МІНІСТЕРСТВО ОСВІТИ І НАУКИ УКРАЇНИ

НАЦІОНАЛЬНИЙ АВІАЦІЙНИЙ УНІВЕРСИТЕТ

Кафедра конструкції літальних апаратів

«ДОПУСТИТИ ДО ЗАХИСТУ»
Завідувач кафедри
д-р техн. наук., проф.
_______________________С.Р.Ігнатович
(підпис)
« _____»____________________ 2022 р.

ДИПЛОМНИЙ ПРОЄКТ

(ПОЯСНЮВАЛЬНА ЗАПИСКА)

ВИПУСКНИКА ОСВІТНЬО-КВАЛІФІКАЦІЙНОГО РІВНЯ

«БАКАЛАВР»

Тема: «Аванпроєкт середньо магістрального пасажирського літака місткістю 156 пасажирів»

Виконала: Ян Шишици ЯН Шишици

Керівник: д-р техн. наук., проф. ___________М.В. КАРУСКЕВИЧ

Нормоконтролер: канд. техн. наук., доц.___________С.В. ХИЖНЯК

Kyiv 2022

MINISRY OF EDUCATION AND SCIENCE OF UKRAINE

NATIONAL AVIATION UNIVERSITY

Aircraft Design Department

AGREED
Professor, Dr. of Sc.
_________S.R. Ignatovych
«___» _______ 2022

DIPLOMA WORK

(EXPLANATORY NOTE)
OF EDUCATIONAL DEGREE

«BACHELOR»

Theme: «Preliminary design of the mid-range passenger aircraft with 156 passengers capacity»

Performed by: Yang Shishiqi Yang Shishiqi

Supervisor: Dr. of Science, Professor __________ M. V. Karuskevich

Standard controller: PhD, associate professor __________ S.V. Khizhnyak

Kyiv 2022

NATIONAL AVIATION UNIVERSITY

Aerospace Faculty

Aircraft Design Department

Educational degree «Bachelor»

Speciality 134 "Aviation and Space Rocket Technology"

APPROVED
Professor, Dr. of Sc.
_________S.R. Ignatovych
«____» ___________ 2022

TASK
for bachelor diploma work

YANG SHISHIQI

1. Theme: «**Preliminary design of the mid-range passenger plane with 156 passenger capacity**»

   Confirmed by Rector's order №.489/СТ from 10.05.2022.
2. Period of work execution: 23.05.2022 to 19.06.2022
3. Work initial data:
   - Maximum payload – n = 156 passengers;
   - Flight range with maximum payload – $L_{пол}$= 5700 km;
   - Cruise speed – V_{cr} = 850 km/h at operating altitude H_{op}= 12100km;
   - Landing speed – V_{land} = 135km/h.
4. Explanation notes (list of topics to be developed):
   - selection of design parameters;
   - choice and substantiations of the airplane scheme;
   - calculation of aircraft masses;
   - determination of basic geometrical parameters;
   - aircraft layout;
   - center of gravity position calculation;
   - determination of basic flight performance;
   - description of the aircraft design;

– engine selection;

– special part;

5. List of the graphical materials:

– general view of the airplane (A1×1);

– layout of the airplane (A1×1);

– Crashworthiness improvement algorithm (A1×2).

6. Calendar Plan

№ п/п	Task	Execution period	Signature
1	Task receiving, processing of statistical data	23.05.2022-28.05.2022	Yang Shishiqi
2	Aircraft take-off mass determination	28.05.2022-31.05.2022	Yang Shishiqi
3	Aircraft layout	31.05.2022-03.05.2022	Yang Shishiqi
4	Aircraft centering determination	03.06.2022-05.06.2022	Yang Shishiqi
5	Graphical design of the parts	05.06.2022-12.06.2022	Yang Shishiqi
6	Completion of the explanation note	12.06.2022-14.06.2022	Yang Shishiqi
7	Defense of diploma work	14.06.2022-19.06.2022	Yang Shishiqi

7. Task date: 23.05.2022

Supervisor of diploma work: ________________ M.V. Karuskevich
(signature)

Student: Yang Shishiqi ________________ Yang Shishiqi
(signature)

НАЦІОНАЛЬНИЙ АВІАЦІЙНИЙ УНІВЕРСИТЕТ

Аерокосмічний факультет

Кафедра конструкції літальних апаратів

Освітній ступінь «Бакалавр»

Спеціальність 134 «Авіаційна та ракетно-космічна техніка»

Освітньо-професійна програма «Обладнання повітряних суден»

ЗАТВЕРДЖУЮ
Завідувач кафедри, д.т.н, проф.
__________ Сергій ІГНАТОВИЧ
«_____» _____________ 2021 р.

ЗАВДАННЯ
на виконання дипломної роботи студента

ЛЮ ЦЗЯНЬВЕЙ

1. Тема роботи: «Аванпроєкт середньо магістрального пасажирського літака місткістю 156 пасажирів», затверджена наказом ректора № 489/ст від 10 травня 2022 року.

2. Термін виконання роботи: з 23 травня 2022 р. по 19 червня 2022 р.

3. Вихідні дані роботи:

Максимальне корисне навантаження – n = 156 пасажирів;

Дальність польоту з максимальним корисним навантаженням

– Lпол= 5700 км;

Крейсерська швидкість – Vcr = 850 км/год на робочій висоті Hor= 12100 км;

Посадочна швидкість – Vland = 135 км/год.

4. Пояснення (перелік тем, які будуть розроблені):

вибір параметрів конструкції;

вибір та обґрунтування схеми літака;

розрахунок маси літака;

визначення основних геометричних параметрів;

компонування літака;

розрахунок положення центру ваги;

визначення базових льотних характеристик;

опис конструкції літака;

вибір двигуна;

особлива частина;

5. Перелік графічних матеріалів:

загальний вигляд літака (А1×1);

макет літака (А1×1);

Алгоритм підвищення стійкості до ударів (А1×2).

6. Календарний план-графік:

Завдання	Термін виконання	Відмітка про виконання
Вибір вихідних даних, аналіз льотно-технічних характеристик літаків-прототипів	23.05.2022–28.05.2022	Ян Шишици
Вибір та розрахунок параметрів проєктованого літака	28.05.2022–31.05.2022	Ян Шишици
Виконання компонування літака	31.05.2022–03.06.2022	Ян Шишици
Розрахунок центрування літака	03.06.2022–05.06.2022	Ян Шишици
Виконання креслень літака	05.06.2022–12.06.2022	Ян Шишици
Оформлення пояснювальної записки та графічної частини роботи	12.06.2022–14.06.2022	Ян Шишици
Захист дипломної роботи	14.06.2022–19.06.2022	Ян Шишици

7. Дата видачі завдання: 23.05.2022 рік

Керівник дипломної роботи __________ М.В. КАРУСКЕВИЧ

Завдання прийняв до виконання Ян Шишици ЯН Шишици

ABSTRACT

Explanatory note to the diploma work «Preliminary design of the mid-range passenger aircraft with 156 passenger capacity»

50 pages, 10 figures, 9 tables, 9 references and 2 drawings

Object of the work is a process of the mid-range aircraft design with 156 passengers capacity.

Aim of the diploma work is the preliminary design of the aircraft and its primary design characteristics.

The methods of the design are comparative analysis of the similar aircraft, geometry and masses optimization, mathematical simulation of the crashworthiness.

The special part describes the effect of luggage on the crashworthiness of fuselage structure.

The diploma work contains preliminary design of the mid-range aircraft, calculations and drawings, recommendations on crashworthiness improvements.

AIRCRAFT,PRELIMINIARY DESIGN,GEOMETRY CHARACTERISTICS, MASS CHARACTERISITICS,CRASHWORTHINESS

CONTENT

LIST OF ABBREVIATIONS

RPK	Revenue passenger-kilometres
LG	Landing gear
APU	Auxiliary power unit
LP	Low pressure
HP	High pressure
IATA	International aviation transport association
ICAO	International civil aviation organization
FAR	Federal aviation regulation
CS	Certification specification
CCAR	Chinese civil aviation regulation
CW	Crashworthiness
CM	The center of the mass

List of drawings

Introduction

The Diploma work is carried out in the compliance with the program for Specialty: 134 Aviation and Rocket-Space Engineering, Field of study: 13 Mechanical Engineering.

The diploma work considers two problems: 1 – preliminary design of the mid-range passenger aircraft, and 2 – concept of the improvement of the designed aircraft crashworthiness by the use of luggage for absorption and dissipation of the crash strain energy.

The aim of the preliminary design was to find geometry, layouts, masses of components, primary characteristics of the mid-range passenger aircraft with 156 passengers capacity. The results confirms compliance to International and Chinese airworthiness requirements. Calculations have been carried out according to the guide[6] for the Course projects and Diploma works developed by the Aircraft Design Department of National Aviation University[7]. The notes of the lectures, delivered by the department teachers were used along the work on diploma paper as well[8].

The main characteristics of the proposed plane are:

- Lower fuel consumption ;
- Longer range ;
- Greater carrying capacity ;
- More advanced operating system ;
- Superior flight systems .

The aim of the second part of the work was to find theoretical decision for the problem of the aircraft servivebility enhancement.

The practical value of the work. The analysis of the works performed in this field has shown that the crashworthiness behaviour of a narrow-body transport category aircraft structure depends on the possibility of the luggage compartment to absorb energy of the fuselage deformation under the out-of-landing gear landing.

Investigation of the aircraft crashworthiness with with luggage and without luggage, conducted by Zhu Xianfei, Feng Yunwen, Xue Xiaofeng and Qin Qiang[10] and presented in their work “Evaluate the crashworthiness response of an aircraft fuselage section with luggage contained in the cargo hold” has been adapted for the designed aircraft.

Further research and development activity are recommended to be focused on the optimization of the luggage storage for crashworthiness improvement.

PART 1. PRELIMINARY DESIGN OF THE PLANE

1.1. Analysis of similar planes' parameters and choice of preliminary parameters

The selection of aircraft optimal design parameters is a complex iterative process which is intended to provide the take-off weight and cost of the aircraft as low as possible. The choice of aircraft configuration is related to the entire complex flight technology, weight, geometric parameters, aerodynamics and economic characteristics. The first stage is to determine the "aircraft shape", using approximate aerodynamics, and mathematical and statistical correlations[5]. The second stage adopts full aerodynamic calculations, and determines the aggregate weight based on the aircraft's initial data, physical theory, related mathematical formulas and aerodynamic principles.

The contemporary aircraft A320-200 is used to design the new aircraft and to provide relevant data as a reference[9]. It can accommodate 150-170 passengers. Aircraft like Airbus 320-200, Airbus 319-100 and Airbus 321-100 is able to compete with newly designed aircraft in this field segment. The statistical data of the closest prototypes is shown in Table 1.1.

