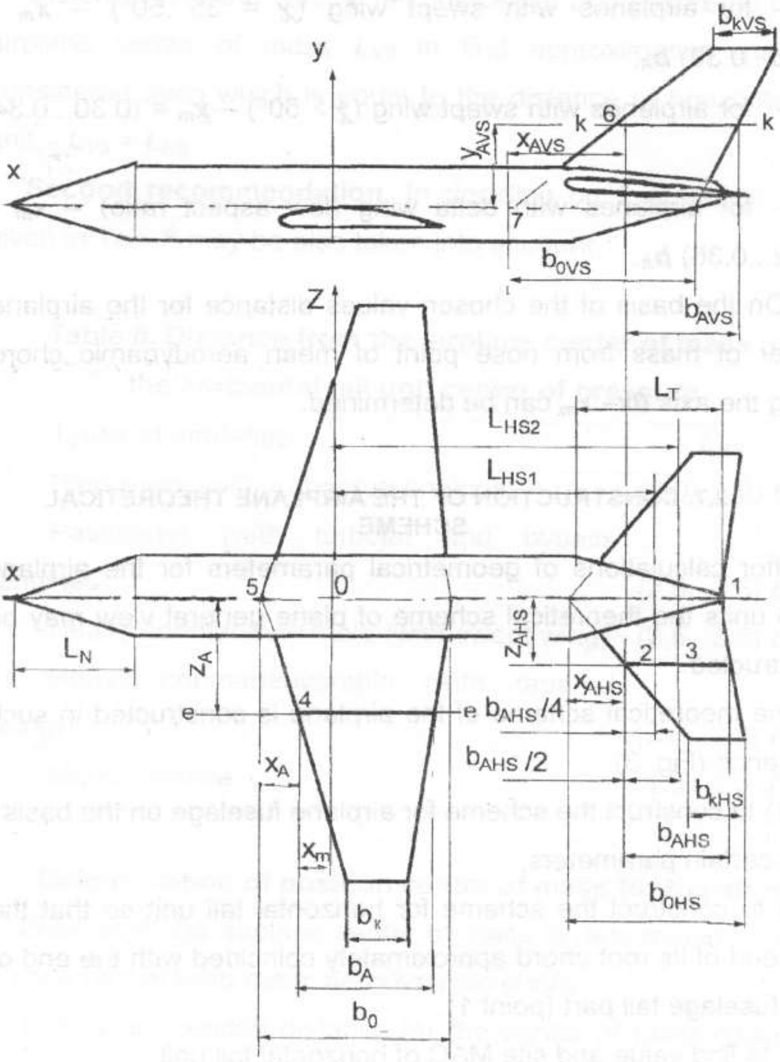

Fig. 3. Construction for two projections of a theoretical drawing of airplane general view with ordinary tail unit

- 0.25 b_{AHS} from MAC nose of horizontal tail unit for subsonic airplanes (point 2);

- 0.5 b_{AHS} from MAC nose of horizontal tail for supersonic airplanes (point 3);

5) to find position of MAC concerning an axis of a fuselage by value z_A (on this distance the line parallel to an axis of a fuselage is drawn, it is **e-e** line);
6) to find position for MAC nose of a wing along the axis Ox from the center of mass (point O). For this purpose draw the size x_m, and on a line **e-e** MAC nose point of a wing is received (point 4);
7) on **e-e** line draw (backward) value for length of wing MAC (b_A) from point 4;
8) draw value x_A (forward) from the point 4 and find position for a root chord nose of wing along this coordinate (point 5);
9) construct the wing planform accoding to known geometrical parameters;
10) find position of vertical tail unit MAC concerning an axis of a fuselage (in a projection side view). For this purpose draw the line parallel to axis of the fuselage at a distance y_{AVS} (it is ***k-k*** line);
11) relate the projection for nose point of horizontal tail unit MAC to this line and determine position for nose of vertical tail unit MAC along the axis Ox (point 6);
12) draw value b_{AVS} (backward) from point 6;

13) draw value x_{AVS} from point 6 and thus find position for nose of root chord of vertical tail unit along the axis ***Ox*** (point 7);

14) construct the form of vertical tail unit according to known parameters .

Fig. 3 illustrates construction for a theoretical drawing of airplane general view with ordinary tail unit (L_{HS1} – for subsonic airplanes, and L_{HS2} – for supersonic airplanes). In this case condition $L_{VS} = L_{HS}$ is performed.

When "T-shaped" tail unit is applied, the vertical surface must be attached first to the fuselage and then horizontal surface to vertical one (Fig. 4). Nose point of root chord HS approximately coincides with nose point of a tip chord VS (point 8, Fig. 4). Certainly, in this case condition $L_{VS} = L_{HS}$ is not performed.

The landing gear parameters are determined after construction of the airplane general view.

