MINISTRY OF EDUCATION AND SCIENCE OF UKRAINE

National Aerospace University
Kharkiv Aviation Institute

Faculty of Aircraft Engineering

Airplane and Helicopter Design Department

Explanatory Note
to the diploma project

(type of qualification work)

Master (second)

(degree)

on the topic: ***Investigation of the Adaptive Wing Passenger Aircraft and Creation of Its Conception***

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Applicant of 2 Year 161FD Group

Field of knowledge *13 «Mechanical Engineering»*

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Educational program *«Aircraft and Helicopter Design»*

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(surname and initials of student)

Supervisor: Buival L. Y.

Reviewer: Kostenko O. I.

Kharkiv – 2023

Міністерство освіти і науки України
Національний аерокосмічний університет ім. М. Є. Жуковського
«Харківський авіаційний інститут»

Факультет *Літакобудування*

Кафедра *103 «Проектування літаків і вертольотів»*

Рівень вищої освіти *другий (магістерський)*

Галузь знань *13 «Механічна інженерія»*

Спеціальність *134 «Авіаційна та ракетно-космічна техніка»*

(код та найменування)

Освітня програма *«Літаки і вертольоти»*

(найменування)

ЗАТВЕРДЖУЮ
Завідувач кафедри

к.т.н., доц. Андрій ГУМЕННИЙ
"____" ___________202___року

З А В Д А Н Н Я
НА КВАЛІФІКАЦІЙНУ РОБОТ

Цао Їсюань
(прізвище, ім'я та по батькові)

1.Тема кваліфікаційної роботи ***Investigation of the Adaptive Wing Passenger Aircraft and Creation of Its Conception (Дослідження пасажирських літаків із адаптивним крилом і розробка їх концепції створення)***

керівник кваліфікаційної роботи Буйвал Лілія Юріївна, к.т.н.
(прізвище, ім'я, по батькові, науковий ступінь, вчене звання)

затверджені наказом Університету №__________уч від « ______» _____ 2023 року
2.Термін подання студентом кваліфікаційної роботи 18 травня 2023 р.

Вихідні дані до роботи n_{crew} = 2 persons, n_{pass} = 20 persons, V_{CR} = 980 km/h
H_{CR} = 12000 m, $L_{TO.}$ = 1600 m, $L_{(mp=max)}$ = 6600 km

Зміст пояснювальної записки (перелік завдань, які потрібно розв'язати)
ABSTRACT
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1.1. Technical Requirements Specification
1.1.1 Introduction.

1.1.2 Aircraft Assignment.
1.1.3 The basis for development.
1.1.4 Expected Operational Conditions.
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1.2.1 Scientists, who dealt with vehicle and what issues they claim.
1.2.2 Issues, which remained unexplored or not fully covered.
1.2.3 The prototypes of adaptive wing.
1.2.4 Rationale for the choice of the object of scientific research.

1.3. Determining the parameters of an airplane

1.3.1. Aircraft tactical-technical requirements development.
1.3.2. Calculation of aircraft zero approximation take-off mass.
1.3.3. Calculation of structural mass of the main aircraft assemblies, power plant mass, fuel mass, mass of the equipment and control.
1.3.4. Airplane geometrical parameters calculation (wing, fuselage, tail, landing gear).

1.4. Justification of the aircraft scheme. Three-dimensional parametric modelling.

1.5. Integrated design of the wing of the designed aircraft.

2. ECONOMIC SECTION

2.1. Specific advantages of the designed aircraft and evaluation of possible sales in the markets.
2.2. Initial data for the calculation of technical and economic indicators of the efficiency of the aircraft.
2.3. Calculation of the main technical and economic indicators of aircraft design and production.
2.4. Conclusions.

Reference

3. Спеціальне завдання

Не передбачено

Перелік графічного матеріалу (з точним зазначенням обов’язкових креслень)

– майстер-геометрія літака,
– креслення загального вигляду або схема в трьох проекціях (формат А1).

Консультанти розділів кваліфікаційної роботи

Розділ	Прізвище, ініціали та посада консультанта	Підпис, дата	
		завдання видав	завдання прийняв
1. Конструкторський розділ	Буйвал Л. Ю., к.т.н., доцент	**08.02.2023**	**17.05.2023**
2. Економічний розділ	Павленко Т. Ю., к.е.н., доцент	**15.05.2023**	**17.05.2023**

Нормоконтроль ________ Лілія БУЙВАЛ «__18__» _травня___2023_ р.
(підпис) (ім’я та прізвище)

Дата видачі завдання «_08___» ___лютого_____ 2023 р.

КАЛЕНДАРНИЙ ПЛАН

№ п/п	Назва етапів кваліфікаційної роботи	Строк виконання етапів кваліфікаційної роботи	Примітка
1	Конструкторський розділ		
2	Економічний розділ		

Здобувач ____________ **Cao Yixuan**
(підпис) (ім’я та прізвище)

Керівник кваліфікаційної роботи
____________ **Лілія БУЙВАЛ**
(підпис) (ім’я та прізвище)

ABSTRACT

The explanatory note contains: 52 p., 3 Table, 24 Fig., 20 References

Object of research: Adaptive Wing Passenger Aircraft.

The purpose of the work: develop Technical Requirements Specification, to do statistical data analysis of scientists, who dealt with adaptive wing passenger aircraft and what issues they claim, issues, which remained unexplored or not fully covered, the prototypes of adaptive wing and do rationale for the choice of the adaptive wing. Determine the parameters of an airplane, three-dimensional parametric modelling and do integrated design.

Research methods: statistical, analytical methods, Siemens NX software.

The results of the master's diploma project and its novelty:

1) The new medium, long-range business jet adaptive wing aircraft pilot project was carried out according to the developed concept on preliminary design stage has been developed. The aircraft can accommodate 22 persons including two pilots.

2) Current research is based on result of analysis by following: issues, which were claimed by scientists, who dealt with design of adaptive wing aircraft; issues which remained unexplored or not fully covered; adaptive wing structure of known real aircraft (F-111A fighter-bomber, F-14 multirole carrier-borne fighter, F/A-18 Strike Fighter, A-340 passenger family, Gulfstream III).

3) Adaptive wing technology features are considered to be economical, fuel saving, weight reduction, drag reduction, noise reduction, improved efficiency, reduced aircraft manufacturing and operating costs, multi-mission performance, improved flight performance, and improved stealth capability. It has broad application prospect in military and civilian fields. Key technologies include shape optimization design, structural innovation, actuator, aerodynamics, control technology, new engine technology, system integration technology, etc.

4) At present, most of the research on adaptive wing is the trailing edge deformation of the wing. Because the trailing edge of the wing bears relatively small load, it is easier to achieve deformation in structure, and can also improve fuel efficiency, produce better aerodynamic performance.

5) The adaptive trailing edge variable camber wing designed in this paper mainly includes the main wing, ribs, spars, stringers, flexible trailing edge part, upper and lower surface skin. The space distribution model of the wing have built.

6) The evaluating of the economic efficiency of the proposed aircraft project have done. It is expected to produce 50 aircraft a year at a total cost of around 338,555.62 th.dol. The break-even point for the aircraft is about 9. And the aircrafts are mainly used for trade between Ukraine and Beijing of China. Therefore, the aircraft can be sold to large-scale commercial companies to achieve a profit.

ADAPTIVE WING, NOISE REDUCTION, EFFICIENCY, OPTIMAL DESIGN, AERODYNAMIC PERFOMANCE.

Conditions for obtaining: with the written permission of the head of the Department of Aircraft and Helicopter Design of the National Aerospace University «Kharkiv Aviation Institute».

CONTENT

1 DESIGN SECTION

1.1 Technical Requirements Specification

1.1.1 Introduction

The traditional design method of the wing is to optimize the shape of the wing in a certain flight state, and then to design the wing. However, this design method can only ensure the best performance of the wing in this specific flight state, and can not fully play the best performance of the aircraft, which has great limitations. Therefore, for some special purpose aircraft, in order to better reflect its optimal maneuverability, fuel efficiency and economic performance, we can adopt the design concept of adaptive wing.

The adaptive wing can automatically change geometric parameters in flight according to the flight situation to obtain optimal performance, and the wing can be bent to the appropriate position as required.

A new type of adaptive wing aircraft (with adaptive flexible trailing edge technology) can reduce the weight of the wing structure, improve aerodynamic performance, and promote improved fuel economy and operational efficiency.

The adaptive wing can be used in both fighter and civilian airliners. Adaptive wing has rarely been used in civil aircrafts before, but this paper will design an airliner with adaptive trailing edge variable bending wing. Because the general civilian passenger aircraft, because of the need to carry passengers, so the weight of the aircraft may be greater than the fighter. An important role of the adaptive wing is to reduce the weight of the aircraft, so it can be used in civil aircraft to better reflect this advantage. The adaptive wing can automatically change the geometric parameters according to the flight conditions to obtain the optimal performance, which can better ensure the flight safety of passengers and improve the performance of the aircraft.