Table 1.1 – Operational-technical data of prototypes

PARAMETER	PLANES		
	A319-100	A320-200	A321-100
The purpose of airplane	Passenger	Passenger	Passenger
Crew/flight attend. Persons	2/2	2/2	2/2
Maximum take-off weight, m_{tow}, kg	75500	78000	93500
Most pay-load, $m_{\text{к.max}}$, kg	17700	19900	25300
Passenger's seat	124	150	185
Flight range with $G_{payload.\ max.}$, km	6800	5700	5600
Range of cruising altitudes, km	6945	6112	5926
Take off distance , m	1950	2100	2180
Number and type of engines	2 IAE V2500-A5	2 IAE V2500-A5	2 IAE V2500-A5
Fuel capacity,L	24210~30190	24210~27200	24050~30030

Ending of table 1.1 – Operational-technical data of prototypes

Fuselage length, m	33.84	37.57	44.51
Fuselage width and cabin width,m	3.95/3.7	3.95/3.7	3.95/3.7
Sweepback on 1/4 chord, ^{0}	25	35	25
Unit load devices	4×LD3-45	7×LD3-45	10×LD3-45

The layout is determined by the relative position, number and shape of each aircraft module. The aerodynamic and operational characteristics of the aircraft depend on the layout of the aircraft and the aerodynamic design of the aircraft[2]. Fortunately, the selected procedure improves the safety and regularity of the flight, as well as the economic efficiency of the aircraft.

1.1.1. Wing

The wing adopts a cantilevered lower wing aerodynamic scheme. It has a supercritical rear-loaded airfoil similar to that of the prototype, which can be well adapted to the characteristics of short and average flight range. Engines are installed under the wings on both sides. The wing characteristic changes little with the cruising speed and altitude, and the drag is lower in the entire range of use. The thickness of the rear spar of the wing is increased, and there is enough space to accommodate the flaps and its control system. The "wingtip sail" installed at the tip of the wing has a circular front edge and a wedge-shaped trailing edge. There is a "spindle" extending backward along its chord plane to suppress the wingtip vortex. This device has a better drag reduction effect than the winglet. When encountering a crosswind, the "wingtip sail" itself will not stall. The full-span leading edge slat is divided into 5 sections, and the engine nacelle is located close to the lower surface of the wing, without compromising high-speed cruise performance. The engine pylon was partially modified to improve the spanwise continuity of the open position of the leading edge slat. Both the inner and outer trailing edge flaps use large single-slotted Fowler flaps. There are 5 spoilers on the left and right wing surfaces. When used as a lift reducing board, 5 pieces are used at the same time, the inner aileron is eliminated, and the spanwise continuity of the trailing edge flap during take-off and landing is improved, thereby enhancing the lift effect. Each wing surface component is made of composite materials.

The wing structure consists of ribs, spars, stringers and skin. The distance between the lower wing surface and the ground is relatively small. When the landing gear is retracted, the wings on both sides absorbs energy to reduce the impact on the cabin, and can provide additional buoyancy when landing on water.

1.1.2. Tail Unit

The tail fin is swept back, and the flat tail reverse camber airfoil is composed of vertical and horizontal tail fins. This conventional tail wing can provide sufficient stability and maneuverability under the lightest weight. The vertical tail includes a fixed vertical stabilizer and a movable rudder, and the horizontal tail includes a stabilizer and an elevator. The vertical tail is mounted on the fuselage. The sweep angle of the vertical and horizontal tails is greater than the sweep angle of the wing, so as the speed increases, the aerodynamic characteristics of the tail unit are more stable than those of the wing.

The vertical and horizontal tail profile are symmetrical(fig1.1.1) The symmetrical profile allows the same aerodynamic load characteristics to be maintained during the deflection of the rudder in different directions, while having less resistance.

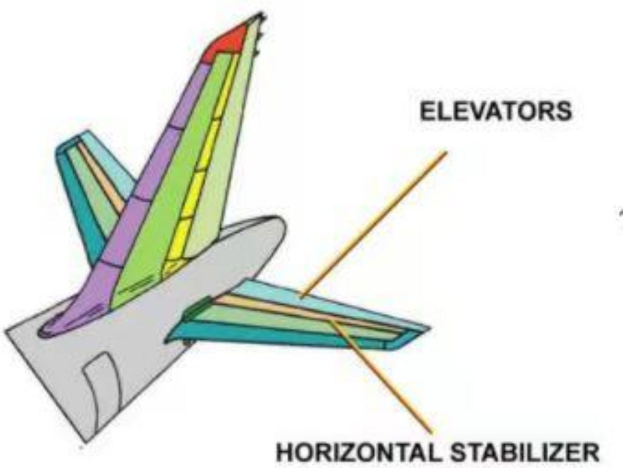

Figure 1.1.1 – Tail structure distribution

1.1.3. Power Plant

Following the tendency in the the engines application for the planes of transport category the dual rotor, axial flow, high channel ratio turbofan engines suspended under the wings on both sides.

1.1.4. Landing Gear

The landing gear adopts hydraulic retractable three-point type. Each landing gear is two-wheeled, with oil-gas shock absorbers installed. The main landing gear is housed in the rectifying foreskin of the wing and fuselage. The nose landing gear can be turned and longitudinally guided and moved forward into the fuselage. The main landing gear door is made of composite materials with carbon brakes.

The landing gear includes: two inward retractable main landing gear. The landing gear compartment door closes the potato carrier compartment. The landing gear and doors are electrically controlled and hydraulically operated. The hatch connected with the landing gear pillar is driven by the landing gear machinery, and the hatch is closed after the landing gear is fully retracted. During the retracting and retracting of the landing gear, all landing gear doors were opened.

The retracting and retracting of the landing gear and the operation of the door are controlled by the two landing gear control and interface components (LGCIUS). Two LGCIUS, one for current use and one for spare. Each LGCIU independently completes a complete cycle of retracting and retracting the landing gear. When the current LGCIU fails, it will automatically switch to the backup LGCIU to work.

LGCIU provides landing gear information to ECAM for display and air-ground logic signals for other aircraft systems. If the aircraft loses its hydraulic system or power, the crew can use a hand crank on the center console to lower the landing gear. Each main landing gear is a two-wheeled landing gear with an oil-gas type shock absorber, including the following components: main strut, shock strut outer tube, shock strut inner tube, retractable actuator tube, side palm lever, lock Simple connecting rod and actuation, anti-torsion arm and slave connecting rod.

The two-wheel front landing gear includes an oil-gas damping strut and a front wheel turning control system, including: a shock strut assembly, a resistance strut assembly, a lock rod assembly, a landing gear actuator, and a front wheel turning mechanism.

1.1.5. Flight Control System

The entire flight process from take-off to landing is controlled by the fly-by-wire control system. The main fly-by-wire control system is two independent systems with a total of 5 computers. Two are used for elevator and aileron control, and three are used for spoiler control. The rudder trim is completed by two flight stabilization computers, and the slat and flap control are calculated and controlled by two dedicated computers. All use fly-by-wire control, which greatly improves flight safety, reduces the workload of the pilot, and reduces the weight of the control system. Two measures have been taken to improve the reliability of the fly-by-wire system: the signal transmission cables leading to each control surface are separated from each other, and the aileron and spoiler cables are arranged in front of and behind the front beam of the wing. In order to prevent lightning strikes, all Fly-by-wire cables are all installed in metal shielding sleeves, and the exposed part is sleeved in the cable duct. The joystick uses a side stick instead of a common steering stick and hand-wheel, which can reduce the weight of the system. The side stick control device includes a joystick that tilts inward and forward, a roll and pitch sensor box, and an artificial sensor system. When using the autopilot, a clip driven by an electromagnetic coil locks the joystick in the center position. The electronic circuit is connected between the two sets of fly-by-wire control devices. In order to solve the contradiction between the two pilots' instructions to control the aircraft at the same time, a comparison device is installed in the electronic circuit. The fly-by-wire control system can superimpose the two input algebras. If the driver wants to cancel the input of the other driver, just press and hold the "takeover button" to cancel the other party's control input. There are 3 sets of hydraulic systems, respectively marked in green, yellow and blue. The green and yellow systems are interconnected, and each engine drives one. The blue hydraulic system is driven by an air ram turbine. Two of the generators are driven by the engine ice for normal power supply, and the other generator is driven by an auxiliary power unit. In addition to being used on the ground, it can also be used as an emergency backup power supply in the air. In the event that all three generators fail, there is also an emergency generator, driven by the blue hydraulic system. There is also a converter on the machine to provide DC power, and there are animal batteries.

Except for the 12 spare main digital instruments, all the data of the airborne equipment are displayed on the 6 color cathode ray tube displays arranged in a "+" shape. The installed display system belongs to the second generation, which consists of three display management computers and two system data acquisition devices. It adopts the second-generation digital automatic flight control system, which is integrated with all the capabilities of the fly-by-wire control system, the full power electronic control system of the engine, and the flight management system. Therefore, there is no special flight guidance computer and engine thrust control computer on the aircraft, and there is no independent co-server organization for autopilot and auto throttle. These functions are all included in the flight management computer. The instruction signal of the computer is provided to the pitch and roll control surface through the fly-by-wire control computer, and the yaw axis is controlled by the flight stabilization computer. Thrust control is included in the full-function electronic control system of the engine. The use of the second-generation digital automatic flight system with a higher degree of comprehensiveness improves safety and reliability, simplifies the system, reduces costs, and also reduces the weight of the aircraft. A centralized fault display system is adopted. When the aircraft system fails, the two cathode-ray tubes in the center of the dashboard respectively display warning signs and system movement conditions, and the multi-purpose controller and display device on the central console pass the integrated fault The display system automatically analyzes the cause of the failure.

1.1.6. Fuselage.

The fuselage is an all-metal semi-monocoque structure. It is designed to have the drag is low as possible, and it has the advantages of high critical value M.

There is space in the cockpit for the first and second pilots, and also have space for the flight engineer.

The main pilot is on the left side of the cockpit, the co-pilot is on the right, and the flight engineer is behind the co-pilot. The instrument panel is installed in front of the pilot, and the pilot's console is usually located between the two. The upper electrical panel is located above the roof glass. The left side of the fuselage is the first main side console, and the right side is the side console of the co-pilot.

Flight navigation instruments used to monitor the operation of power plants and other instruments and signal equipment are installed on the dashboard.

There are windows on the left and right sides of the salon, and emergency exits on the left and right sides. There are luggage racks for storing passengers' personal belongings on both sides of the cabin. The bottom of the luggage rack is equipped with an air-conditioning port with independent ventilation system, a lamp with controllable buttons and independent lighting, and the corresponding number of the call button and seat lamp of the flight attendant. Service panel. Generally, the internal lighting platform is located in the center of the ceiling: in addition, there are lights for the lower part of the luggage compartment and the inside of the passenger cabin.