2.8. DETERMINATION OF THE LANDING GEAR PARAMETERS

There are following parameters for three-strut landing gear:

- landing gear wheelbase ***b***, i.e. distance (side view) between axes of main and nose struts;

- landing gear wheeltrack ***B***, i.e. distance (front view) between planes of symmetry of main wheels (or bogies);

- offset ***e***, i.e. distance (side view) between vertical line passing through the airplane center of mass (point ***0***), and the axis of the main wheel (or the centre line of the bogie);

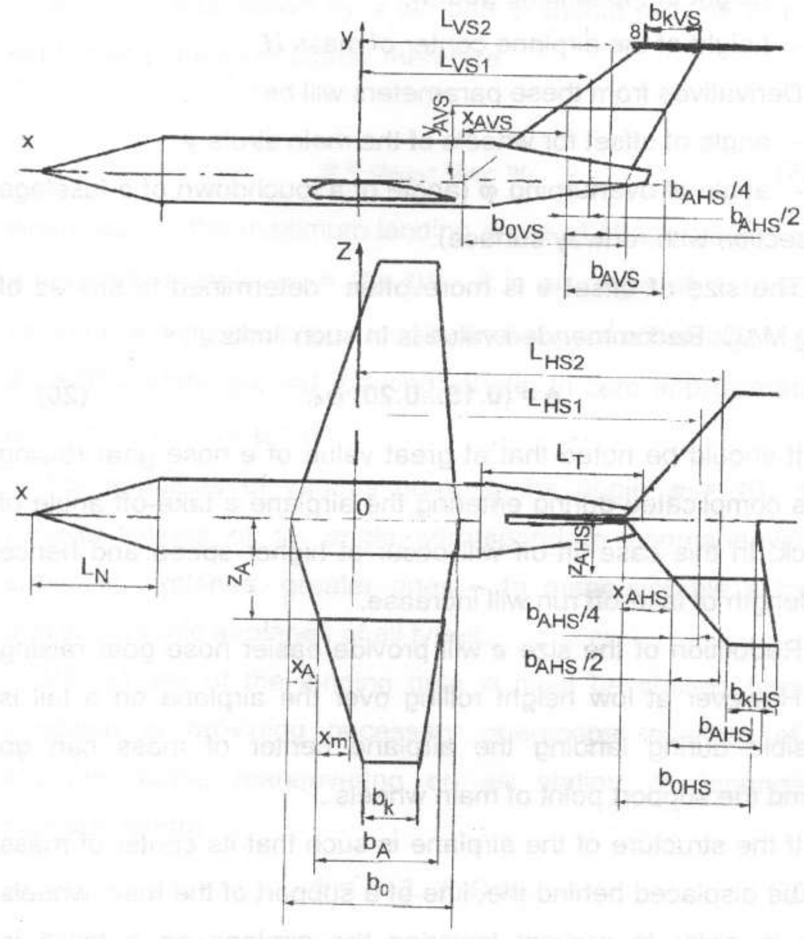

Fig. 4. Construction of two projections of theoretical airplane general view with "T-shaped" tail unit

- offset of a nose wheel **a**, i.e. distance (side view) between vertical line passing through the airplane center of mass and the axis of a nose wheel;

- height of the landing gear ***h***;
- height of the airplane center of mass ***H***.

Derivatives from these parameters will be:

- angle of offset for wheels of the main struts γ;
- angle of overturning φ (angle of a touchdown of a fuselage tail section with runway surface).

The size of offset **e** is more often determined in shares of wing MAC. Recommended value is in such limits:

$$e = (0.15...0.20)\, b_A. \qquad (20)$$

It should be noted that at great value of **e** nose gear raising off is complicated during entering the airplane a take-off angle of attack. In this case lift-off will occur at higher speed and hence the length of take-off run will increase.

Reduction of the size **e** will provide easier nose gear raising off. However at low height rolling over the airplane on a tail is possible during landing the airplane center of mass can go behind the support point of main wheels .

If the structure of the airplane is such that its center of mass can be displaced behind the line of a support of the main wheels then in order to prevent lowering the airplane on a tail it is necessary to provide the auxiliary tail support.

Angle of main wheels setoff γ should be higher than angle of a touchdown by a tail part:

$$\gamma = \varphi + (1...2)^\circ. \qquad (21)$$

Angle of touchdown by a tail part φ should provide using the set landing angles of attack, therefore

$$\varphi = \alpha_{max} - \alpha_{Wi} - \psi, \qquad (22)$$

where α_{max} – the maximum landing angle of attack depends on a wing stalling angle $\alpha_W = (0...4)°$ – it is an angle of wing setting between a wing chord and longitudinal axis of a fuselage); $\psi = (-2...+2)°$ – static ground (parking) angle. In zero approximation it is possible to take $\psi = 0$.

For airplanes of various purpose the angle φ = 10...18°. Smaller values of an angle correspond to nonmaneuverable subsonic airplanes, greater ones – to maneuverable subsonic and supersonic airplanes of all types.

Wheelbase of the landing gear ***b*** must be chosen from the condition of providing necessary operational qualities of the airplane during maneuvering on air station. It depends on fuselage length

$$b = (0.3...0.4)\, L_F. \qquad (23)$$

Distance between a nose strut and center of mass ***a*** is chosen so that during airplane parking loading on nose strut would equal (6...12)% of airplane mass. Then

$$a = (0.88...0.94)\, b, \qquad e = (0.12...0.06)\, b. \qquad (24)$$

The height of the landing gear is determined from the condition of providing the minimum gap 200...250 mm between runway surface and the airplane structure (fuselage, wing, engines, propellers, ventral fins, etc.) with separate and simultaneous compression of (pneumanic) tires and shock-absorbers. This gap must be determined for airplane landing with rolling. In this case the pitch angle of the airplane is equal to landing angle and a rolling angle is equal to 4°.