1.1.2 Aircraft Assignment

This new adaptive wing aircraft is a small aircraft. It is a medium, long-range business jet powered by two high performance turbofan engines. The aircraft can accommodate 22 persons including two pilots. The aircraft is mainly used for business transactions, a company to carry out business transactions overseas personnel generally will not exceed 20 persons, and carry a certain weight of luggage (According to airline regulations, the total weight of hand luggage carried by passengers should not exceed 5 kg, and the volume of each item should not exceed 20 × 40 × 55 cm. Do not allow to carry liquids, weapons, flammable and explosive and other dangerous goods. Its maximum range is 7,000 kilometers.

The main purpose of this aircraft design is to reduce the weight of the fuselage, save fuel, reduce aircraft noise during takeoff and landing, and enable the aircraft to automatically change the geometric parameters in flight according to the flight situation to achieve the best performance, better reflect its optimal maneuverability, fuel efficiency and economic performance.

The straight-line distance between Kiev and Beijing is 6,450 kilometers. In order to strengthen trade between Chinese and Ukrainian companies, so the route chosen to Beijing to Kiev, which is within the maximum range of the new deformable wing aircraft, and a direct route would greatly reduce flight time. It is shown in figure 1.1.

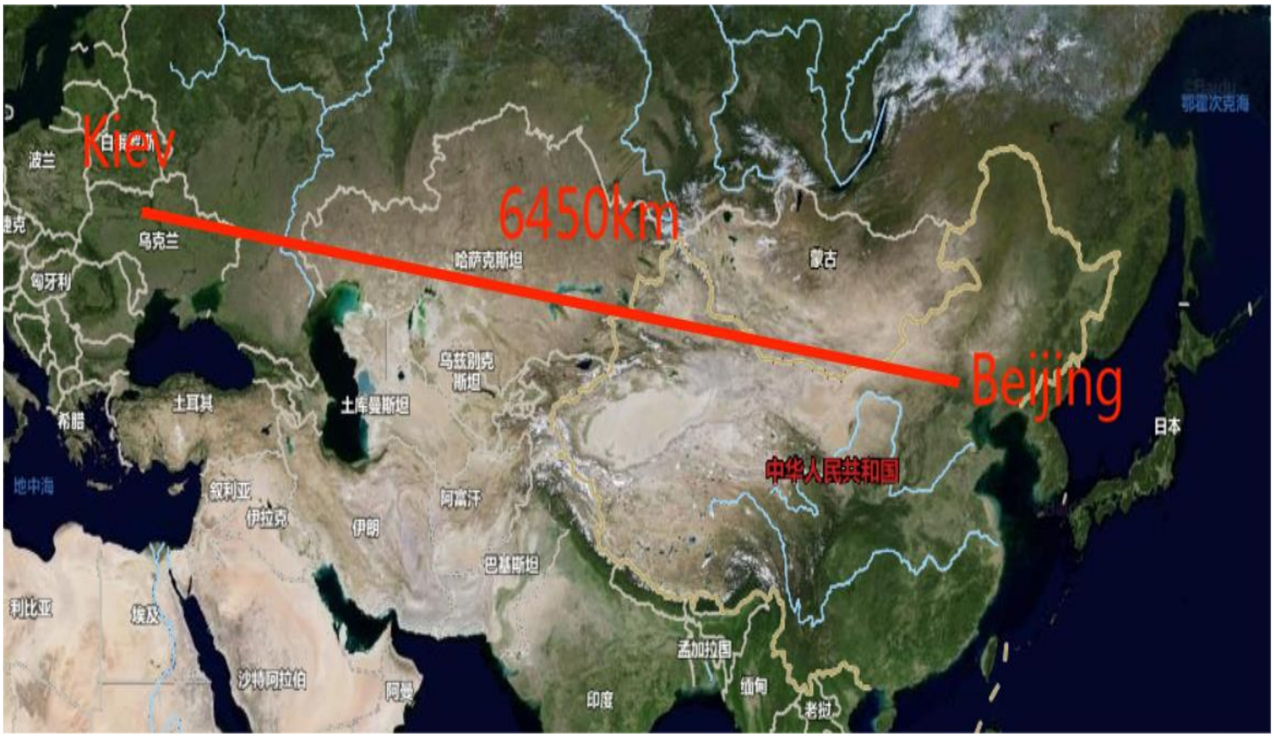


Figure 1.1 – Planned route of this new adaptive wing aircraft

1.1.3 The basis for development

This new adaptive wing aircraft can carry 20 passengers, it complies with Certification Specifications for Normal Category Aeroplanes [1]:

— Certification Specifications for NormalCategory Aeroplanes (CS-23).

— Certification Specifications for NormalCategory Aeroplanes (CS-25).

— Climb requirements: CS 23.2120.

— Trim: CS 23.2140.

— Stability: CS 23.2145 .

— Flight control systems: CS 23.2300.

— Landing gear systems: CS 23.2305.

— Performance: CS 25.101 to CS 25.125.

— Trim: CS 25.161.

— Stability: CS 25.171 to CS 25.181.

— Flight loads: CS 25.321.

— Control systems: CS 25.671 to CS 25.705.

— Landing gear: CS 25.721 to CS 25.745.

1.1.4 Expected Operational Conditions

— Environmental characteristics and factors:

a) Barometric pressure - for all range of flight altitudes in accordance with requirements; on ground ΔPH = +20 mm HG.

b) Temperature of ambient air (tAA) represents the temperature

change in dependence of altitude in accordance with state standard. Temperature deviation from mean value for different altitudes along the following lines:

"Minimum arctical".

"Maximum tropical".

In this case the aircraft and its systems operability shall maintain within the mentioned range after aircraft stay at ground at tAA down to minus 60°C.

c) Relative humidity on ground is of 98% at an ambient air temperature of +35° C.

d) Air mass density shall correspond to ranges of ambient air temperature and barometric pressure according to international standard atmosphere.

e) Maximum wind components at takeoff and landing on dry paved RWY:

lateral component – 12 m/s (friction coefficient $\mu \geq 0.6$);

following component – 5 m/s;

headwind – 30 m/s.

Note: under the highest unfavorable RWY (runway) conditions ($\mu \geq 0.6$), reduction of lateral wind component limiting values is allowable.

— Operational factors:

1) Crew: pilot, co-pilot.

2) RWY type – A road with no obstacles and a width of not less than 18 meters.

3) Aerodrome class according to civil aviation classification – B (L RWY = 1800 m, width – 30 m), unpaved RWY with soil strength at least of 8 kg/cm².

4) Aerodrome elevation above sea level is from minus 300 m to 3000 m.

5) Condition of paved RWY:

dry;

moist;

wet;

water areas;

occupied by water;

covered with slush, wet snow of up to 1.5 cm thickness;

snow-covered (depth of new-fallen snow up to 8 cm).

6) Characteristics of unpaved RWY:

without turf;

loamy, clay;

sandy, sandy-loam;

rocky, crushed-stone;

chernozem soil.

7) Condition of paved RWY:

dry;

snow-covered (depth of new-fallen snow up to 8 cm).

8) Aircraft operation features.

The aircraft is intended to perform the flights:

according to visual and instrument flight rules;

under simple and difficult weather conditions, under icing;

in daytime and in nightime;

above plain and mountain surface of local airlines;

above water distanced from ground for up to 30 min. of flight;

range of geographical latitudes - up to 70° north and 55° south latitude.

9) Operational meteorological minimums for takeoff and landing:

— minimum for takeoff: visibility range on RWY of at least 300 m;

— minimum for landing: I category (decision altitude 60 m at visibility range on RWY of at least 800 m).

10) Components and characteristics of ground facilities for flight provisions.

11) Maintenance periodicity is equal to 300 flight hours of airframe.

12) Service life and life time for:

a) aircraft (up to discarding):

landings — 50000

flight hours — 40000

service life — 25 years.

b) engine:

assigned service life, h—20000

life to first overhaul and life between overhauls—6000

c) vendor items – as a rule the vendor items service life shall correspond to aircraft service life or their shall be multiple of repair periodicity.

13) The take-off speeds , the accelerate-stop distance , the take-off path, the take-off distance and take-off run, and the net take-off flight path, must bedetermined in the selected configuration for take-off at each weight, altitude, and ambient temperature within the operational limits selected by the applicant -

(a)In non-icing conditions; and

(b)In icing conditions:

(i) The stall speed at maximum take-off weight exceeds that in non-icing conditions by more than the greater of 5.6 km/h (3 knots) CAS or 3% of VSR; or

(ii) The degradation of the gradient of climb is greater than one-half of the applicable actual-to-net take-off flight path gradient reduction

14) No take-off made to determine the data required by this paragraph may require exceptional piloting skill or alertness.

15) The take-off data must be based on:

(a) Smooth, dry and wet, hard-surfaced runways; and

(b) At the option of the applicant, grooved or porous friction course wet, hardsurfaced runways.

16) The take-off data must include, within the established operational limits of the aeroplane, the following operational correction factors:

a) Not more than 50% of nominal wind components along the take-off path opposite to the direction of take-off, and not less than 150% of nominal wind components along the take-off path in the direction of take-off.

b) Effective runway gradients.