Under the floor of the airtight part of the fuselage, there are the following rooms and compartments: the front landing gear of the chassis (air leak), the front cargo compartment, the rear cargo compartment, and the technical compartment. The front and rear cargo compartments (sealed), each cargo compartment has a hatch on the right side, and is equipped with a container locking system.

1.2 Substantiation of the new aircraft parameters

Layout of the aircraft consists from composing the relative disposition of its parts and constructions, and all types of the loads (passengers, luggage, cargo, fuel, and so on).

Choosing the scheme of the composition and aircraft parameters is directed by the best conformity to the operational requirements.

1.2.1 Wing geometry calculation

Geometrical characteristics of the wing are determined from the take off weight m_0 and specific wing load P_0.

Wing area is : $S_{wfull} = \frac{m_o * g}{P_o} = \frac{82867 \times 9.8}{6139}$ =132.28 (m^2).

So wing area is: S_w=132.28m^2.

Wing span is: $l = \sqrt{S_w * \lambda_w} = \sqrt{132.28 \times 9.48}$=35.4 ($m$).

Root chord is: $b_o = \frac{2S_w * \eta_w}{(1+\eta_w) * l} = \frac{2 \times 132.28 \times 4}{(1+4) \times 35.4} = 5.97$ (m).

Tip chord is: $b_t = \frac{b_o}{\eta_w} = \frac{5.97}{4} = 1.49$ (m).

Maximum wing width is determined in the forehead i-section and by its span it is equal:

$$c_i = c_w * b_t = 2 \times 1.49 = 2.98 \ (m).$$

At a choice of structural scheme of the wing we determine quantity of spars and its position from the leading edge of a wing.

According to the design of prototypes our aircraft has three spars.

I use the geometrical method of mean aerodynamic chord determination (Figure 1.2.1). Mean aerodynamic chord is equal: b_{MAC}=*4.178 m.*

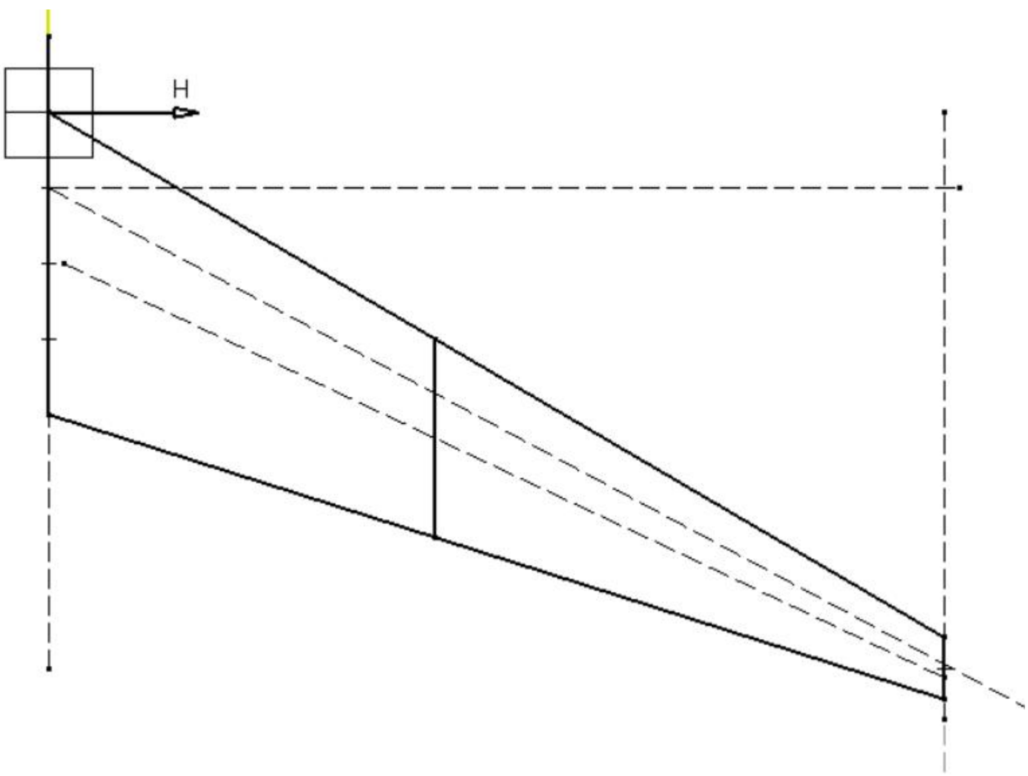

Figure 1.2.1 – Determination of mean aerodynamic chord

After determination of the geometrical characteristics of the wing we come to the estimation of the ailerons geometrics and high-lift devices.

Ailerons geometrical parameters are determined in next consequence:

Ailerons span: l_{ai}=(0.3....0.4) · $(\frac{l_w}{2})$=0.35× $\frac{35.4}{2}$ = 6.195(m).

Aileron area: S_{ai}=(0.05....0.08) · $(\frac{S_w}{2})$=0.065× $\frac{132.28}{2}$ = 4.3 (m).

Chord of aileron: C_{ai}=0.23C_W=0.23×2=0.46 (m).

Slat geometrical parameters are determined in next consequence:

Chord of slat: C_s=b_o×0.12=0.71 (m).

Flap geometrical parameters are determined in next consequence:

Length of flap: l_f=(0.3....0.4)C_w=0.35×2=0.7 (m).

Aileron geometrical parameters are determined in next consequence:

Ailerons trim tab length: l_{tr} = (0.04....0.06)S_{ai} = 0.05×4.3 = 0.215 (m).

Area of ailerons trim tab:

For two engines airplane: S_{tail}=(0.05....0.06)*S_{ail} = 0.055×4.3=0.2365 (m^2)

Range of aileron deflection

Upward $\delta'_{ail} \geq 20°$;

Downward $\delta''_{ail} \geq 10°$.

The aim of determination of wing high-lift devices geometrical parameters is the providing of take of and landing coefficients of wing lifting force, assumed in the previous calculations with the chosen rate of high-lift devices and the type of the airfoil profile.

Before doing following calculations it is necessary to choose the type of airfoil due to the airfoil catalog, specify the value of lift coefficient $C_{ymax\ bw}$ and determine necessary increase for this coefficient C_{ymax} for the high-lift devices outlet by the formula: $\Delta C_{ymax} = (\frac{C_{ymaxl}}{C_{ymax\ bw}})$.

Where C_{ymax} is necessary coefficient of the lifting force in the landing configuration of the wing by the aircraft landing insuring (it is determined during the choice is the aircraft parameters).

In the modern design the rate of the relative chords of wing high-lift devices is:

b_f = 0.3..0.4 – for three slotted flaps and Faylers flaps;

b_s = 0.1..0.15 – slats.

Effectiveness of high-lift devices (C_{ymaxl}) rises proportionally to the wing span increase, serviced by high-lift devices, so we need to obtain the biggest span of high lift devices ($l_{hld} = l_w - D_f - 2l_{ail} - l_n$) due to use of flight spoiler and maximum diminishing of the are of engine and landing gear nacelles.

During the choice of structurally-power schemes, hinge-fitting schemes and kinematics of the high-lift devices we need to come from the statistics and experience of domestic and foreign aircraft construction. We need to mention that in the majority of existing constructions elements of high-lift devices are done by longeron structurally-power schemes.

1.2.2 Fuselage layout

During the choice of the shape and the size of fuselage cross section we need to come from the aerodynamic demands (streamlining and cross section).

Applicable to the subsonic passenger aircrafts (V < 340 m/s) wave resistance doesn't affect it. So we need to choose from the conditions of the list values friction resistance C_{xf} and profile resistance C_{xp}.

During the transonic and subsonic flights, shape of fuselage nose part affects the value of wave resistance C_{xw}. Application of circular shape of fuselage nose part significantly diminishing its wave resistance.

Except aerodynamic requirements consideration during the choice of cross section shape, we need to consider the strength and layout requirements.

For ensuring of the minimal weight, the most convenient fuselage cross section shape is circular cross section. In this case we have the minimal fuselage skin width. As the partial case we may use the combination of two or more vertical or horizontal series of circles.

To geometrical parameters we concern: fuselage diameter D_f; fuselage length l_f (shown at figure 1.2.2); fuselage fineness ratio λ_f ; fuselage nose part fineness ratio

λ_{np} ; tail fineness ratio λ_{TU} . Fuselage length is determined considering the aircraft scheme, layout and airplane center-of-gravity position peculiarities, and the conditions of landing angle of attack α_{land} ensuring.

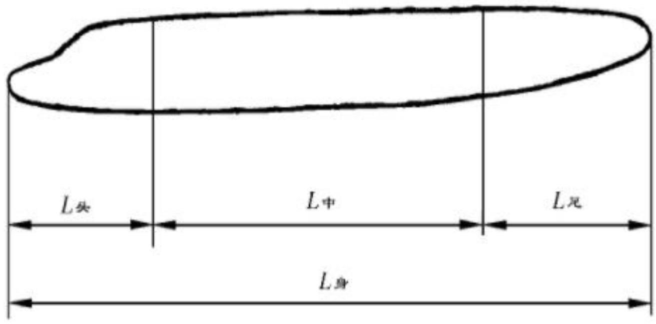

Figure 1.2.2 – Three-sections of the fuselage(the length of the nose part, central part and rear part of the fuselage)

Fuselage length is equal: l_f=$D_f \cdot \lambda_f$ = 3.95×8.57=33.85 (m).

Determine the length of the fuselage nose part and length of the fuselage rear part according to the fuselage aspect ratio λ_{fnp} and λ_{frp} .we may take the ratio as:

Fuselage nose part fineness ratio $\lambda_{fnp}=1.2.....2.5$.

Fuselage rear part fineness ratio $\lambda_{frp}=2.....5$.

Length of the fuselage nose part is equal: $l_{fnp}=\lambda_{fnp} \cdot D_f$ =1.5×3.95=5.925 (m).

Length of the fuselage rear part is equal: $l_{frp}=\lambda_{frp} \cdot D_f$ =2.5×1.3=3.25 (m).

During the determination of fuselage length we seek for approaching minimum mid-section S_{ms} from one side and layout demands from the other.

For passenger airplanes fuselage mid-section first of all comes from the size of passenger cabin. One of the main parameter, determining the mid-section of passenger airplane is the width of the passenger cabin.

For three seats of one block distribution we may take the width as: (3*3)

economic class seat width $b_{3ec}=1455.....1650$ (mm);

business class seat width $b_{3bu}=1500.....1770$ (mm).

The distance from the outside of the seat handle to the inner wall of the fuselage $\delta_1=40.....50$ (mm).

The distance between inner and outer walls of the fuselage δ_{wall}=80.....120 (*mm*).

For two seats of one block distribution we may take the width as:

economic class seat width b_{2ec}=960.....1090 (*mm*);

business class seat width b_{2bu}=1020.....1200 (*mm*).

For aisle width we may take as:

$b_{ais\text{-}ec}$=400.....510 (*mm*);

$b_{asi\text{-}bu}$=500.....600 (*mm*).