For the airplanes having high mass the track must be made wider than runway plates width, the size of which is equal to 7 m. It results in decreasing loading on plates, because in this case there will be only one support of the airplane on each of them.

The maximum size of a track ***B*** should be limited up to 12 m to provide an opportunity of safe taxiing of airplanes on taxiways of aerodromes with width being 15 m. The minimum track for landing gear is chosen also from condition of exeption of airplane overturning while moving on runway. It depends mainly on height of the airplane center of mass ***H***.

On this basis the track of the landing gear should be in such limits:

$$2H < B < 15\ m. \qquad (25)$$

The height of airplane center of mass is determined by parameters γ and ***e*** (it is necessary to aspire to possible smaller value ***H***), the structure of the airplane (low-wing monoplane, high-wing monoplane).

2.9. CONSTRUCTION OF THE AIRPLANE GENERAL VIEW

After definition of airplane parameters the drawing of airplane general view in three projections is carried out on the basis of the theoretical drawing (Fig. 5).

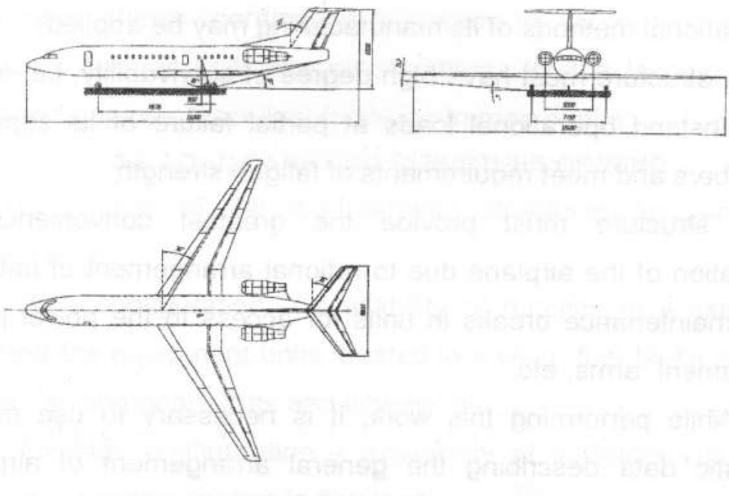

Fig. 5. Airplane general view

The obtained result is compared with general view of airplanes-prototypes. If necessary, corresponding correctives are amended.

3. DETERMINATION OF LOAD-CARRYING STRUCTURE FOR AIRPLANE UNITS

3.1. DETERMINATION OF THE GENERAL ARRANGEMENT FOR AIRPLANE UNITS

In solving the general arrangement for airplane units such fundamentals must be taken into account :

- the mass of an airplane airframe structure for the set conditions should be the least, which is achieved by rational transfer of load to structural members at necessary rigidity;

- structure must be adaptable, i.e. such that the simplest and rational methods of its manufacturing may be applied;

- structure must have high degree of survivability, i.e. ability to withstand operational loads at partial failure of its separate members and meet requirements of fatigue strength;

- structure must provide the greatest convenience in operation of the airplane due to rational arrangement of hatches and maintenance breaks in units for access to the power-plant, equipment, arms, etc.

While performing this work, it is necessary to use mainly statistic data describing the general arrangement of airplane units.

Examples of the most widespread load-carrying structures of the main units of airplanes and their general arrangement were given in [1].

The diagram of mutual coordination of the main primary members is defined after the choice of general arrangement for units (wing, fuselage, horizontal and vertical tail units, landing gear).

On the drawing of the general arrangement of unit it is necessary to show the arrangement of spars, stringers, normal and reinforced ribs, for a fuselage – the arrangement of longerons, stringers, normal and reinforced frames.

Distances between members of longitudinal and transverse structural members of the wing, tail unit, fuselage should be specified.

Circuit of coordination for the primary structural members of the airplane shows coordination of position for spars, false spars of a wing and tail unit, attachment fittings for the landing gear struts and engines (nacelles) to strong frames or ribs.

3.2. LOAD-CARRYING STRUCTURE OF WING

The choice of the load-carrying structures for wing is determined by:

- wing configuration – availability of hatches in a skin for servicing the equipment units located in a wing, fuel tanks inside a wing, landing gear struts and wheels, etc.;
- fuselage configuration – availability of sufficient volumes for the wing centre section in fuselage;
- rigidity requirements.

Determination of load-carrying structure of a wing

For the approximate choice of the load-carrying structure of wing two criteria may be used.