The expected take-off distance is 1600 meters, the landing distance is 750 meters, the maximum range is 7000 kilometers. And this adaptive wing aircraft can fly in some severe weather conditions, such as thunderstorms, snowy days and windy weather. The adaptive wing can automatically change geometric parameters in flight according to the flight situation to obtain optimal performance, and the wing can be bent to the appropriate position as required. There are no restrictions on flight time, and it can fly at night as well as during the day.

1.1.5 General Requirements

The expected maximum flight altitude is 13000 meters, and the maximum flight speed is 1029 kilometers per hour. The original design was for a light aircraft with six passengers. One advantage of the adaptive wing is that it can reduce the structural weight of the aircraft, which can not be well reflected in light aircraft, so this paper designed the number of passengers to be 20.

The following compares the design requirements of the two types of aircraft.

CS-23:

1.Climb requirements

(a) The design must comply with the following minimum climb performance out of ground effect: with all engines operating and in the initial climb configuration(s): for Level-1 and -2 high-speed aeroplanes and all Level-3 and -4 aeroplanes, a climb gradient at take-off of 4 %.

(b) after a critical loss of thrust on multi-engine aeroplanes: for Level-1 and -2 high-speed aeroplanes, a 1 % climb gradient at 122 m (400 ft) above the take-off surface with the landing gear retracted and flaps in the take-off configuration;

(c) a climb gradient of 3 % during balked landing, without creating undue pilot workload, with the landing gear extended and flaps in the landing configuration(s).

2.Trim

(a) The aeroplane must maintain lateral and directional trim without further force upon, or movement of, the primary flight controls or corresponding trim controls by the pilot, or the flight control system, under the following conditions: for Level-1, -2, and -3 aeroplanes, in cruise.

(b) The aeroplane must maintain longitudinal trim without further force upon, or movement of, the primary flight controls or corresponding trim controls by the pilot, or the flight control system, under the following conditions:

(1) climb,

(2) level flight,

(3) descent,

(4) approach.

(c) Residual control forces must not fatigue or distract the pilot during normal operations of the aeroplane and likely abnormal or emergency operations, including a critical loss of thrust on multi-engine aeroplanes.

3. Stability

No aeroplane may exhibit any divergent longitudinal stability characteristic so unstable as to increase the pilot’s workload or otherwise endanger the aeroplane and its occupants.

4. Flight control systems

(a) The flight control systems are designed to:

(1) operate easily, smoothly, and positively enough to allow proper performance of their functions;

(2) protect against likely hazards.

(b) Trim systems, if installed, are designed to:

(1) protect against inadvertent, incorrect, or abrupt trim operation;

(2) provide information that is required for safe operation.

5. Landing gear systems

(a) The landing gear is designed to:

(1) provide stable support and control to the aeroplane during surface operation;

(2) account for likely system failures and likely operation environment (including anticipated limitation exceedances and emergency procedures).

(b) Aeroplanes must have a reliable means of stopping the aeroplane with sufficient kinetic energy absorption to account for landing. Aeroplanes that are required to demonstrate aborted take-off capability must account for this additional kinetic energy.

(c) For aeroplanes that have a system that actuates the landing gear, there is:

(1) a positive means to keep the landing gear in the landing position; and

(2) an alternative means available to bring the landing gear in the landing position when a nondeployed system position would be a hazard.

CS-25:

1. Performance

(a) Unless otherwise prescribed, aeroplanes must meet the applicable performance requirements of this Subpart for ambient atmospheric conditions and still air.

(b) The performance, as affected by engine power or thrust, must be based on the following relative humidities:

(1) 80%, at and below standard temperatures; and

(2) 34%, at and above standard temperatures plus 28°C (50°F).

Between these two temperatures, the relative humidity must vary linearly.

(c) The performance must correspond to the propulsive thrust available under the particularambient atmospheric conditions, the particular flight condition, and the relative humidity specified. The available propulsive thrust must correspond to engine power or thrust, not exceeding the approved power or thrust, less –

(1) Installation losses;

(2) The power or equivalent thrust absorbed by the accessories and services appropriate to the particular ambient atmospheric conditions and the particular flight condition.

(d) Unless otherwise prescribed, the applicant must select the take-off, en-route, approach, and landing configuration for the aeroplane.

(e) The aeroplane configurations may vary with weight, altitude, and temperature;

(f) Unless otherwise prescribed, in determining the accelerate-stop distances, take-off flight paths, take-off distances, and landing distances, changes in the aeroplane's configuration, speed, power, and thrust, must be made in accordance with procedures established by the applicant for operation in service;

(g) Procedures for the execution of balked landings and missed approaches must meet the prescribed conditions;

(h) Be able to be consistently executed in service by crews of average skill; Use methods or devices that are safe and reliable; Include allowance for any time delays in the execution of the procedures, that mayreasonably be expected in service;

(i) The accelerate-stop and landing distances prescribed must be determined with all the aeroplane wheel brake assemblies at the fully worn limit of their allowable wear range.

2. Trim

(a) Each aeroplane must meet the trim requirements of this paragraph after being trimmed, and without further pressure upon, or movement of, either the primary controls or their corresponding trim controls by the pilot or the automatic pilot.

(b) Lateral and directional trim. The aeroplane must maintain lateral and directional trim with the most adverse lateral displacement of the centre of gravity within the relevant operating limitations, during normally expected conditions of operation.

(c) Longitudinal trim. The aeroplane must maintain longitudinal trim during –

(1) A climb with maximum continuous power at a speed not more than 1·3 V_{SR1}, with the landing gear retracted, and the wing-flaps (i) retracted and (ii) in the take-off position;

(2) Either a glide with power off at a speed not more than 1·3 V_{SR1}, or an approach within the normal range of approach speeds appropriate to the weight and configuration with power settings corresponding to a 3° glidepath, whichever is the most severe, with the landing gear extended, the wing-flaps retracted and extended, and with the most unfavourable combination of centre of gravity position and weight approved for landing;

(3) Level flight at any speed from 1·3 V_{SR1}, to V_{MO}/M_{MO} , with the landing gear and wing-flaps retracted, and from 1·3 V_{SR1} to V_{LE} with the landing gear extended.

(d) Longitudinal, directional, and lateral trim. The aeroplane must maintain longitudinal, directional, and lateral trim (and for lateral trim, the angle of bank may not exceed 5°) at 1·3 V_{SR1} , during the climbing flight with –

(1) The critical engine inoperative;

(2) The remaining engines at maximum continuous power;

(3) The landing gear and wing-flaps retracted.

3. Stability

The aeroplane must be longitudinally, directionally and laterally stable in accordance with the provisions. In addition, suitable stability and control feel (static stability) is required in any condition normally encountered in service, if flight tests show it is necessary for safe operation.

4. Flight loads

(a) Flight load factors represent the ratio of the aerodynamic force component (acting normal to the assumed longitudinal axis of the aeroplane) to the weight of the aeroplane. A positive load factor is one in which the aerodynamic force acts upward with respect to the aeroplane.

(b) Considering compressibility effects at each speed, compliance with the flight load requirements of this Subpart must be shown –

(1) At each critical altitude within the range of altitudes selected by the applicant;

(2) At each weight from the design minimum weight to the design maximum weight appropriate to each particular flight load condition; and

(3) For each required altitude and weight, for any practicable distribution of disposable load within the operating limitations recorded in the Aeroplane Flight Manual.

(c) Enough points on and within the boundaries of the design envelope must be investigated to ensure that the maximum load for each part of the aeroplane structure is obtained.

(d) The significant forces acting on the aeroplane must be placed in equilibrium in a rational or conservative manner. The linear inertia forces must be considered in equilibrium with the thrust and all aerodynamic loads, while the angular (pitching) inertia forces must be considered in equilibrium with thrust and all aerodynamic moments, including moments due to loads on components such as tail surfaces and nacelles. Critical thrust values in the range from zero to maximum continuous thrust must be considered.

5. Control systems

(a) Each flight control system must operate with the ease, smoothness, and positiveness appropriate to its function. In addition, the flight control system shall be designed to continue to operate, respond appropriately to commands, and must not hinder aeroplane recovery, when the aeroplane is in any attitude or experiencing any flight dynamics parameter that could occur due to operating or environmental conditions.

(b) Each element of each flight control system must be designed to minimise the probability of incorrect assembly that could result in the failure or malfunctioning of the system. Distinctive and permanent marking may be used where design means are impractical, taking into consideration the potential consequence of incorrect assembly.

(c) The aeroplane must be shown by analysis, test, or both, to be capable of continued safe flight and landing after any of the following failures or jams in the flight control system within the normal flight envelope. In addition, it must be shown that the pilot can readily counteract the effects of any probable failure. The jam must be evaluated as follows:

(i) The jam must be considered at any normally encountered position of the control surface, or pilot controls;

(ii) The jam must be assumed to occur anywhere within the normal flight envelope and during any flight phase from take-off to landing; and In the presence of a jam considered under this sub-paragraph, any additional failure conditions that could prevent continued safe flight and landing shall have a combined probability of 1/1 000 or less.