For economic class with the scheme of allocation of seats in the one row (3 + 3) determine the appropriate width of the cabin (figure 1.2.3), I choose the next parameters:

Economic class cabin width is equal:

$B_{ec}=n_{bl3}\cdot b_{3ec}+n_{aisle}\cdot b_{aisle}+2\cdot\delta_1+2\cdot\delta_{wall}$

$=2\times1600+1\times450+2\times50+2\times100=3950$ (*mm*).

Business class cabin width is equal:

$B_{bu}=n_{bl3}\cdot b_{3ec}+n_{aisle}\cdot b_{aisle}+2\cdot\delta_1+2\cdot\delta_{wall}$

$=2\times1600+1\times500+2\times40+2\times80=3940$ (*mm*).

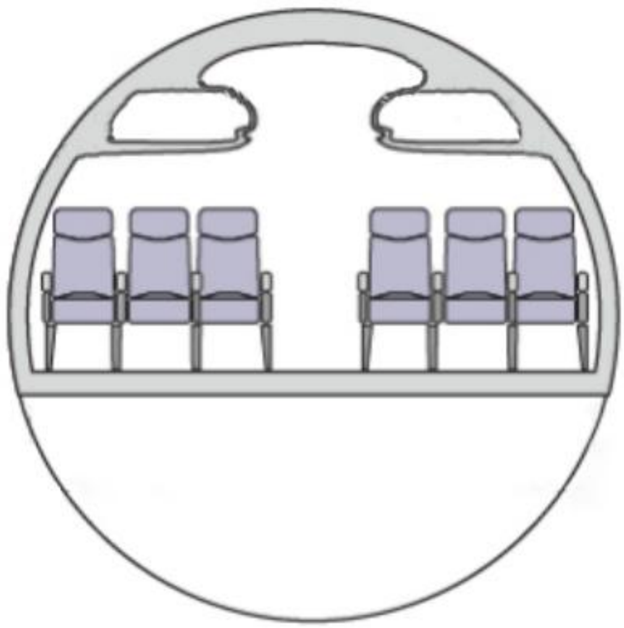

Figure 1.2.3 – Passenger cabin cross-section

From the design point of view it is convenient to have round cross section, In the most cases, one of the most suitable ways is to use the combination of two circles intersection, or oval shape of the fuselage. We need to remember that the oval shape is not suitable in the production, because the upper and lower panels will bend due to extra pressure and will demand extra bilge beams, and other construction amplifications.

Step of normal bulkhead in the fuselage construction is in the range of 360...500 mm, depends on the fuselage type and class of passenger saloon.

Form the design consideration with the diameter less than 2800 mm we don’t use such shape and we follow to the intersecting circles cross section. In this case the floor of the passenger saloon is done in the plane of are closing.

The windows are placed in one light row. The shape of the window is round, with the diameter of (300...400 mm), or rectangular with the rounded corners. The window step corresponds to bulkhead step and is (500...510 mm).

Next, calculate the length of the cabin. I use economy class 3+3 and business class 3+3 seat distribution, so economy class can be arranged with 144 passenger seats (24 rows in total), and business class can be arranged with 12 passenger seats (2 rows in total), according to the cruise time, we can select the required value for the seat pitch (figure 1.2.4) from the following parameters:

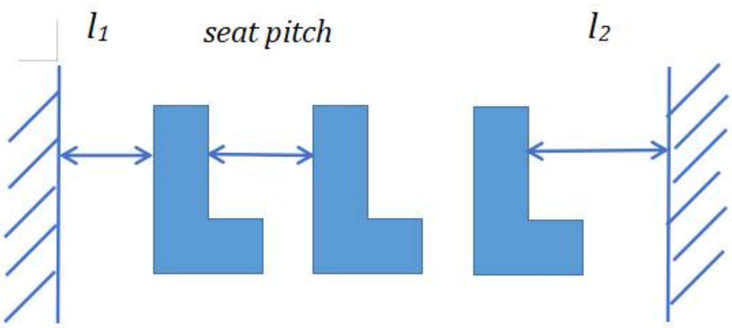

Figure 1.2.4 – Seat pitch dimension: l_1=250 mm; l_2=1200.....1300 mm

Seat pitch select shown at table 1.2.

Table 1.2 – Seat pitch selection reference table

type of class	less than 3hours	more than 3hours
first	1020	1080
business	960	960
economic	810	870

Length of the economic cabin:

$L_{ec}=l_2$+seat pitch(n-1)+l_1=1200+(24-1)×870+250=21460 (*mm*).

Length of the business cabin:

$L_{bu}=l_2$+seat pitch(n-1)+l_1=1250+1×960+250=2460 (*mm*).

1.2.3 Layout and calculation of basic parameters of tail unit

One of the most important tasks of the aerodynamic layout is the choice of tail unit placing. For ensuring longitudinal stability during overloading its center of gravity should be placed in front of the aircraft focus and the distance between these points, related to the mean value of wing aerodynamic chord, determines the rate of longitudinal stability.

$$m^{Cy}{}_{x} = \bar{\chi}_T - \bar{\chi}_F < 0$$

Where $m^{Cy}{}_{x}$ –is the moment coefficient; $\bar{\chi}_T$, $\bar{\chi}_F$ center of gravity and focus coordinates. If $m^{Cy}{}_{x}$=0, then the plane has the neutral longitudinal static stability, if $m^{Cy}{}_{x}$>0, thn the plane is statically instable. In the normal aircraft scheme (tail unit is behind the wing), focus of the combination wing – fuselage during the install of the tail unit of moved back.

The values of L_{htu} and L_{vtu} depend on several factors. Their values are affected by: the length of the fuselage nose and tail, the position of the swept wings and the wings, and the stability and control conditions of the aircraft. This is determined according to the parameters of the prototype and the related proportions.

Determination of the tail unit geometrical parameters(in order to obtain a more precise result, we calculate them by 2 methods then select the average):

Length of horizontal tail unit is equal:

Compare with prototype:

$$\frac{L_{fuse}}{L_{pro\ fuse}} = \frac{L_{HTU}}{L_{pro\ HTU}}. \qquad L_{HTU} = 10.56\ (m).$$

$$\frac{L_{fuse}}{b_{MAC}} = \frac{L_{pro\ fuse}}{b_{pro\ MAC}}$$

$\frac{L_{HTU}}{b_{MAC}}$= (2.5.....3.5);

L_{HTU} = 2.6×4.178 = 10.86 (m).

Area of horizontal tail unit is equal:

A_{HTU} = (0.65.....0.8);

$$S_{HTU1} = \frac{b_{MAC} * S_W * A_{HTU}}{L_{HTU}} = \frac{4.178 \times 132.28 \times 0.7}{10.86} = 35.62\ (m^2).$$

S_{HTU2} = (0.18.....0.25)S_W = 0.2×132.28=26.456 (m^2);

Finally S_{HTU} = 30.33 (m^2).

Length of vertical tail unit is equal:

$L_{VTU} \approx L_{HTU}$=10.86 (m).

Area of vertical tail unit is equal:

A_{VTU} = (0.08.....0.12).

$$S_{VTU1} = \frac{lw * S_W * A_{VTU}}{L_{VTU}} = \frac{0.1 \times 132.28 \times 35.4}{10.86} = 43.11\ (m^2).$$

S_{VTU2} = (0.12.....0.2)S_W = 0.16×132.28 = 21.165 (m^2).

Finally S_{VTU} = 31.57 (m^2).

Determination of the elevator area and direction:

Altitude elevator area: S_{el}=(0.3......0.4) · S_{HTU}= 9.7 (m^2).

Rudder area: S_{rud}=(0.2.....0.22) · S_{VTU}= 6.62 (m^2).

Choose the area of aerodynamic balance.

$0.3 \le M \le 0.6$, S_{abel}=(0.22.....0.25)S_{el}, $S_{abrudder}$=(0.2.....0.22)S_{rud}.

Elevator balance area is equal:

S_{abel} =(0.22.....0.25)S_{el} = 2.231 (m^2).

Rudder balance area is equal:

S_{abrud} =(0.2.....0.22)S_{rud} = 1.39 (m^2).

The area of altitude elevator trim tab:

$S_{te} = 0.08 \cdot S_{el} = 0.67\ (m^2)$.

Area of rudder trim tab is equal:

$S_{tr} = 0.06 \cdot S_{rud} = 0.42\ (m^2)$.

Determination of the TU span.

TU span is related to the following dependence:

$L_{ro} = (0.32..0.5) l_w = 0.4 \times 35.4 = 14.16\ (m)$.

In this dependence the lower limit corresponds to the turbo jet engine aircraft, equipped with all-moving stabilization.

The height of the vertical TU h_{bo} is determined accordingly to the location of the engines. Taking it into account we assume:

Low wing, EonW, M<1 $h_{bo} = (0.14..0.2) l_w$

Engine in the root part of the wing $h_{bo} = (0.13..0.165) l_w$

$h_{bo} = 0.16 \times 35.4 + 3.7 = 9.369\ (m)$

Tapper ratio of horizontal and vertical TU we need to choose:

For planes M<1 ; $\eta_{ro} = (2.....3)$ $\eta_{bo} = (1.....1.33)$

TU aspect ratio, We may recommend:

For transonic planes $\lambda_{bo} = (0.8.....1.5)$

$\Lambda_{ro} = (3.5.....4.5)$

Relative thickness (t/c ratio) of the profile.

For horizontal and vertical TU in the first approach, . $\bar{C}_{TU} \approx 0.08\ \bar{C}_w$;

For more accurate: Transonic $\bar{C}_{TU} \approx (0.06.....0.09)$

1.2.4 Galleys and wardrobes

International standards stipulate that if the aircraft is made into a mixed layout, two things must be provided. If the flight time is less than 3 hours and no meals are delivered, tea will be provided in this case. No buffet will be provided if the flight time is less than one hour. The kitchen cabinets must be close to the front door, preferably a separate door between the cockpit and the passengers or cargo. This aircraft is designed with two galleys and two wardrobes, located at the nose and tail of the aircraft.

Volume of galley is equal:

V_{gally}=(0.1.....0.12)n_{pass}=0.11×156=17.17 (m^3).

Height of cabin: H=1.48+0.17D_f= 1.48+0.17×3.95=2.151 (m).

Area of galley is equal:

$S_{area}= \frac{V_{gally}}{H} = \frac{17.17}{2.151}$ =7.98 (m^2).

Aera of the floor for wardrobe is equal:

S_{war}=(0.035.....0.04)n_{pass}=0.036×156=5.616 (m^2).

1.2.5 Lavatories

Number of toilet facilities is determined by the number of passengers and flight duration: with t > 4h one toilet for 40 passengers, at t=2.....4 hours and 50 passengers t < 2 hours to 60 passengers.This aircraft has a total of four toilets, two are located the nose of fuselage which directly between the cockpit and business class, and the other two are located at the rear of the aircraft.