First, this is a concept ***of a conventional spar.*** The cap width of the specified spar is 0.6 of wing chord in reference section. Root chord b_0 is accepted as such section. In this case cap thickness of a conventional spar is determined by the formula

$$\delta_y = \frac{(p_0 S z_A - 2m_0 g z_i - m_W g z_A)n^p}{0{,}96\bar{C} b_0^2 \sigma_p}, \qquad (26)$$

where p_0 – wing loading at take-off, H/m^2; S – the wing area, m^2; z_A – coordinate of mean aerodynamic chord of a wing spanwise, m; m – mass of cargo located on a wing, kg; g – acceleration of free fall, 9.8 m/c^2; z_i – coordinate of the center of mass for the cargo located on a wing from a longitudinal axis of the airplane, m; n^P – g-load (values n^P are given in Tab. 9); m_W – the wing mass, kg; $\bar{c}$ – airfoil thickness ratio; b_0 – the root wing chord, m; σ_p – a breaking stress of a material for a spar cap.

It is possible to accept values of breaking stresses for: aluminium alloy σ_p=330 MPa, alloyed structural steel σ_p= 880 MPa, titanium alloy – σ_p = 800 MPa.

For other structural materials σ_p my be determined approximately by relation $\sigma_p = 0{,}8\sigma_b$, where σ_b – ultimate strength of a material while stretching.

Table 9. Data of g-load

Types of airplanes	n^P
Maneuverable	12…13.5
Midmaneuverable	6…9
Nonmaneuverable	3…4

If as a result of calculation for cap thickness value of a conventional spar δ_{n} is less than 3 mm, then the wing skin will be thin. In this case its critical stresses of instability will be low (in a compression zone). A material (as designers speak) is lacking to form a skin with stringers which reinforce a skin and increase its critical stresess. In such a case spar-type wing will be more rational. In such a wing the bending moment basically is taken up by spar caps (ref. [1]).

If δ_{m} is more than 3 mm, then the wing skin may be made rather thick and with high critical stresess of instability. Monoblock-type (stressed-skin-type) or torsion-box-type wing in mass criterion are useful to be applyied in this case.

Second, the load-carrying structure of a wing may be chosen by criterion ***of load moment intensity***.

Load moment intensity for root wing section is determined by the formula

$$\frac{M}{H^3} = \frac{\left[\left(p_0 S - m_{f,p} g\right) Z_A - 2 m_i g Z_i\right] n^P}{1{,}03\left(\bar{C} b_0\right)^3}, \qquad (27)$$

where M – bending moment, H·m; H – design thickness of airfoil section, m; H = 0.8 H_{max}, H_{max} – maximum thickness of wing airfoil section in its root wing portion, m.

If the size of load moment intensity does not exceed 10…15 MPa, in this case spar-type wing is more favourable in the mass criterion.

If relation M/H^3 is more 10…15 MPa, monoblock-type (stressed-skin-type) or torsion-box-type wing will have advantage.

The above does not mean, that the load-carrying structure must be accepted, utility of which is determined by specified calculations. There are still requirements of aerodynamics, reliability, manufacturing (technology), operation, etc.

Only considering a wide spectrum of requirements to a wing gives possibility to make solution of its load-carrying structure.

Determination of distance between ribs and stringers

Distance **a** between ribs is chosen depending on skin thickness and the sizes of stringer sections. Very short distance is disadvantageous owing to a plenty of rivets.

It results in quality deterioration of a wing surface, complication of its manufacturing, appearance of stress increase in large quantity.

In stringerless wings depending on skin thickness rib pitch ***a*** may be accepted equal 120 ... 220 mm with skin thickness δ_{SK} = 1.2 ... 2.0 mm.

In wings with stringers set the distance between ribs is chosen depending on stringers cross section and skin thickness – (***a*** =200 ... 300 mm).

In modern torsion-box-type wings with high specific loading (p_0 >4000 H/m^2) with skin thickness δ_{SK} = 1.5 ... 2.0 mm the distance between ribs is taken equal 250 ... 300 mm. With thicker skin (δ_{SK} =3 ... 5 mm) value ***a*** can reach 700 ... 900 mm.

The distance between stringers in spar-type wings (one or two spars) $\boldsymbol{b}_{STR}$ = 200 ... 300 mm.

In torsion-box-type wings the distance between stringers must be taken short (b_{STR} = 100 ... 180 mm), because in this case the panel carries compressive stresses better .

Example of load-carrying structure of wing is shown in Fig. 6.

3.3. DETERMINATION OF LOAD-CARRYING STRUCTURE FOR FUSELAGE

Modern airplanes in the majority have semimonocoque-type fuselage. It consists of skin, stringers and frames. Fuselages of longeron-type and monocoque-type structures can be applied too. Truss-type fuselages are used for light airplanes.

The distance between frames depends on thickness of a fuselage skin, configuration and mass of the airplane. On real structures frame pitch is accepted in such limits:

- 200... ... 300 mm – for light airplanes;
- 300... ... 400 mm – for medium airplanes ($15 < m_0 < 30$ t);
- 450... ... 500 mm – for heavy airplanes.

The distance between stringers in a fuselage is chosen for the same reasons as in a wing. Depending on thickness of a skin distance between stringers is accepted the following:

- 100...150 mm – for light airplanes;
- 150...200 mm – for heavy airplanes.