(d) The aeroplane must be designed so that, if all engines fail at any time of the flight:

(1) it is controllable in flight;

(2) an approach can be made;

(3) a flare to a landing, and a flare to a ditching can be achieved; and

(4) during the ground phase, the aeroplane can be stopped.

(e) The aeroplane must be designed to indicate to the flight crew whenever the primary control means is near the limit of control authority.

(f) If the flight control system has multiple modes of operation, appropriate flight crew alerting must be provided whenever the aeroplane enters any mode that significantly changes or degrades the normal handling or operational characteristics of the aeroplane.

6. Landing gear

(a) The landing gear system must be designed so that when it fails due to overloads during take-off and landing, the failure mode is not likely to cause spillage of enough fuel to constitute a fire hazard. The overloads must be assumed to act in the upward and aft directions in combination with side loads acting inboard and outboard. In the absence of a more rational analysis, the side loads must be assumed to be up to 20% of the vertical load or 20% of the drag load, whichever is greater.

(b) The aeroplane must be designed to avoid any rupture leading to the spillage of enough fuel to constitute a fire hazard as a result of a wheels-up landing on a paved runway, under the following minor crash landing conditions:

(1) Impact at 1.52 m/s (5 fps) vertical velocity, with the aeroplane under control,

at Maximum Design Landing Weight,

(i) with the landing gear fully retracted and, as separate conditions,

(ii) with any other combination of landing gear legs not extended.

(2) Sliding on the ground, with -

(i) the landing gear fully retracted and with up to a 20° yaw angle and, as separate conditions,

(ii) any other combination of landing gear legs not extended and with 0° yaw angle.

(c) For configurations where the engine nacelle is likely to come into contact with the ground, the engine pylon or engine mounting must be designed so that when it fails due to overloads (assuming the overloads to act predominantly in the upward direction and separately predominantly in the aft direction), the failure mode is not likely to cause the spillage of enough fuel to constitute a fire hazard.

Choose the general requirementes of CS-25.

1.2 Statistical data analysis

1.2.1 Scientists, who dealt with vehicle and what issues they claim

From the history of the aircraft industry, the idea of adaptive wings was developed by the Wright brothers through their concept of wing warping. Today, the notion that an aircraft can deform its geometry in flight without losing its surface integrity often gains tangible expression. S. Barbarino, O. Bilgen, R.M. Ajaj, M.I. Friswell and D.J. Inman aimed to research and develop structures and components of the adaptive part of the wing [2], and R. Pecora, F. Amoroso and L. Lecce studied the problem of adaptive (changing) aircraft wing geometry according to non-static (changing in flight) criteria [3]. These standards are designed to ensure optimal aerodynamic characteristics while maintaining maximum lift coefficient. One possible approach is to solve this problem by using elements that smoothly change the geometry, which allows the aircraft to adapt the geometry of its wings according to flight patterns. As a result, over the past few years, as aircraft technology and materials have developed, experts have increasingly focused on the potential to change the shape of an aircraft's wing based on flight patterns without losing wing surface integrity. In other words, they want to use adaptive wings.

New aircraft structures and technologies such as adaptive deformed trailing edges offer the possibility of improving the fuel efficiency of commercial transport aircraft. In order to accurately quantify the benefits of deformable wing technology for commercial transport aircraft, high fidelity design optimization of aerodynamic and structural design with a large number of design variables needs to be considered.

To meet this requirement, David A. Burdette and Joaquim R.R.A. Martins used A high-fidelity aerodynamic structure that allows detailed optimization of wing shape and size using hundreds of design variables. They performed a number of multi-point aerodynamic structural optimizations to demonstrate the performance advantages afforded by the deformation technique and to determine how these advantages were achieved. In an optimization comparison considering seven flight conditions, adding a deformed trailing edge device to 40% of the tail of the wing reduced cruise fuel consumption by more than 5%. Fuel consumption is largely reduced due to trailing edge deformations due to reduced adaptive maneuvering loads which significantly reduce structural weight. They also show that smaller deformations at 30% of the tail of the wing produce nearly the same reduction in fuel consumption as larger deformations, and that the deformations technique is particularly effective for high aspect ratio wings [4].

Another technique is adaptive deformation trailing edge, also known as adaptive trailing edge, or simply adaptive trailing edge. Companies such as FlexSys have developed such devices, which have been flight-tested in partnership with NASA and the US Air Force Research Laboratory. This technology offers the potential to create wings that can actively adapt to flight conditions, allowing engineers to design wing shapes and sizes with more robust performance relative to flight conditions. Another variation of this technique is the variable radian continuous trailing edge flap, which changes radian using three parts, rotating rigidity [4].

Various studies have examined deformation mechanisms and explored and reviewed the benefits of wing designs that apply this technology. In the late 1990s, Hanselka and Monner et al. outlined the aerodynamic benefits associated with deformable trailing edge devices and provided designs for deformable mechanisms. More recently, Molinari et al. explored the potential of this technique using a multidisciplinary optimization approach that considers mission, aerodynamics, materials, and structural disciplines. The work used a low-fidelity model, so it was unable to capture the effects of tiny shape changes that have proven critical to transonic aerodynamic performance. Reynolds-averaged Navier–Stokes (RANS) simulations are required to capture the viscous and compressibility effects in that flight regime. Lyu and Martins used RANS-based aerodynamic shape optimizations to design a wing with a morphing trailing edge showing drag reductions between one and five percent, depending on the flight condition. Wakayama et al. found similar results in their work considering morphing devices on three different aircraft configurations. Other studies have also considered dynamic aeroelastic constraints. Multidisciplinary design optimization (MDO) provides the computational approach to make the most of these technologies [4].

David A. Burdette and Joaquim R.R.A. Martins previously evaluated wings with deformed trailing edges at single track points using MDO coupled hi-Fi aerodynamics and structural models, showing that deformed trailing edges greatly affect wing lift distribution in the longitudinal direction. In the work presented here, they attempt to build on these previous results by expanding the number of flight conditions that consider wing performance. The main advantage of the morphing trailing edge technique is its ability to adapt the wing to changing flight conditions, so they hope that multi-point analysis will provide a better chance of morphing trailing edges to improve performance. Structural analysis is also important because structural deflection will vary under different flight conditions, providing an opportunity to deform trailing edges to further improve performance. They assume that an ideal deformable mechanism can achieve the specified shape, in which the weight of the mechanism is comparable to that of a conventional control actuator. While not a realistic assumption, these studies provide an upper bound on the benefits of deformation techniques and open the door to more detailed research using high-fidelity aircraft structural design optimization.

1.2.2 Issues, which remained unexplored or not fully covered

From the current research and technology development trend, "adaptive deformation" is an inevitable technological approach to improve the performance of future aircraft. The changes in the airfoil structure, skin material, aerodynamic characteristics, measurement and control system of adaptive deformable aircraft make the mature aircraft materials, structure, mechanics and control face new great challenges. It must have enough stiffness and strength to withstand the aerodynamic load of the aircraft in the course of flight. At the same time, it must have high elasticity to achieve large deformation to meet the stretching or compression deformation requirements generated when the front and rear edges of the wing are deflected. The skin must have 4% ~ 6% or even greater deformation ability to meet the control requirements of the wing. At present, there is no good material or structure can fully meet the demand of wing skin [5].

The difficulties in the structural design of variant aircraft are as follows: how to realize the variant wing internal structure and skin structure with large deformation and high load-bearing capacity; The problem of determining how a variant wing will deform under different conditions [6].

More comprehensive future studies should include the weight of the morphing mechanism, which would likely shift the balance between aerodynamic performance and structural weight. Mechanism sizing would additionally require implementation of the associated constraints, like power requirements from mission and control system optimization, along with the necessary adjoint derivative calculations. Future studies may also consider the feasibility of the lighter structures optimized herein with respect to flight conditions [4].

Therefore, the future research goal is to achieve topology optimization of reconfigurable cellular cores, and select the optimal combination of cellular forms for each instance. At the same time, it is necessary to further find out the solution of manufacturing the adaptive flexible wing skin, analyze the adaptive wing optimization design method, optimize the wing shape and size in detail, and study the wing trailing edge deformation related problems.

1.2.3 The prototypes of adaptive wing

General description of F-111A

The F-111A fighter-bomber (Figure 1.2), designed by General Dynamics in 1962, is the world's first operational variant of the swept-wing aircraft, designed primarily to perform interruption and nuclear attack missions at night and in complex weather conditions.

The F-111A has a crew of 2 (pilot and weapon system operator), a length of 22.4 meters, a full spread wingspan of 19.2 meters, a full sweep of 9.75 meters, a height of 5.22 meters, a wing area of 61.07 square meters, a full sweep of 48.77 square meters, and an empty weight of 21,400 kilograms. It has a maximum take-off weight of 45,300 kg, two TF30-P-100 after power engines, a maximum flight speed of Mach 2.5 (2,655 kilometers per hour), a service ceiling of 20,100 meters, a range of 6,760 kilometers, a combat radius of 2,140 kilometers, and a climb rate of 131.5 meters per second. The wing load is 741 kg/m^2 and the thrust-weight ratio is 0.5 The three views are shown in figure 1.3 [7].