The number of lavatories I choose according to the original airplane and it is equal:

n_{lava}=4.

Area of lavatory:

S_{lava}=9.50×1.150=1.0925 (m^2).

Toilets design similar to the prototype.

1.2.6 Landing gear design

In the primary stage of design, when the airplane center-of-gravity position is defined and there is no drawing of airplane general view, only the part of landing gear parameters may be determined(shown at figure 1.2.5).

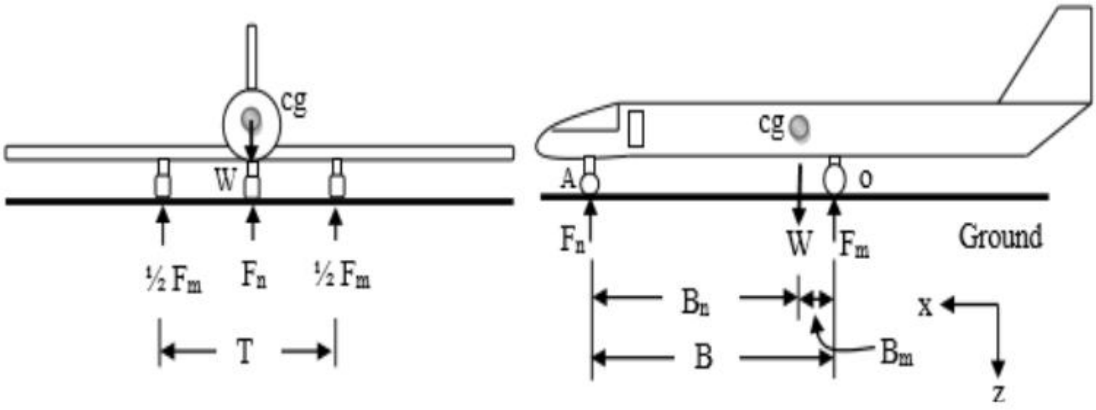

Figure 1.2.5 – Schematic diagram of aircraft landing gear length

The distance from the centre of gravity to the main LG

B_m = (0.15..0.20)b_{MAC} = 0.188×4.178=0.785 (m).

With the large distance the lift of the nose gear during take of is complicated, and with small, the strike of the airplane tail is possible, when the loading of the back of the airplane comes first. Besides the load on the nose LG will be too small and the airplane will be not stable during the run on the slickly runway and side wind.

Landing gear wheel base comes from the expression:

B = (0.3..0.4)l_f = 0.35×33.85=11.85 (m).

Large value belongs to the airplane with the engine on the wing.

The last equation means that the nose support carries 6..10% of aircraft weight.

The distance from the centre of gravity to the nose LG:

$B_n = B - B_m$ =11.85 – 0.785 = 11 (m).

Wheel track is: T = (0.7..1.2)B =10.66$m \le$ 12m.

On a condition of the prevention of the side nose-over the value K should be > 2H, where H – is the distance from runway to the center of gravity.

Wheels for the landing gear is chosen by the size and run loading on it from the take off weight; for the front support we consider dynamic loading also.

Type of the pneumatics (balloon, half balloon, arched) and the pressure in it is determined by the runway surface, which should be used. We install breaks on the main wheel, and sometimes for the front wheel also.

The load on the wheel is determined:

K_g = 1.5......2.0 – dynamics coefficient.

Nose wheel load is equal:

$$F_{nose} = \frac{B_m * m_o * 9.81 * K_g}{B * z}$$

$$= \frac{0.785 \times 82867 \times 9.81 \times 1.75}{11.85 \times 2} = 47120.5\ (N).$$

Main wheel load is equal:

$$F_{main} = \frac{(B - B_m) m_o * 9.81}{B * n * z}$$

$$= \frac{(11.85 - 0.785) \times 82867 \times 9.81}{11.85 \times 2 \times 4} = 94884.2\ (N),$$

where-- n, and z – are the quantities of the supports and wheels on the one leg.

By calculated F_{main} and F_{nose} and the value of V take off and Vl_{anding}, pneumatics is chosen from the catalog, the following correlations should correspond.

$$P^K_{slmain} \geq P_{main}\ ;\ P^K_{slnose} \geq P_{nose}\ ;\ V^K_{landing} \geq V_{landing}\ ;\ V^K_{take\ off} \geq V_{take\ off},$$

where-- K is the index designated the value of the parameter allowable in catalog.

A common classification of aircraft tires is by type as classified by the United States Tire and Rim Association.In our case, I select Radial aircraft tires(Three part type). All new sizes being developed are in this classification. This group was developed to meet the higher speeds and loads of today’ s aircraft. It is the most modern design of all tire types.(shown at figure 1.2.6).

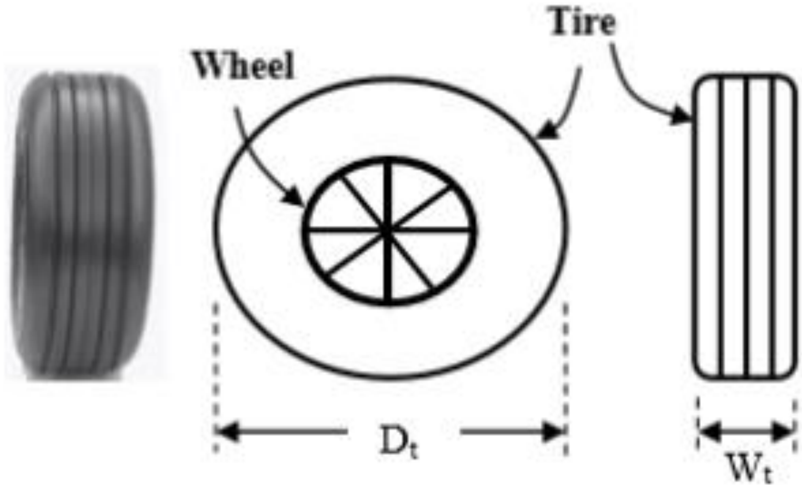

Figure 1.2.6 – Tire principle dimensions

According to the Goodyear aircraft tire data book. The dimensions can be sure:

Main tire: 50×18R22;

Auxiliary tire: 30×8.8R15;

As for brakes, I select carbon brakes, there are superior performance, are lighter, easier to maintain and qualified requirements.

1.2.7 Power Plant choice

With reference to the A320 prototype, this aircraft uses the engine-IAE V2500. There is a dual rotor, axial flow, high channel ratio turbofan engine-IAE V2500 suspended under the wings on both sides(the main characteristics are presented in the table 1.3). The engine uses wide-chord hollow fan blades without bosses. With the characteristics of "floating wall" combustion and efficient fuel rate, each engine has a digital full-power electronic throttle control system and thrust reverser.

Table 1.3 – The main parameters of the IAE V2500

Variant	Take off thrust	Thrust to weight ratio	Bypass ratio	Compression ratio
V2500-A1	110.31KN(24800lbf)	4.68	5.4:1	35.8

General characteristics:

Type: Dual rotor, axial flow, high bypass turbofan

Length: 3.201 m (126.0 in)

Diameter: 1.682 m (66.2 in) width, 63.5 in (1.613 m) Fan diameter [a]

Dry weight: 2,404–2,595 kg (5,300–5,721 lb)

Components:

---compressor: 1 fan, 4 stages of low pressure, 10 stages of high pressure;

---annular combustors;

turbine: 2 stages of high pressure, 5 stages of low pressure.

Performances:

Maximum thrust: 102.48–140.56 kN (23,040–31,600 lbf);

Thrust-to-weight ratio: 4.18-5.73;

Rotor speed: LP: 5,650 RPM, HP: 14,950 RPM;

Control: Dual channel FADEC.

1.3 Calculation of the aircraft center of gravity positions

The aircraft center of gravity affects its stability and control, thus it must be found for typical variants of loading.

1.3.1 Determination of centering of the equipped wing

Mass of the equipped wing contains the mass of its structure, mass of the equipment placed in the wing and mass of the fuel. Regardless of the place of mounting (to the wing or to the fuselage), the main landing gear and the front gear are included in the mass register of the equipped wing. The mass register includes names of the objects, mass themselves and their center of gravity coordinates. The origin of the given coordinates of the mass centers is chosen by the projection of the nose point of the mean aerodynamic chord (MAC) for the surface XOY. The positive meanings of the coordinates of the mass centers are accepted for the end part of the aircraft.

The example list of the mass objects for the aircraft, where the engines are located under the wing, included the names given in the table 1.4. The mass of aircraft is 82867 kg. Coordinates of the center of power for the equipped wing are defined by the formulas:

$$X'_w = \frac{\Sigma m_{i'} x_{i'}}{\Sigma m_{i'}}$$

Table 1.4 - Trim sheet of equipped wing masses

N	object name	Mass		C.G coordinates Xi, 10^{-3}m	Mass moment, Xi * mi
		Units 10^{-3}	total mass m(i),10^{-3} kg		
1	Wing (structure)	106.34	8812076.78	1796.54	15831.24
2	Fuel system	9.90	820383.30	1775.65	1456.71
3	Flight control system , 30%	1.80	149160.60	2506.80	373.91
4	Electrical equipment, 10%	3.22	266831.74	417.80	111.48
5	Anti-ice system , 40%	8.72	722600.24	417.80	301.90
6	Hydraulic systems , 70%	11.55	957113.85	2506.80	2399.29
7	Power plant	88.70	7350302.90	-1600.00	-11760.48
9	**Equipped wing without landing gear and fuel**	**230.23**	**19078469.41**	**456.75**	**8714.07**
10	Nose landing gear	5.50	455768.50	-12860.00	5861.18
11	Main landing gear	31.20	2585450.04	850.00	2197.63
12	Fuel	334.28	27700780.76	200.00	55401.56
13	**Total**	**601.21**	**49820469.07**	**1213.40**	**60452.08**

According to the table 3.1: $X'_w = \frac{\Sigma m_{i'} x_{i'}}{\Sigma m_{i'}} = \frac{\mathbf{60452.082}}{\mathbf{49820.469}} = 1.21.$

1.3.2 Determination of centering of the equipped fuselage

Origin of the coordinates is chosen in the projection of the nose of the fuselage on the horizontal axis. For the axis X the construction part of the fuselage is given. The example list of the objects for the aircraft, which engines are mounted under the wing, is given in table 1.5.

The CG coordinates of the fully equipped fuselage are determined by formulas:

$$X_f = \frac{\Sigma m_{i'} x_{i'}}{\Sigma m_{i'}}$$

After we determined the C.G. of fully equipped wing and fuselage, we construct the moment equilibrium equation relatively to the fuselage nose:

$$m_f x_f + m_w (x_{MAC} + x'_w) - m_o C = m_o (x_{MAC} + C)$$

From here we determined the wing MAC leading edge position relative to fuselage, means $\boldsymbol{X_{MAC}}$ value by formula:

$$X_{MAC} = \frac{mf\ xf + mw * x'w - moC}{mo - mw}$$

where $\boldsymbol{m_0}$ – aircraft takeoff mass, kg; $\boldsymbol{m_f}$ – mass of fully equipped fuselage, kg; $\boldsymbol{m_w}$ – mass of fully equipped wing, kg; $\boldsymbol{C}$ – distance from MAC leading edge to the C.G., determined by the designer.