In a big cut-out zone longitudinal reinforced and strong members (strong stringers, short longerons) are mounted as fringes

Such requirements and recommendations must be taken into account while designing a fuselage structure:

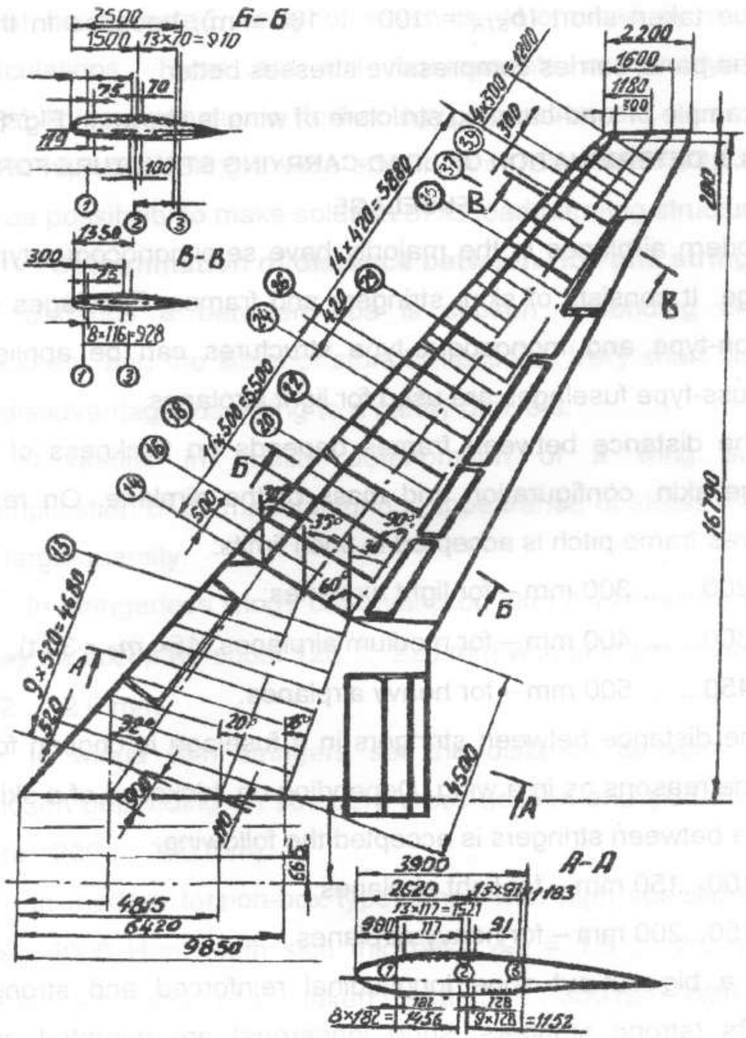

Fig. 6. Load-carrying structure of a wing

- concentrated forces applied to the fuselage main members (longerons, frames, short longerons, stringers) must be smoothly distributed to the fuselage skin;

- great concentrated forces (from engines, tail unit, wing, landing gear) must be transmitted to the skin by main members which are directed in parallel to force. The forces directed along the fuselage must be transmitted to a skin through stringers and longitudinal beams. The forces acting crosswise a fuselage must be transmitted to skin through strong frames;

- concentrated forces directed under a sharp angle to fuselage axis must be transmitted to the skin through stringers and frames;

- while designing pressurized fuselage compartments it is necessary to spesify correct boundary of the pressurization zone (with taking into account the cut-outs for the landing gear, wing, etc.). In this case applying flat surfaces for taking up internal pressure differential should be avoided.

The example of the load-carrying structure of a fuselage is shown in Fig. 7.

3.4. DETERMINATION OF LOAD-CARRYING STRUCTURES FOR THE TAIL UNIT

Structural members of horizontal and vertical surfaces must be coordinated with each other and structural members of the fuselage.

In the design of tail unit the two-spar structure is usually applied, for heavy airplanes – sometimes the torsion-box-type or multispar-type structures are used.

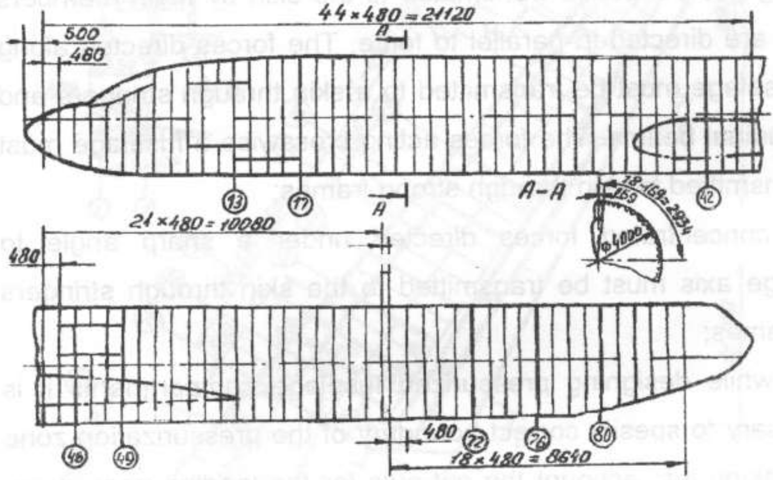

Fig. 7. The load-carrying structure of fuselage

Control surfaces are most often designed according to monospar structure with ribs located perpendicularly to a spar. Very often the tail section of the control surfaces is a structure with honeycomb filler (without using normal ribs).