The F-111A uses an overall layout of two seats, twin engines, upper single wing, and inverted T-tail, with a front three-point landing gear. The most prominent feature is the use of variable swept wings, which is the first time this technology has been applied to a practical aircraft.

Features of wing structure

The F-111A wing is a variable swept wing, cantilevered single wing with no reverse Angle. The airfoil is the NACA64A series with conventional wing negative torsion (washed out, reduced Angle of attack towards the wing tip). There are 5 wing spars, and the integral panel skin is between front and back edges, from wing root to wing tip, and honeycomb sandwich is adopted at front and back edges (filling material). The movable segment has full wingspan leading edge slats and trailing edge double slats flaps. The advantage of variable swept-back wing is that it can improve the take-off and landing performance of supersonic aircraft, take into account the aerodynamic requirements between high and low speed, and expand the use range of aircraft. The F-111A has a wing sweep range of 16-72.5'. 16' for take-off, 26' for landing and subsonic cruise, and appropriate angles below 72'5' can be selected for high and low altitude supersonic flight. The two movable wings are connected by a 4.2-meter-long box beam, and their sweep Angle is controlled by the pilot through a hydraulic system.

Problems of the fighter jet F-111 design, which was caused by adaptive wing investigation

The F-111 continued to improve in production use. As F-111 is the first practical aircraft to adopt the variable swept wing technology, it is inevitable that it has some immaturity. The aircraft performance cannot reach the predetermined target due to the overweight structure, and the plane was destroyed due to the fracture of the lower plate of the wing rotating shaft joint in flight. The engine afterburner flares out and the inlet throttling causes engine surge. These problems were gradually solved in the later development of the model.

Figure 1.2 — The fighter jet F-111

Figure 1.3 — Three views of the F-111

General description of F-14

The F-14 is an American supersonic multirole carrier-borne fighter (in Figure 1.4). The F-14 is a dual-seat, dual-engine, dual-vertical tail, swept-back, medium-wing layout. It was developed according to the requirements of the United States Navy for fleet air defense and escort in the 1970s and 1980s. It is the third-generation fighter aircraft in the generation, mainly performing fleet defense, interception, strike and reconnaissance missions. The F-14 was developed by Grumman (now Northrop Grumman) in late 1967, with the first flight of the prototype on 21 December 1970 and delivery in May 1972. In 1987, the F-14B went into production with an improved engine to replace the Navy's F-4 Phantoms. In 1988, the F-14 underwent further improvements and upgrades in radar, avionics, and missile mounting capabilities, and was designated the F-14D Super Tomcat.

The F-14 has a crew of 2 (pilot and radar interception officer), a length of 19.10 meters, a wingspan of 19.54 meters (sweep Angle 20°), 11.65 meters (sweep Angle 68°), 10.15 meters (sweep Angle 75°), a height of 4.88 meters, a wing area of 52.46 square meters, and an empty weight of 18,191 kilograms. It has a maximum take-off weight of 33,724 kg and two F110 afterpower turbofan engines. It has a maximum flight speed of Mach 2.38, a service ceiling of 18,290 m, a range of 2,573 km, a climb rate of 152 m/s, a wing load of 643 kg/m2 and a thrust-to-weight ratio of 0.75 [8].

Features of wing structure

The F-14, with its variable swept-back wing, has a complex box-shaped structure on its back called the wing box. In order to make the wing box as light as possible without affecting strength, Grumen is made of titanium alloy with high strength and light weight. Since titanium alloy cannot be welded by conventional methods, vacuum electron beam welding technology is also developed for this purpose. In addition to bearing power, the wing box also forms an integral fuel tank. Composite materials are used on the Tomcat's radome and belly skin, and for the first time, boron fiber/epoxy composite materials are used on the horizontal tail structure for greater fatigue resistance.

Problems

Increased fuselage weight, reduced wing suspension points, reduced load, reduced flexibility. It increases the complexity of the mechanism and the number of firmware, reduces reliability geometrically, and increases production complexity and maintenance costs geometrically, resulting in redundant returns. And even the sacrifice of so many aspects can not compensate for its structural strength reduction.

Figure 1.4 — The supersonic multirole carrier-borne fighter F-14

General description of F/A-18 Strike Fighter

F/A-18 Strike Fighter is A multi-role carrier-based fighter aircraft with a single-seat/tandem two-seat swept-wing aerodynamic layout and two turbofan engines (in Figure 1.6). It is the U.S. military's first fighter and attack aircraft with excellent air, ground and sea attack capabilities. As the U.S. Navy's most important carrier-based aircraft, it is versatile, both for sea air defense and ground attack. The F/A-18 was developed by McDonnell Douglas (now Boeing). The prototype first flew on 9 June 1974 and entered service with the U.S. Navy in 1983. After the F-14 fighter was retired on 28 July 2006, the aircraft continued to serve as carrier-based fighters aboard U.S. aircraft carriers. Figure 1.5 shows the general layout of the F/A-18 fighter attack aircraft.

The F/A-18C has A crew of 1 person, a length of 17.07 meters, a wingspan of 11.43 meters, a height of 4.66 meters, a wing area of 37.16 square meters, an empty weight of 10,455 kilograms, a maximum take-off weight of 22,328 kilograms, and a power system of 2 F404 afterpower turbofans with a maximum thrust of 65.3 kN each. The afterthrust is 98.9 kN per aircraft, the internal fuel carrying capacity is 4926 kg, the external fuel carrying capacity is 3053 kg, the maximum flight speed is Mach 1.8 (1814 km/h), the practical ceiling is 15,000 m, the range is 2,346 km, the combat radius is 722 km, and the climb rate is 254 m/s. The wing load is 459 kg/m2 and the thrust-weight ratio is 0.67 [9].

Features of wing structure

The F/A-18 fighter-attack aircraft has an overall layout of double swept wings and double vertical tail, with A wing area of 37.16 square meters to improve low-speed performance. The wing is single wing in cantilever, with low sweep Angle, full wingspan maneuvering flaps on leading edge, hydraulic action flaps and ailerons on inside trailing edge, and deflection of front and rear edge flaps is controlled by computer. Automatically changes wing camber to achieve the best lift-drag ratio across the performance envelope. The ailerons on the outside of the trailing edge can be used as flaperons to further enhance low-speed handling, and the flaps and ailerons can also be differential used for roll control. When landing on the ship, the outer wing segment can be folded (ailerons are at the trailing edge of the outer wing) and the hinge is at the junction of ailerons and flaps. The leading edge of the wing root is a pair of large edges that extend forward to either side of the cabin, thus enabling the aircraft to fly at an Angle of attack of 60 degrees.

Problems

The manufacturing process is complicated because of the large area of the strip wing.

Figure 1.5 — General layout breakdown of F/A-18 combat attack aircraft

Figure 1.6 — F/A-18 combat attack aircraft

General description of A-340 family

The Airbus A340 is a four-engine, long-range twin-aisle wide-body airliner built by the European company Airbus. The basic design is similar to the twin-engine Airbus A330, but with two more engines for a total of four. The A340 was originally planned to be powered by Super Fan Engines developed by IAE (International Aero Engines). IAE later stopped development and the Airbus A340 switched to the CFM56-5C engine. In 1991, after the A340 made its first flight, engineers discovered a potentially important flaw: the wing was not strong enough to bend and wobble when cruising at high speeds with outboard engines. The solution was to add a tortoiseshell-shaped bulge under the wing to improve airflow near the engine hanger. The modified A340 entered service in 1993. The three views of the Airbus A340-200 are shown in figure 1.7.

The A340-200 has 2 pilots, 240 seats, 59.39 meters in length, 60.30 meters in wingspan, 30° in wing sweep Angle, 16.71 meters in height, 5.28 meters in cabin width, 129000 kg in empty weight, 275,000 kg in maximum take-off weight, and a cruising speed of 896km/h. Take-off runway length is 2990 m, full range is 14800 km, service limit is 11887 m, engine (x4) is CFM56-5C2(138.78 kN), CFM56-5C3(144.57 kN), CFM56-5C4(151.25 kN) [10].

Features of structure

The aircraft as a whole adopts a basic structure made of some composite materials. The A340's wings are computer-controlled, with the computer manipulating the trailing edge flaps to get the best airfoil shape based on the altitude and speed at which it is flying, as well as the load it is carrying. The auto-camber airfoil improves aerodynamic efficiency, reduces drag, and reduces structural weight by relieving the load on the wing. Wingtips are equipped with wingtip winglets. The A340's lift-drag ratio is 40% higher than the A300's. The fuselage and tail fins are made of a large number of aluminum-lithium alloy and composite materials. The tail fins, each control surface, the recirculating envelope and the cabin floor are all made of composite materials. The A340 has a four-wheel trolley main landing gear and a two-wheel nose landing gear. The A340-200/300 adds a two-wheel auxiliary landing gear in the mid-fuselage midline position. The A340-500/600 adds a four-wheel trolley landing gear in the mid-fuselage position.

Problems

Due to rising fuel prices, dual engines are better than the four-engine A340 in terms of both operating costs and economy.