$\boldsymbol{C}$ = (0,22...0,25) B_{MAC} – low wing.

Table 1.5 – Trim sheet of equipped fuselage masses

N	Objects names	Mass Units 10^{-2}	Mass total mass 10^{-2}	C.G coordinates Xi, m	mass moment
1	Fuselage	6.80	563329.87	18.00	101399.37
2	Horizontal tail	0.86	71514.22	32.00	22884.55
3	Vertical tail	0.85	70602.68	31.50	22239.84
4	Radar	0.31	25688.77	1.00	256.88
5	Radio equipment	0.23	19059.41	1.00	190.59
6	Instrument panel	0.54	44748.18	2.00	894.96
7	Aero navigation equipment	0.47	38947.49	2.00	778.94
10	Flight control system 70%	0.42	34804.14	16.93	5890.60
11	Hydraulic system 30%	1.16	95711.39	23.70	22678.81
12	Electrical equipment 90%	2.90	240148.57	13.93	33440.68
13	Not typical equipment	0.33	27346.11	3.00	820.38
14	Lining and insulation	0.65	53863.55	16.93	9116.40
15	Anti ice system, 20%	0.44	36130.01	27.08	9784.01
16	Airconditioning system, 40%	0.87	72260.02	16.93	12230.01
17	Passenger seats (bussiness)	0.12	9600.00	5.93	568.81
18	Passenger seats (economic class)	1.22	100800.00	20.00	20159.99
19	Seats of flight Attendence	0.03	2400.00	1.82	43.68
20	Seats of pilot	0.02	2000.00	1.48	29.60
	Emergency equipment	0.14	11986.51	8.10	970.90
	Lavatory1, galley 1	0.60	49720.20	2.96	1471.71

Ending of table 1.5 – Trim sheet of equipped fuselage masses

	Lavatory2, galley 2	0.60	49720.20	30.00	14916.06
	Operational items	0.22	17940.71	26.00	4664.58
21	Additional eguipment	0.33	27428.98	5.00	1371.44
22	**Equipped fuselage without payload**	**20.10**	**16657.51**	**17.22**	**286802.87**
	Passengers(economy)	13.38	11088.00.	20.00	221760.00
23	Passengers(bussiness)	1.12	924.00.	5.93	5474.70
24	On board meal	0.28	234.00.	30.00	7020.00
25	Baggage	3.77	3120.00.	18.00	56160.00
26	Cargo, mail	0.78	650.00.	18.00	11700.00
	Flight attend	0.29	240.00.	17.00	4080.00
27	Crew	0.19	154.00.	2.40	369.60
29	**TOTAL**	**39.90**	**33067.51**	17.94	**593367.17**

According to the table 3.1: $X_f = \frac{\Sigma m_{i'} x_{i'}}{\Sigma m_{i'}} = \frac{\mathbf{593367.17}}{\mathbf{33067.51}} = 17.94;$

$$X_{MAC} = \frac{mf\ xf + mw * x'w - moC}{mo - mw} = 17.165.$$

1.3.3 The range of aircraft centers of gravity

The list of mass objects for aircraft centre of gravity variant calculation given in Table 1.6 and Center of gravity calculation options given in table 1.7, completes on the base of both previous tables.

Table 1.6 – Calculation of C.G. positioning variants

Name	mass in kg	coordinate	mass moment
Object	m_i	X_i, м	Kgm
equipped wing (without fuel and landing gear)	19078.47	17.62	336208.10
Nose landing gear (extended)	455.77	5.00	2278.84
Main landing gear (extended)	2585.45	17.86	46176.14
Fuel reserve	2873.83	19.17	55078.73

Ending of table1.6-Calculation of C.G.positioning variants

Fuel for flight	24826.95	19.17	475824.31
Equipped fuselage (without payload)	16657.51	17.22	286802.87
Passengers(economy)	11088.00	20.00	221760.00
Passengers(bussiness)	924.00	5.93	5474.70
On board meal	234.00	30.00	7020.00
Baggage	3120.00	18.00	56160.00
Cargo, mail	650.00	18.00	11700.00
Mlight attend	240.00	17.00	4080.00
Crew	154.00	2.40	369.60
Nose landing gear (retrected)	455.77	4.00	1823.07
Main landing gear (retrected)	2585.45	17.86	46176.14

Table 1.7 – Airplanes C.G. position variants

№ п/п	Names	Маса, m_i кГ	Mass moment m_iX_i	Center of mass $X_{цм}$	Center of gravity position
1	Take off mass (L.G. extended)	82867.00	1508933.30	18.21	0.25
2	Take off mass (L.G. retracted)	82867.00	1508477.53	18.20	0.24
3	Landing weight (LG extended)	58061.03	1033108.99	17.79	0.15
4	Ferry version (without payload, max fuel, LG retracted)	66631.98	1202282.83	18.04	0.21
5	Parking version (without payload, without fuel foe flight, LG extended)	41651.03	726544.69	17.44	0.16

As it is seen from the table 1.7 the aircraft center of gravity range is correspondent to the characteristics of the advanced contemporary planes of the similar category and aerodynamic scheme.

Conclusion to the part 1

First part of the diploma work considers the primary stages of the typical preliminary design process.

The input parameters have been selected on the base of contemporary aircraft of the similar category, capacity, speed, aerodynamic scheme, etc.

Results of the calculations meet requirements of the Native (CCAR-25 – Chinese Civil Aviation Regulations), and International Airworthiness standards (CS-25 – Certification specification; FAR-25 – Federal Aviation Regulation).

PART 2. EFFECT OF LUGGAGE ON THE CRASHWORTHINESS OF FUSELAGE STRUCTURE

2.1. Introduction

Crashworthiness refers to the ability of the aircraft structure and its internal systems to protect occupants from injury in the event of a collision[10]. Collision accidents mostly occur during take-off and landing stages. Once a crush occurs, the survivability of the crew and passengers depends on the how much energy can the aircraft structure absorb, the more energy the aircraft structure absorbs, the more living space can be left for the crew, we need to limit the load delivered to the passengers.

After research and analysis, the luggage compartment has a significant impact on the entire cargo hold structure, which can absorb the energy generated by the collision and play a certain supporting role,the structure can remain intact. If there is no luggage, the cabin structure will be seriously damaged. In addition to this, based on the detailed nonlinear finite element model of the aircraft fuselage section, it can be concluded that the model with luggage has lower peak acceleration values and shorter pulse duration compared to the model results without luggage . Otherwise, luggage stiffness, luggage viscosity, and worst-case scenarios also have an impact on crashworthiness, and a stronger and stiffer luggage can stimulate the cargo floor structure's potential ability to absorb more energy during an impact.

The medium-range passenger aircraft in this paper has a large cargo space under the cabin floor and has a large deformation space.Therefore, the crashworthiness response analysis is carried out.

2.2. Calculation part

2.2.1 Finite element model

The total weight of the finite element model is 3013kg, the weight of fuselage structure is 1879kg and that of the luggage is 1134kg. The fuselage section (fig2.2.1) was configured to simulate the load density at the maximum take-off weight condition.

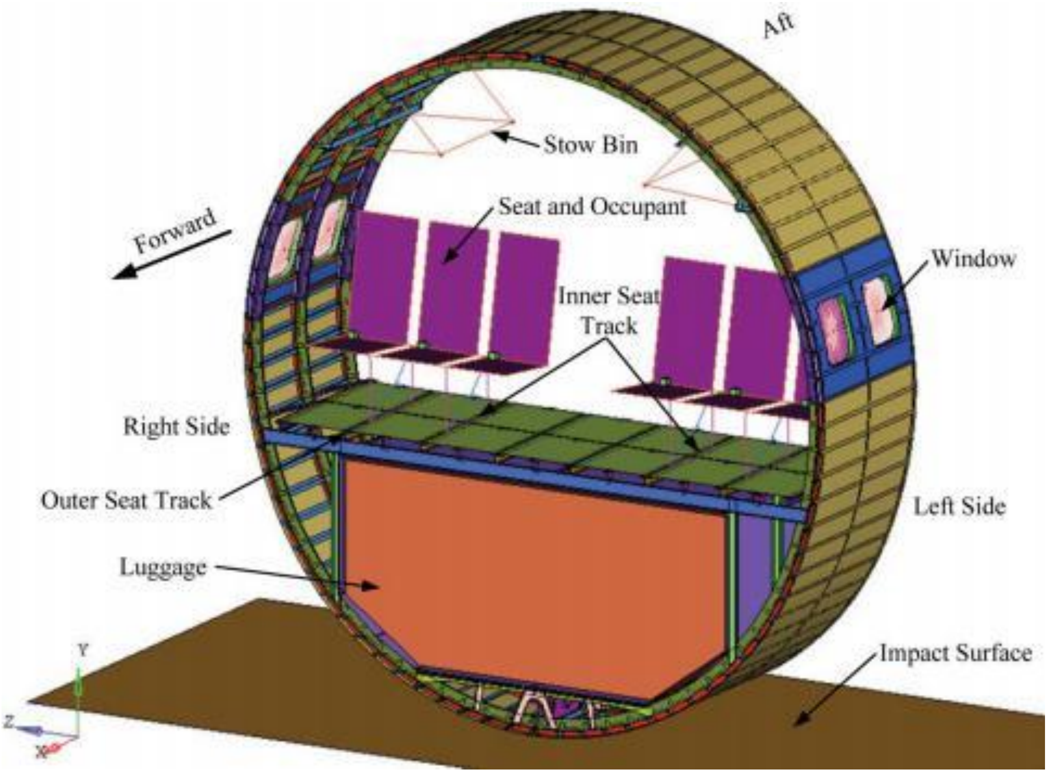

Figure 2.2.1 - Finite element model of fuselage section with luggage

The total energy generated in the event of a collision is dissipated through the fuselage structure, luggage and friction and can be written as the following formula:

$$E_{total} = \frac{1}{2}MV2 + Mgh = E_{absorb} = E_{frame}+E_{luggage}+E_{friction}$$

According to the existing conclusions, it can be inferred that the energy dissipated by the luggage, the frame and the friction part accounts for about 45%, 19.5% and 11.5%, respectively, and the luggage consumes nearly half of the energy, so it is the most important part of energy dissipation.