The lightest structure of tail unit can be in that case if the primary structural members of the tail unit carrying bending stresses (spars and torsion-box) my be passed through the fuselage.

To reduce structure mass whenever possible vertical unit and stabilizer should be fixed to the same strong frames of the fuselage.

If the stabilizer with continuous longitudinal members is placed in the upper portion of a fuselage section,stabilizer center portion (caisson) passes between strong frames of the fuselage.

The stabilizer rear spar is attached to rear frame of these frames, the stabilizer front spar is attached to forward frame.

If the stabilizer is designed adjustable, then rear units, as a rule, are made stationary ones, and forward units of attachment are replaced with a bracket. The mechanism of stabilizer drive is attached to this bracket. Hinge unit of stabilizer turning, as a rule, is set on the rear spar of the vertical unit.

The mechanism of a stabilizer drive is attached to the front spar of vertical unit with strong rib.

Arrangement of horizontal tail unit on vertical one (the T-shaped structure) results in increase of load on vertical surface. For taking up these additional loads the structure of vertical tail unit should be more stronger. It results in increase of VS mass .

While choosing the position of VS spars for T-shaped tail unit the optimum distance between them must be sought. It is usually achieved when arranging a front spar in limits (0.15 ... 0.2)$\boldsymbol{b}_{VS}$, and rear one – (0.60 ... 0.65)$\boldsymbol{b}_{VS}$.

Examples for load-carrying structures of horizontal and vertical units are shown in Fig. 8, 9.