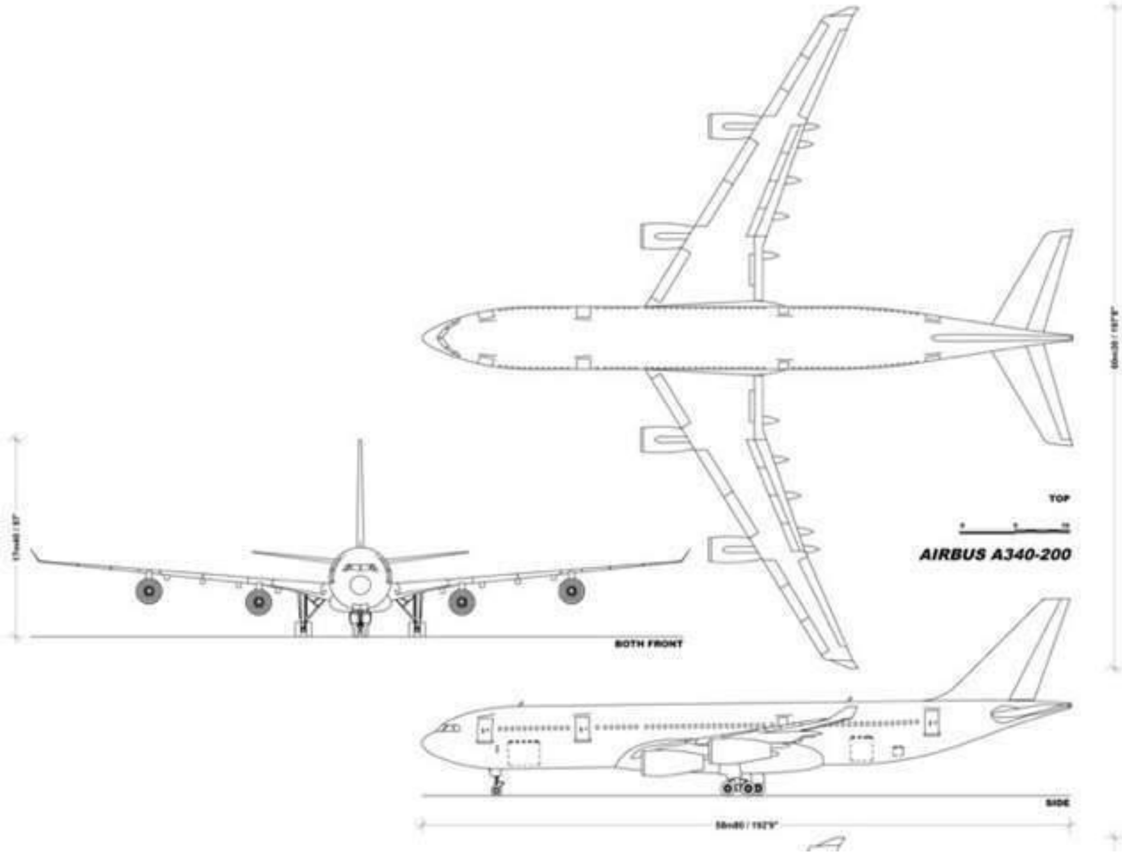


Figure 1.7 — A three view of the Airbus A340-200

General description of Gulfstream III

The Gulfstream III was an improved version of the Gulfstream II. The main differences from the Gulfstream II were redesigned wings with wingtip winglets, increased fuselage length and fuel capacity, and improved fuel efficiency by 18%. Development was announced by Gulfstream in the spring of 1978, and the first prototype first flew on December 2, 1979, receiving FAA type certification on September 22, 1980. Production was halted in September 1988, with a total production capacity of 206. The Gulfstream III is a new deformable wing aircraft developed by NASA. The main feature is the adaptive flexible trailing edge. The top and left views of Gulfstream Ⅲ are shown in figures 1.8 and 1.9.

The Gulfstream III has a crew of 23, a length of 25.3 meters, a wingspan of 23.7 meters, a height of 7.4 meters, an empty weight of 16,576 kilograms, and two Rolls-Royce Spey 511-8 engines. It has a maximum takeoff weight of 31,615 kilograms and a maximum flight speed of 928 kilometers per hour. The maximum range is 6,760 km [11].

Features of wing structure

Gulfstream III aircraft with swept single wing with wing warping; Two Rolls-Royce 157Spey turbofan engines are mounted on either side of the rear fuselage; 5 portholes on each side of fuselage; Swept-back tail, swept-back horizontal tail. Elongated type. With wing warping, the fuel carrying capacity is increased compared with Gulfstream Ⅱ. Adaptive flexible trailing edge technology can retrofit existing aircraft wings or be integrated into new airframes to achieve structural weight reduction, improved aerodynamic performance, and improved fuel economy and operational efficiency. Although the adaptive flexible trailing edge can deform within a certain range, it only flies at a fixed Angle during test flights, collecting data with minimal risk.

Problems

The adaptive trailing edge camber technology still needs to be improved.

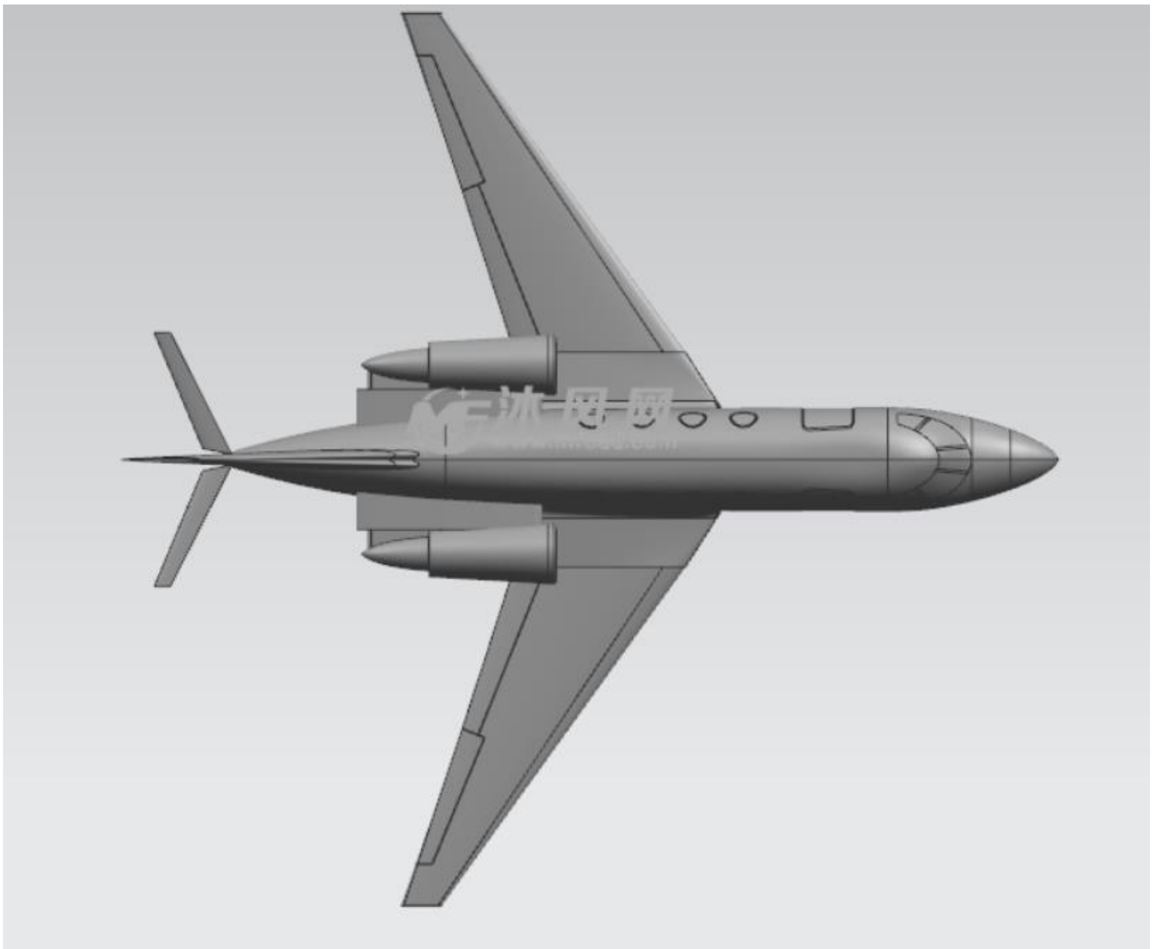


Figure 1.8 — Top view of Gulfstream III

Figure 1.9 — The left view of the Gulfstream III

1.2.4 Rationale for the choice of the adaptive wing of scientific research

Rationale for the choice of the object of scientific research (concretize it) was based on result of analysis by following:

– issues, which were claimed by scientists, who dealt with design of adaptive wing aircraft;

– issues which remained unexplored or not fully covered;

– adaptive wing structure of known real aircraft.

Compared with the conventional wing, adaptive wing technology has its unique advantages:

1) Adaptive wing can make the ratio of friction to lift as small as possible to improve flight efficiency in a variety of flight modes.

2) In the take-off phase of the aircraft, the adaptive wing lift drag is relatively small, which can greatly save the consumption of fuel.