Assuming that the aircraft is in the landing stage, the landing speed $V_{vertical}$=7.12m/s, the fall height is 5 m, and the mass of the aircraft=78000kg:

$$E_{total} = \frac{1}{2}MV2 + Mgh = \frac{1}{2}\times 78000\times 7.122 + 78000\times 9.8\times 5$$

$$= 5.799\times 10^5 \ (J)$$

The energy dissipated by luggage approximately

$$=45\%E_{total} =2.61\times 105(J)$$

2.2.2 Relative material selection

The frame, beam, stanchions, outer skin, and longitudinal beams of this aircraft are all made of aluminum alloy (aluminum 2024-T3), the cabin floor(fig2.2.2), cargo and cargo floor are made of composite materials, and the main components are connected by riveting. When a material on a rigid surface comes into contact with a rigid ground, a huge amount of energy *(E_{total})* is generated.

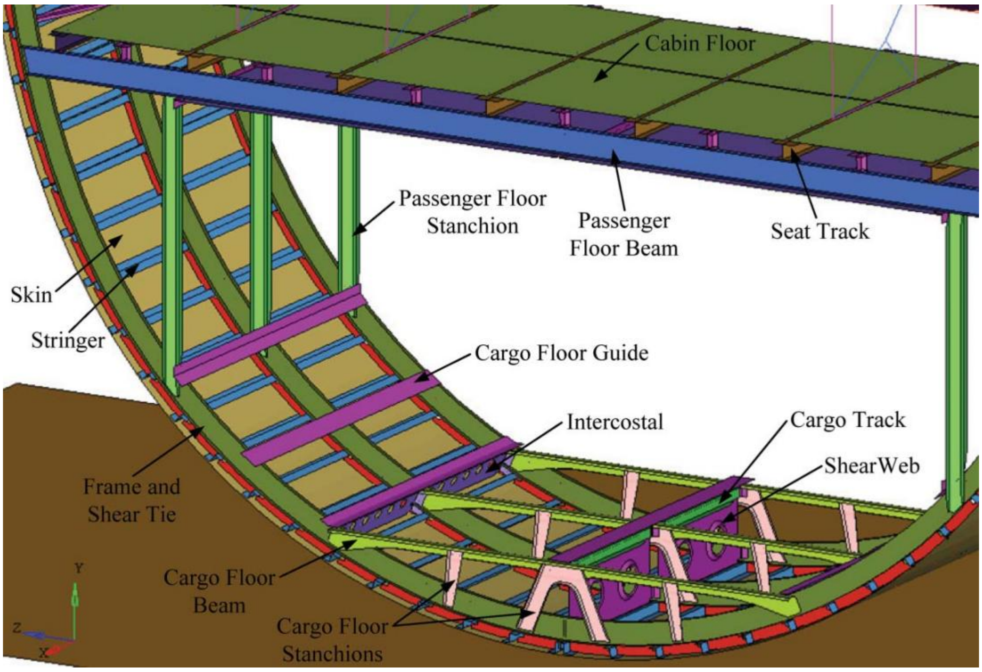

Figure 2.2.2 - The under-floor structure of cabin (luggage and cargo floor hidden)

2.2.3. Luggage stiffness

Increased luggage stiffness results in increased damage to the bottom structure, not only is the acceleration at the seat track correspondingly mitigated, but also means more energy is absorbed.

The stiffness calculation formula ***K=P/δ***, where P is the constant force acting on the structure, δ is the deformation due to the force, the unit is N/m： (assume deformation $\delta=550mm$)

$$Vt^2-V0^2=2ah \;;\; F=ma;$$

$$0-7.12^2=2a\times 5 \;;\; a=-5.0694(m/s^2)$$

$$P=F=ma=78000\times 5.0694=395416(N)$$

$$K=P/\delta=0.000718(Mpa)$$

So luggage stiffness must be at least bigger than *0.000718Mpa.*

2.2.4 Luggage viscosity

As the viscosity of the luggage increases, the damage to the cargo compartment structure and the frame decreases, the rebound speed decreases, the energy that the luggage can absorb increases(fig2.2.3), and the more energy is dissipated, so it is better to choose a larger luggage viscosity.

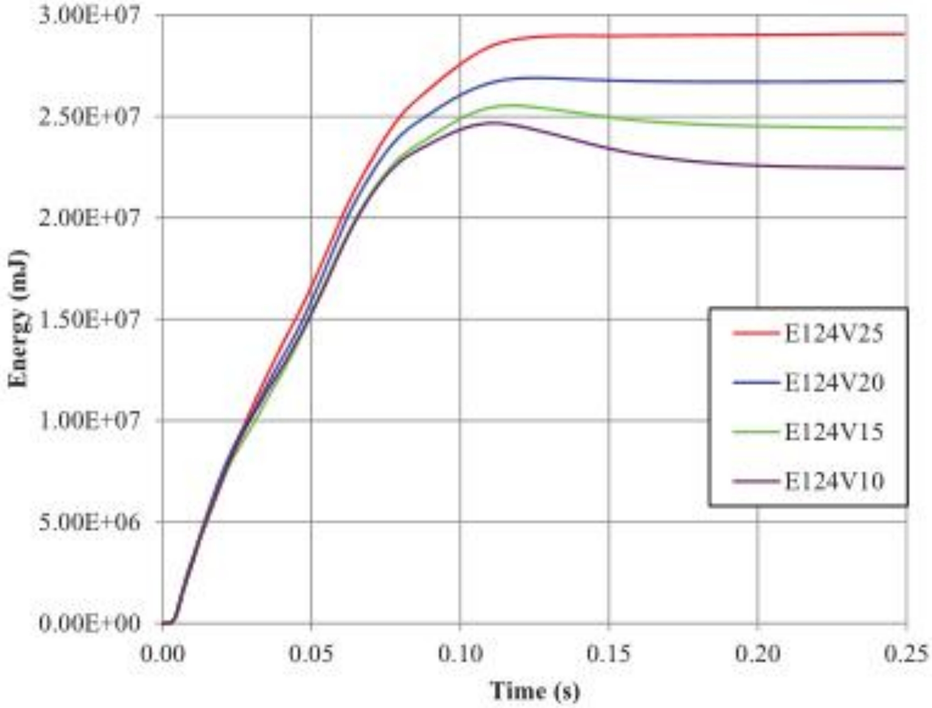

Figure 2.2.3. Energy absorption at different luggage viscosity

2.2.5 The worst-case scenario

The worst case exists when the higher stiffness and lower viscosity cargo configuration is the case, the individual structures are largely destroyed(fig2.2.4), but it is worth noting that even with different stiffness, viscosity and loaded luggage, the resistance of the fuselage Crash is still better than without luggage.

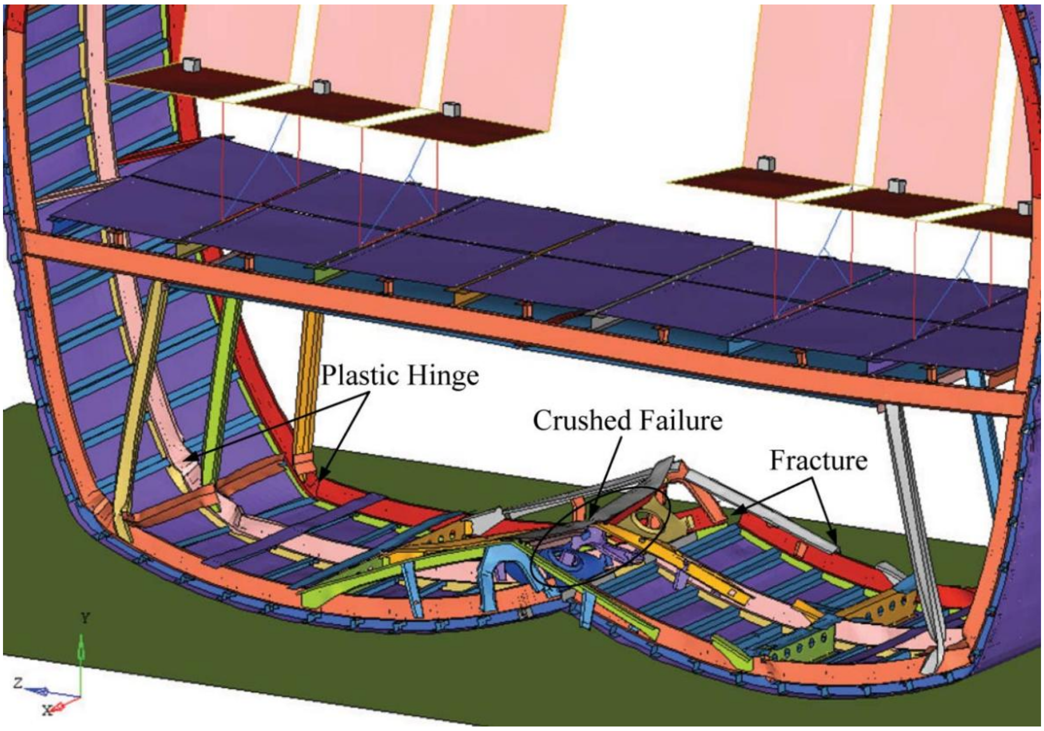

Figure 2.2.4. The failure behaviour of cargo floor structural

2.3. Material and method selection of lower structure

According to the combination of composite material frame and skin, now a brief description of the structure that has the second greatest impact on crashworthiness.I choose a sine-wave beam layout method.The proposed sine-wave beam structure maintains the original aerodynamic shape of the fuselage structure, meets the requirements of aerodynamics, strength and stiffness, and does not affect the use and maintenance of the aircraft. This method can reduce the initial load peak during the impact process and improve the The acceleration curve of the passenger seat track position improves the energy absorption capacity of the fuselage structure and greatly improves the crashworthiness of the fuselage structure.

The sine-wave beam is made of composite materials. Firstly, lift the upper and lower frame edges of the lower part of the fuselage frame to the level, and then install the sine-wave beam between the lower part of the frame and the belly skin by riveting(fig2.3.1). The lift height is equal to the sum of the height of the sine-wave beam and the thickness of the upper support broad(fig2.3.2), the horizontal line length of the upper frame edge is less than the length of the sine-wave beam, the length of the lower frame edge and the sine-wave beam are equal(fig2.3.3).