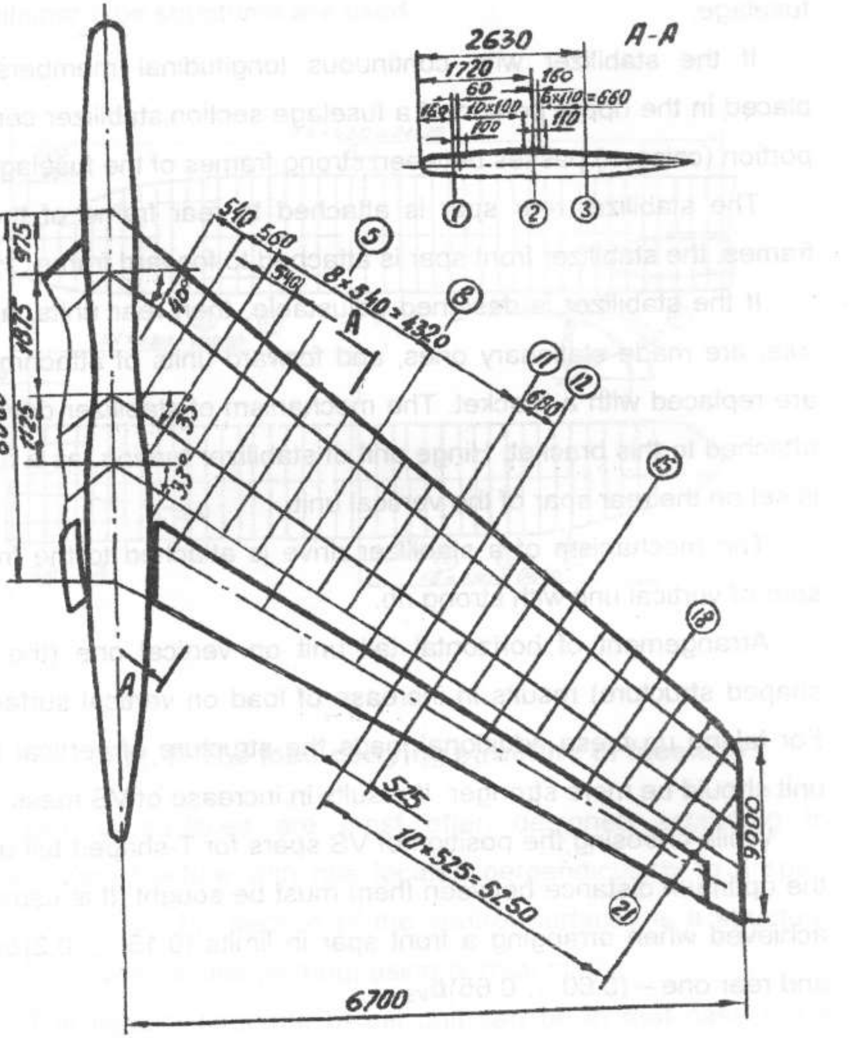

Fig. 8. The load-carrying structures of horizontal tail unit

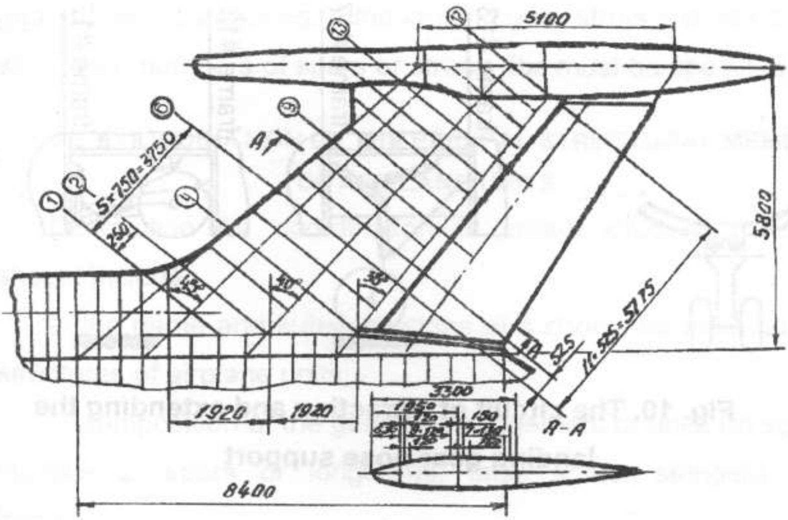

Fig. 9. The load-carrying structures of vertical tail unit

3.5. DETERMINATION OF LOAD-CARRYING STRUCTURE OF THE LANDING GEAR

The load-carrying structures of the landing gear and the devices of its retracting should provide:

- the least mass of the landing gear (in view of reinforcing the cut-outs for landing gear in the structures of a fuselage or wing);
- the least volume of the landing gear in retracted position;
- simplicity of the kinematic structure for mechanisms of extension and retraction.

Besides, wheels and struts while retracting the landing gear should not pass through the space assigned to external devices if they are allocated (Fig. 10, 11).

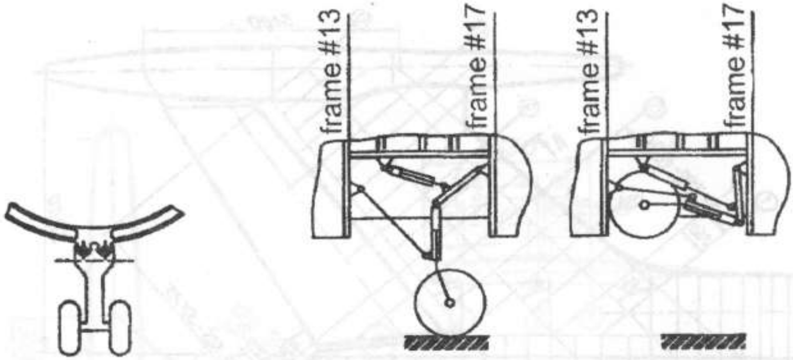

Fig. 10. The circuit of retracting and extending the landing gear nose support

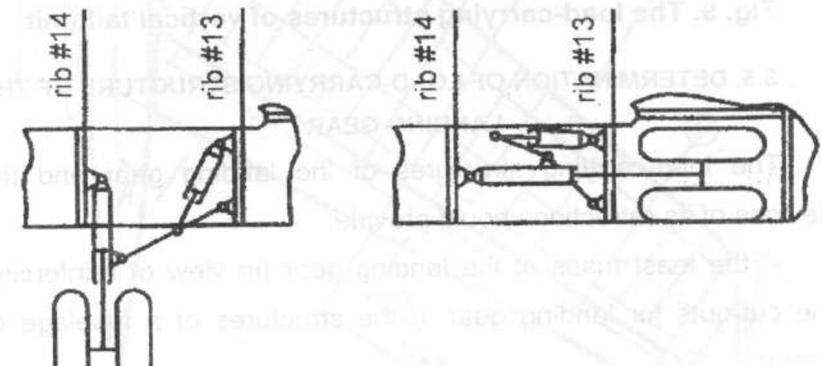

Fig. 11. The circuit of retracting and extending the main landing gear

On circuits of retracting and extending the numbers of strong frames must be specified to which major members of landing gear struts are attached. If the landing gear struts are attached to wing, then numbers of spars or strong ribs must be specified.

3.6. COORDINATION OF THE PRICIPAL STRUCTURAL MEMBERS OF AIRPLANE UNITS

Description for coordination of pricipal structural members should include:

- the name and substantiations of a choice for load-carrying structures of airplane units;

- composition of the general arrangement of units (to specify number of spars or longerons, false spars, stringers, ribs, frames);

- purpose of load-carrying members – strong and reinforced ribs and frames, spars, longerons, false spars with concrete indication of its numbers on the drawing (for example, strong frame №3 is intended for attaching the landing gear nose strut).

After determination of airplane load-carrying structures of airplane units, they must be united in an integral part.

Wing spars must be attached to fuselage strong frames. Therefore the distance between wing spars in its root section must be equal to distance between the strong frames (or to the contrary).

The same requirement concerns the distances between stabilizer (or vertical unit) spars and strong frames in a fuselage tail section. If the stabilizer is attached to the vertical surface (for example, T-shaped tail unit), then the distance between stabilizer spars in its root section must be equal to the distance between vertical surface spars in its tip section.