3) The application of adaptive wing enables the pilot to have more choices in the control mode of the aircraft, and at the same time makes the response of the control faster. In other words, the application of adaptive wing enables the pilot to better control a high-speed aircraft.

4) After the flap is divided into as small a proportion as possible, the stalling problem in cruise state will be greatly alleviated. At the same time, flutter and other phenomena affecting flight performance will occur during flight due to the instability of ordinary wings. The application of adaptive wings will reduce this phenomenon [12].

Chosen theoretically successful structure of the adaptive wing of designed aircraft

The wing frame is based on flexible chiral honeycomb cores. The sandwich layer of honeycomb sandwich structure is a series of hexagonal holes made of metal material, glass fiber or composite material. The upper and lower sides of the sandwich layer are then bonded (or brazed) on the thinner plate. Honeycomb sandwich structure of light weight, rigidity, good stability, high strength, sound insulation, heat insulation performance. The chiral honeycombs can maintain effective elastic properties that do not vary under large deformation.

If the flexible mechanism is applied in the adaptive wing, the performance of the aircraft can be improved better. Large deformation of structure is realized by small denaturation. Thus, advanced composite materials should be selected for skin (such as Shape Memory Polymer). The desire is to realize the chiral honeycomb core can maintain a constant Young's modulus and Poisson's ratio under large deformation conditions and minimize the allowable tolerance between the specified and actual properties within the target strain range.

Since the trailing edge of the wing bears relatively small loads, it is easier to deform structurally, improve fuel efficiency and produce better aerodynamic performance, so adaptive trailing edge bending can be used.

1.3 Determining the parameters of an airplane

1.3.1 Aircraft tactical-technical requirements development

Designed airplane with a capacity of up to 22 persons and a range of L = 7000 km, the runway length L_{TO} = 1600 km is assigned, then the cruising altitude H_{CR} = 12000 km, and cruising speed V_{CR} = 980 km/h. There is recorder in table 1.1.

Table 1.1 – Tactical-Technical data of designed airplane

$\mathrm{M_{H=11km}}$	$\mathrm{M_{max}}$	$\mathrm{L(m_p = max), km}$	$\mathrm{m_p, kg}$	L_{TO}, m	H_{max}, km
0.76	0.84	6600	2700	1600	13000

End of Table 1.1

$\mathrm{V_{CR}, km/h}$	$\mathrm{n_{crew}}$	$\mathrm{n_{pass}}$	$\mathrm{H_{CR}, km}$
980 (Subsonic aircraft)	2	20	12000

Based on the processing of statistical data, the assigned basic geometric parameters of the aircraft are listed in table 1.2.

Table 1.2 – Main geometric parameters of designed airplane

λ	$\chi_{1/4}$	η	$\overline{C}$	$\overline{b}_{fl}$	$\delta_{fl}, degree$	$\overline{S}_{al}$	λ_f	D_f, m	L_f, m
6.5	30	3.75	0.33	0.23	26	0.04	10	2.4	24

End of Table 1.2

$\overline{S}_{HT}$	$\overline{S}_{VT}$	λ_{HT}	λ_{VT}	$\chi_{1/4HT}$	$\chi_{1/4VT}$	$\overline{C}_{HT}$	$\overline{C}_{VT}$	η_{HT}	η_{VT}
0.20	0.12	4.33	1.6	26	28	0.10	0.10	2	1.78

1.3.2 Calculation of aircraft zero approximation take-off mass

The take-off mass of the aircraft in the zero approximation is determined after choice the scheme of the aircraft and calculated by the formula [1]:

$$m_0 = \frac{m_p + m_{OI.CE}}{1 - (\overline{m}_S + \overline{m}_{PP} + \overline{m}_{EQ} + \overline{m}_F)}$$

Where m_p – mass of payload, which calculated

$$m_p = m_{lug} \cdot n_{pass} = 120 \cdot 20 = 2400\ kg$$

here $m_{lug} = m_{pass} + m_{bag} = 90 + 30 = 120\ kg$ — mass of luggage.

$m_{OI.CE}$ — mass of operational items and equipment

$$m_{OI.CE} = (m_{crew} \cdot n_{crew}) + \Delta m_{OI} = (80 \cdot 2) + 1500 = 1660 kg$$

here m_{crew} — he mass of service load and crew, it is assumed that the average weight of each crew member is 80 kg; $\Delta m_{OI} = 1500 kg$ — which includes the meals, drinks (alcohol, water, juice), water for service units, unusable fuel, instruments, magazine, entertainment (screws, films, headphones, plays) cover of the floor and sets, blankets, pillows, toothbrushes, toilet paper, towels, crockery, food cart, oils, medical equipment.

$\overline{m}_s$ — Is the relative mass of the aircraft structure, which includes the relative mass of the wing, fuselage, tail, and landing gear. For subsonic, passenger, main heavy aircraft $\overline{m}_s$ =0.28.

$\overline{m}_{PP}$ — Relative mass of the power plant, which consists of the relative mass of the engine with the means of their installation and servicing system. In this case, it I assumed that $\overline{m}_{PP} = 0.10$.

$\overline{m}_f$ — The relative mass of fuel, which is found by empirical formula:

$$\overline{m}_f = a + b\frac{L}{V_{CR}} = a + b \cdot t_p = 0.06 + 0.05 \cdot 7 = 0.41$$

where L — is the flight range L= 7000 km, V — is the speed, V= 980 km/h, t_p — the estimated flight time, t_p = 7 hours, a =0.06 , b =0.05.

$\overline{m}_{EQ}$ — Relative weight of equipment and control, which, which includes hydraulic system, pneumatic system, power supply system, flight-navigation equipment and elevator control, rudder, ailerons, flaps, slats, interceptors. For transonic aircraft $\overline{m}_{EQ} = 0.10$ [13].

The take-off mass of the aircraft in the zero approximation is calculated by formula:

$$m_o = \frac{2400 + 1660}{1 - (0.28 + 0.10 + 0.41 + 0.10)} = 36909.09 \approx 36909 \text{kg}.$$

1.3.3 Calculation of structural mass of the main aircraft assemblies, power plant mass, fuel mass, mass of the equipment and control

After determining the take-off weight of the aircraft in the zero approximation, it is necessary to determine:

Aircraft design weight: $m_s = 0.28 \cdot m_o = 10334.52 \approx 10335kg$

Wing weight: $m_W = 0.396 \cdot m_s = 4092.66 \approx 4093kg$

Fuselage mass: $m_F = 0.351 \cdot m_s = 3627.585 \approx 3628kg$

Tail weight: $m_{TU} = 0.069 \cdot m_s = 713.115 \approx 713kg$

Landing gear weight: $m_{LG} = 0.184 \cdot m_s = 1901.64 \approx 1901kg$

Fuel mass: $m_f = 0.41 \cdot m_o = 15132.69 \approx 15132kg$

Mass of the power plant: $m_{PP} = 0.10 \cdot m_o = 3690.9 \approx 3691kg$

Mass of the equipment: $m_{EQ} = 0.10 \cdot m_o = 3690.9 \approx 3691kg$

In the table 1.3 are indicated the masses of the airplane parts and units.

Table 1.3 – Masses of the airplane parts and units

m_o, kg	$m_{OI.CE}$, kg	m_p, kg	m_{CR}, kg	m_f, kg	m_{PP}, kg	m_{EQ}, kg	m_S, kg			
							m_W, kg	m_F, kg	m_{TU}, kg	m_{LG}, kg
36909	1660	2400	160	15132	3691	3691	4093	3628	713	1901
							10335			

1.3.4 Airplane geometrical parameters calculation (wing, fuselage, tail, landing gear)

Determination of the geometric parameters of the wing

The specific load on the during take off is p_o =190 KN

$$S = \frac{\mathrm{m_o} \cdot \mathrm{g}}{10 \cdot \mathrm{p_o}} = \frac{36909 \cdot 9.8}{10 \cdot 190} = 190.37 \approx 190\ \mathrm{m}^2$$

Wing span:

$$\mathrm{l} = \sqrt{\lambda S} = \sqrt{6.5 \cdot 190} = 35.14\ \mathrm{m}$$

The root b_0 and tip b_t chord of the wing is determined

$$b_0 = \frac{S}{l} \cdot \frac{2 \cdot \eta}{\eta + 1} = \frac{190}{35.14} \cdot \frac{2 \cdot 3.75}{3.75 + 1} = 8.54\ m$$

$$b_t = \frac{b_0}{\eta} = \frac{8.54}{3.75} = 2.28\ m$$

The mean aerodynamic chord is calculated:

$$b_{MAC} = \frac{2}{3} \cdot b_0 \cdot \frac{\eta^2 + \eta + 1}{(\eta + 1) \cdot \eta} = \frac{2}{3} \cdot 8.54 \cdot \frac{3.75^2 + 3.75 + 1}{(3.75 + 1) \cdot 3.75} = 6.01 \approx 6\ m$$

Determine the coordinate MAC on the wingspan:

$$Z_A = \frac{l}{6} \cdot \frac{\eta + 2}{\eta + 1} = \frac{35.14}{6} \cdot \frac{3.75 + 2}{3.75 + 1} = 7.09\ m$$