1-sine-wave beam, 2-frame, 3-stanchion, 4-cargo floor,

5-cargo floor beam, 6-outside frame, 7-upper frame edge, 8-lower frame edge,

9-cargo floor, 10-upper standing broad, 11-belly skin, 12-fuselage centerline,

13-frame axis

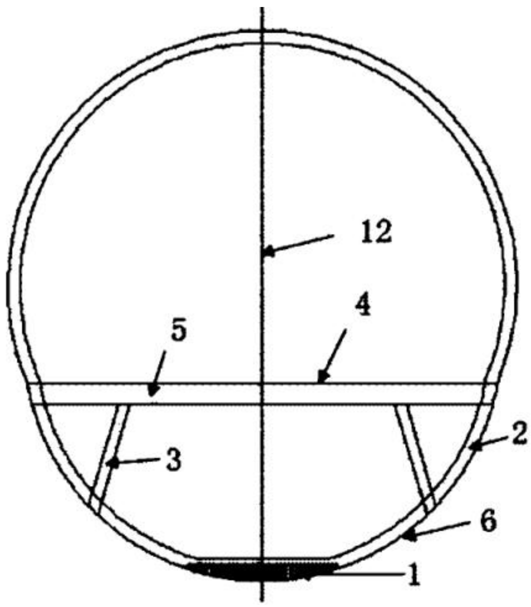

Figure 2.3.1-The fuselage structure section after the sine-wave beam is installed at the bottom

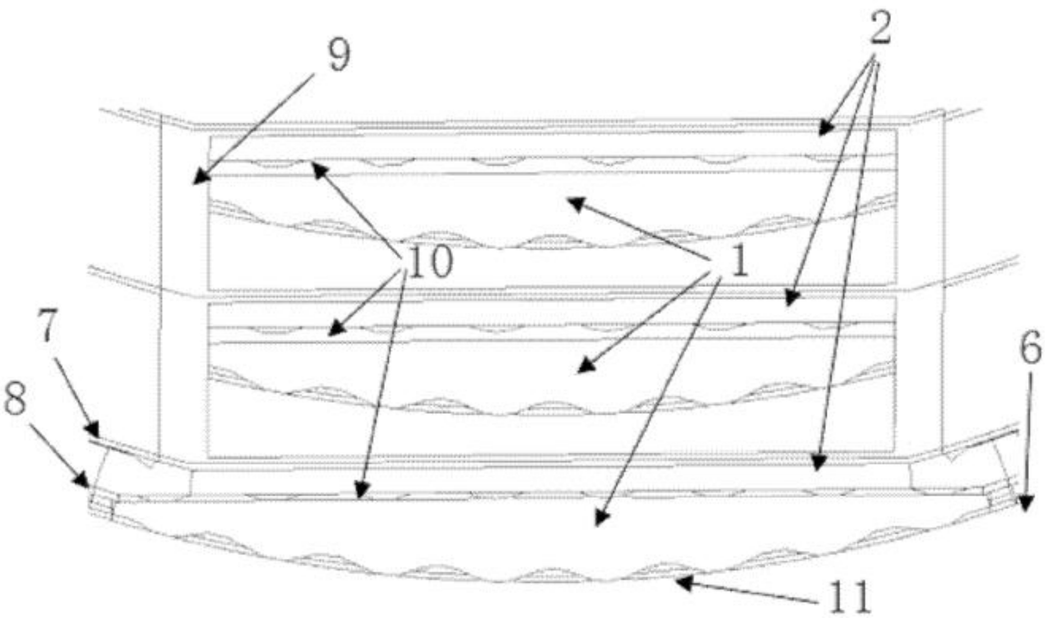

Figure 2.3.2- Bottom sine-wave beam structure

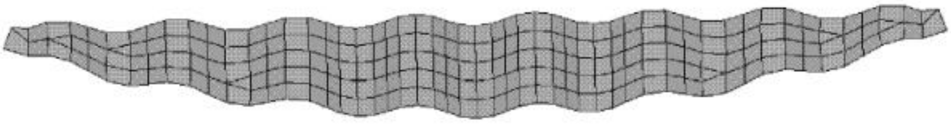

Figure 2.3.3- Structure of sine-wave beam

Conclusion to part 2

The special part of the analysis is the effect of the luggage on the crashworthiness of the fuselage structure, the stiffness of the luggage, the viscosity, and the worst-case scenario that affects the extent to which the luggage absorbs energy. Luggage has a significant impact on the entire fuselage structure when crushed, so we want to choose stronger and stiffer luggage to improve its ability to absorb energy during crush.

GENERAL CONCLUSION

In this design project, I obtain the following results:

-The preliminary design of a medium-range aircraft with a capacity of 156 passengers;

-Cabin layout of medium-range aircraft with 156 passengers;

-Calculation of the center of gravity of the aircraft;

-Calculation of the main geometric parameters of the landing gear;

-Choose wheels brakes that obey the requirements;

-The design of the landing gear;

-The low-wing aircraft uses twin engines and is located under the wings;

-Reasonable layout and convenient service;

-Reasonable optimization of public and personal space;

-Advanced cabin layout and equipment;

-Use turbofan engine to provide maximum thrust for the aircraft.

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Appendix A

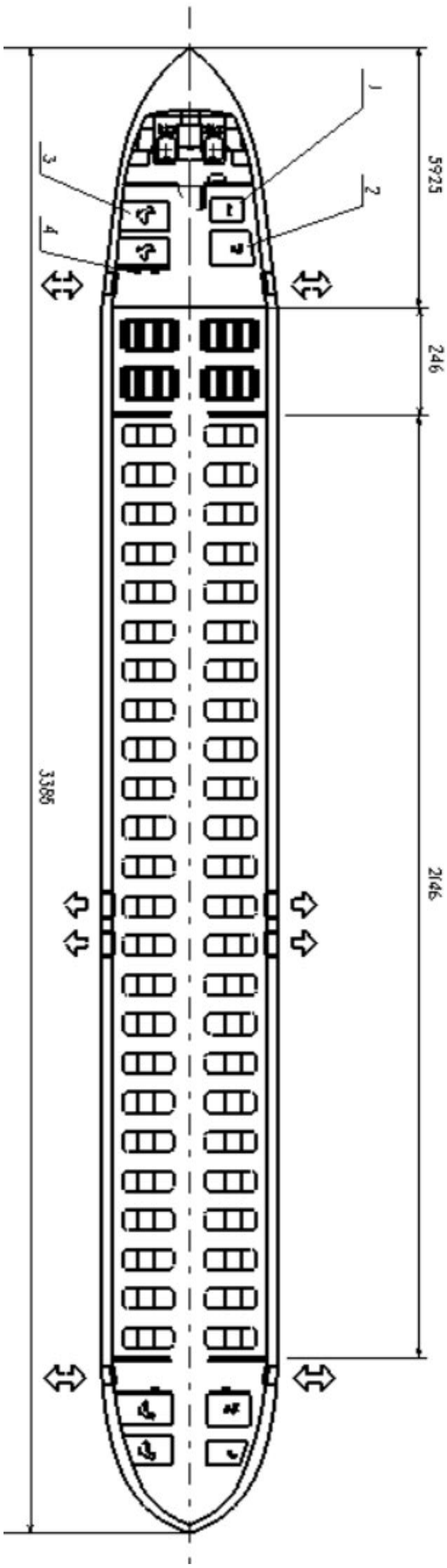
Table of the airplane statistics data

Name of the prototype	**A319-100**	**A320-200**	**A321-100**
Max payload	17700kg	19900kg	25300kg
Flight crew, [persons]	2	2	2
Passengers	124	150	185
Flight range with $G_{payload,max}$, [km]	6800	5700	5600
Range of cruising altitudes, [km]	6945	6112	5926
$V_{cr\ max}$ / H, [km/h / km]	1004.56	1004.56	1004.56
$V_{cr,econ}$ / H, [km/h / km]	955.56	955.56	955.56
Specific fuel consumption [gt/km]			18.2
Power plant data			
Number of engines and their type	2 EA V2500A5	2 IEA V2500A5	2 IEA V2500A5
Take off thrust, [kN]	111.200	111.200	111.200
Airplane mass data			
Maximum Take off Mass, [kg]	75500	78000	93500
Landing Mass, [kg]	62500	66000	77800
Empty weight fraction, %			48100

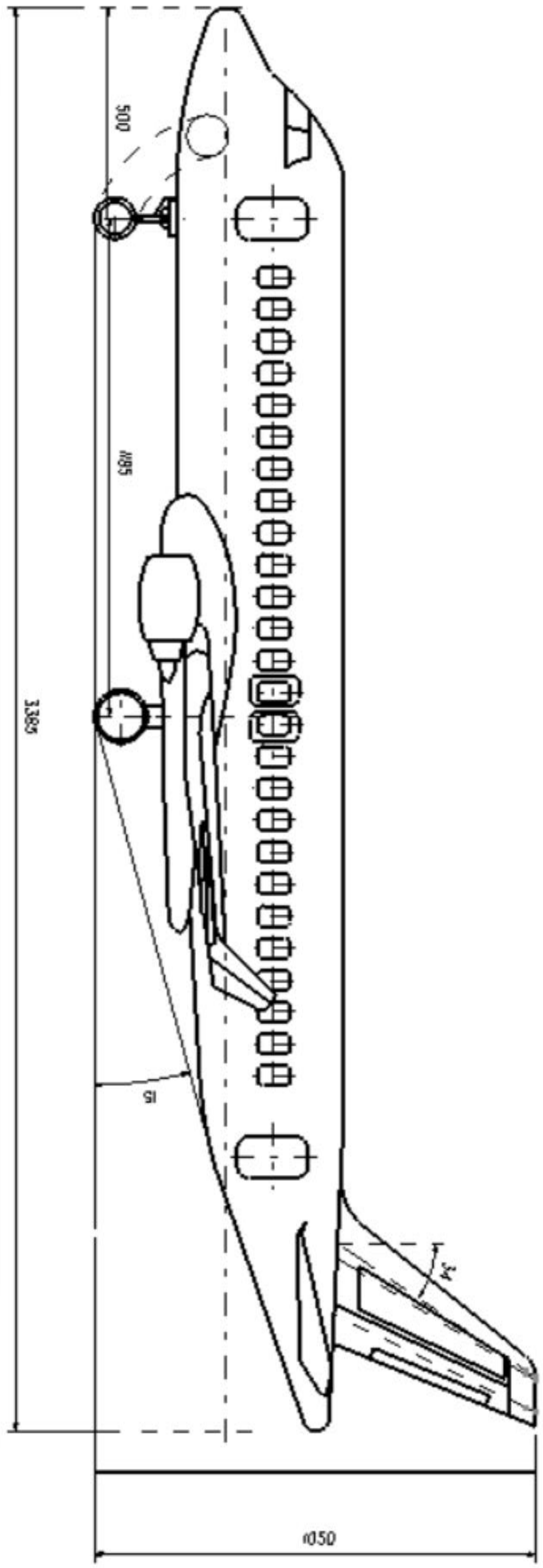
Main geometrical parameters

Name of the prototype	**A319-100**	**A320-200**	**A321-100**
Wing span, [m]	35.8m	35.8m	35.8m
Sweepback angle at ¼ of the chord, [°]	25	25	25
Fuselage length, [m]	33.84m	37.57m	44.51
Fuselage diameter, [m]	3.95m	3.95m	3.95m
Passenger cabin width, [m]	3.7m		3.7m
Passenger cabin length, [m]	23.77m		34.4m
Vertical tail height, [m]		11.755m	
Landing gear base, [m]		7.59m	
Landing gear track, [m]		12.64m	

Appendix B



Appendix C



Appendix D