The main rule is: load-carrying members of one unit must be attached to load-carrying members of another one (Fig. 12).

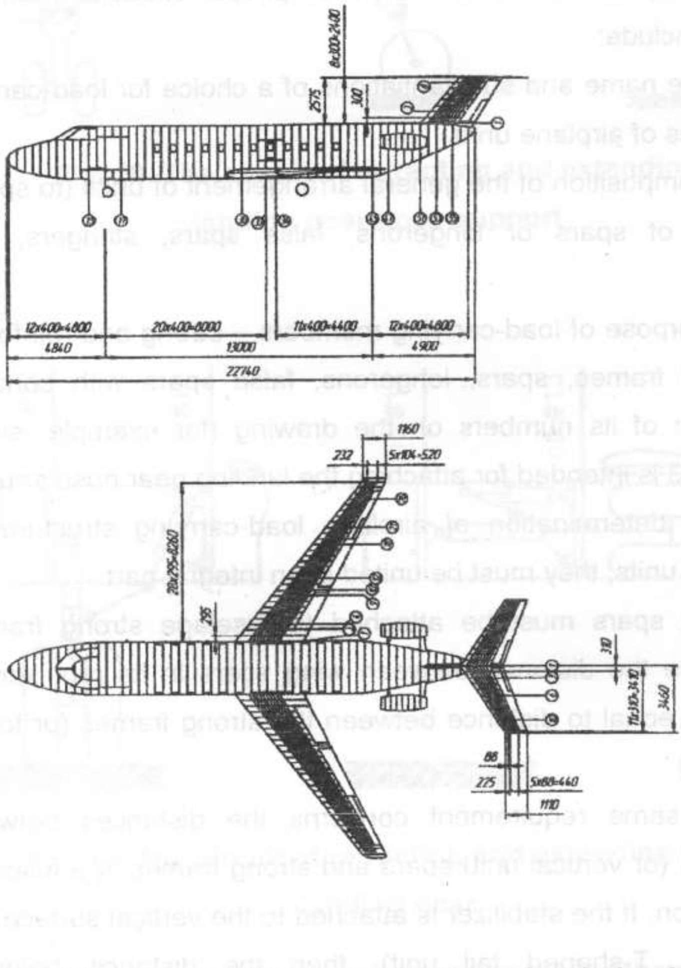

Fig. 12. The circuit for coordination of main load-carrying members

Thus, the airplane general view and load-carrying structures of its main units in zero approximation is obtained, and then the analysis of the found results must be received.

If they meet established TTR, there are the first, second and next approximations - until airplane will be received, which corresponds either of requirements of the customer, or requirements of the market.

List literature

1. Fedotov M.N. General arrangement of airplanes. – The summary of lectures. – Kharkov: N.Ye. Zhukovsky National aerospace university "KhAI", 2006. – 245 p.
2. Глаголев А.Н., Гольдинов М.Я., Григоренко С.М. Конструкция самолетов. – М.: Машиностроение, 1975. – 480 с.
3. Егер С.М., Матвеенко А.М., Шаталов И.А. Основы авиационной техники. – М.: МАИ, 1999. – 576 с.
4. Житомирский Г.И. Конструкция самолетов. – М.: Машиностроение, 1995. – 416 с.
5. Клименко В.Н., Кобылянский А.А., Малашенко Л.А. Приближенное определение основных параметров самолета: Учеб. пособие. – Х.: ХАИ, 1986. – 40 с.
6. Кривцов В.С., Карпов Я.С., Федотов М.М. Інженерні основи функціонування і загальна будова аерокосмічної техніки. – Х.: Нац. аерокосм. ун-т "Харк. авіац. ін-т", 2002.– Ч. 1. – 468 с; Ч. 2. – 723 с.
7. Кривцов В.С., Карпов Я.С., Федотов М.Н. Основы аэрокосмической техники. – Х.: Нац. аэрокосм. ун-т "Харьк. авиац. ин-т", 2003. – Ч. 1. – 620 с; Ч. 2. – 901 с.
8. Методические указания к лабораторным работам по курсу "Общее устройство самолетов и агрегатов"/ Сост. Л.А. Малашенко. – Х.: ХАИ, 1986. – 77 с.

9. Никитин Г.А., Баканов Э.А. Основы авиации: Учебник для вузов гражданской авиации. – 2-е изд., перераб. и доп. – М.: Транспорт, 1984. – 261 с.

10. Основы общего проектирования самолетов с газотурбинными двигателями: Учеб. пособие. / П.В. Балабуев, С.А. Бычков, А.Г. Гребеников и др. – Х.: Нац. аэрокосм. ун-т "Харьк. авиац. ин-т", 2003.– Ч.1. – 454 с; Ч. 2. – 390 с.

11. Проектирование самолетов/ С.М. Егер, В.Ф. Мишин, Н.К. Лисейцев и др.– М.: Машиностроение, 1983. – 616 с.

12. Торенбик Э. Проектирование дозвуковых самолетов. – М.: Машиностроение, 1983. – 648 с.

13. Шульженко М.Н. Конструкция самолетов. – М.: Машиностроение, 1971. – 412 с.

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