Coordinate of MAC nose along an OX axis

$$X_{MAC} = \frac{l}{6} \cdot \frac{\eta + 2}{\eta + 1} \cdot tg\chi_{LE} = \frac{35.14}{6} \cdot \frac{3.75 + 2}{3.75 + 1} \cdot 0.67 = 4.75\ m$$

Where χ_{LE} — Sweep angle of wing leading edge.

$$tg\chi_{LE} = tg\chi + \frac{\eta - 1}{\lambda(\eta + 1)} = tg30° + \frac{3.75 - 1}{6.5(3.75 + 1)} = 0.67$$

Determination of fuselage parameters

Fuselage diameter

$$D_F = 2.4\ m$$

Fuselage length

$$L_F = \lambda_F \cdot D_F = 10 \cdot 2.4 = 24\ m$$

The length of the nose of the fuselage

$$L_N = \lambda_N \cdot D_F = 1.6 \cdot 2.4 = 3.84\ m$$

The length of the rear fuselage

$$L_T = \lambda_T \cdot D_F = 3 \cdot 2.4 = 7.4\ m$$

Determination of tail unit parameters

The horizontal tail is determined according by such dependence:

$$S_{HS} = \overline{S}_{HS} \cdot S = 0.20 \cdot 190 = 38\ m^2$$

Length of horizontal tail:

$$L_{HS} = \sqrt{\lambda_{HS} \cdot S_{HS}} = \sqrt{4.33 \cdot 38} = 12.83\ m$$

Chords of horizontal tail unit

$$b_{OHS} = \left(\frac{S_{HS}}{L_{HS}}\right) \cdot \frac{2\eta_{HS}}{\eta_{HS}+1} = \frac{38}{12.83} \cdot \frac{2 \cdot 2}{2+1} = 3.95m$$

$$b_{t\,HS} = \frac{b_{OHS}}{\eta_{HS}} = \frac{3.95}{2} = 1.975\,m$$

Mean aerodynamic chord of horizontal tail

$$b_{MAC\,HS} = \frac{2}{3} \cdot b_{OHS} \cdot \frac{\eta_{HS}^2 + \eta_{HS} + 1}{\eta_{HS} \cdot (\eta_{HS}+1)} = \frac{2}{3} \cdot 3.95 \cdot \frac{2^2+2+1}{2(2+1)} = 3.07m$$

$$Z_{MAC\,HS} = \frac{L_{HS}}{6} \cdot \frac{\eta_{HS}+2}{\eta_{HS}+1} = \frac{12.83}{6} \cdot \frac{2+2}{2+1} = 2.85m$$

Horizontal distance between wing root to MAC root

$$X_{MAC\,HS} = \frac{l}{6} \cdot \frac{\eta_{HS}+2}{\eta_{HS}+1} \cdot tg\chi_{nHS} = 2.85 \cdot 0.56 = 1.596m$$

where χ_{LE} — sweep angle of wing leading edge.

$$\mathrm{t}g\chi_{\mathrm{nHS}} = \mathrm{t}g\chi + \frac{\eta_{HS}-1}{\lambda_{HS}(\eta_{HS}+1)} = tg26° + \frac{2-1}{4.33(2+1)} = 0.56$$

The vertical tail is determined:

$$S_{VS} = \overline{S}_{VS} \cdot S = 0.12 \cdot 190 = 22.8m^2$$

Height of vertical tail:

$$H_{VS} = \sqrt{\lambda_{VS} \cdot S_{VS}} = \sqrt{1.6 \cdot 22.8} = 6.04m$$

Chords of vertical tail unit

$$b_{OVS} = (\frac{S_{VS}}{L_{VS}}) \cdot \frac{2\eta_{VS}}{\eta_{VS}+1} = \frac{22.8}{6.04} \cdot \frac{2 \cdot 1.78}{1.78+1} = 4.83m$$

$$b_{t\,VS} = \frac{b_{OVS}}{\eta_{VS}} = \frac{4.83}{1.78} = 2.71m$$

Mean aerodynamic chord of vertical tail unit

$$b_{MAC\,VS} = \frac{2}{3} \cdot b_{OVS} \cdot \frac{\eta_{VS}^2 + \eta_{VS} + 1}{\eta_{VS} \cdot (\eta_{VS}+1)} = \frac{2}{3} \cdot 4.83 \cdot \frac{1.78^2 + 1.78 + 1}{1.78(1.78+1)} = 3.87m$$

$$Y_{MAC\,VS} = \frac{h_{VS}}{6} \cdot \frac{\eta_{VS}+2}{\eta_{VS}+1} = \frac{6.04}{6} \cdot \frac{1.78+2}{1.78+1} = 1.37m$$

Vertical distance between wing root to MAC root

$$X_{MAC\ vS} = Y_{MAC\ VS} \cdot tg\chi_{nVS} = 1.37 \cdot 0.92 = 1.26m$$

Determine landing gear parameters

The size of offset e is more often determined in share of wing MAC

$$(0.15\ ...\ 0.20)\ b_{MAC} = 0.15 \cdot 6 = 0.9\text{m}.$$

Angle of main wheels setoff γ should be higher than angle of a touch down by a tail part

$$\gamma = \varphi + (1\ldots2)°$$

φ is angle of touchdown by the tail part

$$\varphi = \alpha_{\max} - \alpha_{\text{w}} - \psi$$

Where $\alpha_w = (0\ldots4)°$, (angle between a wing chord and longitudinal axis of fuselage), ψ=0 (static ground (parking angle) in zero approximation it is ψ=0).

$$\varphi = 10 - 2 - 0 = 8°$$

$$\gamma = 8 + 2 = 10°$$

Regarding the center of mass e with angle Φ the greater the value e the greater the front tail support loads and more difficult to take off a front support during tak off. But the lesser the e reduce γ.

Wheel base of the landing gear b it depends of fuselage length

$$\text{b} = (0.30\ldots0.40)\text{L}_\text{F} = 0.30 \cdot 24 = 7.2\text{m}$$

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

$$\text{a} = (0.88\ldots0.94)\text{b} = 0.90 \cdot 7.2 = 6.48\text{m}$$

$$\text{e} = (0.12\ldots0.06)\text{b} = 0.10 \cdot 7.2 = 0.72\text{m}$$

H — height of loading gear is determined from the condition providing the minimum gap 200...250 mm between the runway surface and the airplane structure.

B – the maximum size of track, which is 12 m by statistical range.

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

$$2\text{H} < \text{B} < 15\text{m}$$

Calculation of H:

$$H = \frac{e}{\tan\gamma} = \frac{0.72}{\tan 10°} = 4.08$$

Chosen size of track is satisfied condition, therefore B = 10 m.

Development of the layout and center-of-gravity.

The position of an aircraft's center of gravity largely determines the state of the aircraft's flight — its balance, stability and maneuverability. The center of gravity positioning should meet the following requirements: The position before and after the center of gravity should be selected to ensure sufficient longitudinal static stability of the aircraft under normal flight conditions. For the stability of the run, the vertical distance between the center of gravity and the ground should not be too large. In order to improve the wind resistance of the aircraft on the ground, the installation position of the aircraft wing should be reduced as far as possible. The jettifiable load and fuel on the aircraft should be located near the center of gravity. During flight, the center of gravity of the whole aircraft should change within a small range.

Position of the airplane center of mass is determined relative to nose part of wing mean aerodynamic chord. The recommendation distance for the center of mass (point 0) from nose part of mean aerodynamic chord X_m has such values for the airplanes with swept wing.

$X_{C.G.} = (0.26\ldots0.30)b_{MAC}$ — distance from the center of mass to the nose of the wing MAC:

For the swept wing χ (30.... 60)° L_{HS} = (2.5 ... 3.5) · b_{MAC} = 3 · 6 = 18 — the distance form one-fourth of the MAC horizontal tail to the center of mass.

Operational cases you can find in regulations, such as CS-24, or in regulations of aircraft certification.

The calculation of the center of mass of the designed aircraft was performed for the following operational case:

— aircraft weight — take-off,
— landing gear — retrectable,
— payload — maximum,
— fuel weight — maximum.

1.4 Justification of the aircraft scheme. Three-dimensional parametric modelling

The airplane has a classical aerodynamic scheme with lower wing and single vertical tail and horizontal tail located in the end part of the fuselage and it has a retractable tricycle landing gear with nose landing gear which is located in nose part of the fuselage and with forward retraction, the main landing gear is located bellow the wing and is retract in the fuselage. The aircraft is equipped by control surfaces (aileron and spoilers on the wing, rudder on the vertical tail, and elevator on horizontal stabilizer) and mechanization (lift devices) – double slotted flap and retractable slats. The aircraft have two turbofan engines located under the wing fixed with pylon" – rewrite it to your designed aircraft.

To create parametric model of master-geometry of designed aircraft the three-dimensional software Siemens NN 7.5 were used.

In order to meet the requirements of flight speed, the fuselage adopts the stringer fuselage in the semi-hard shell fuselage. The fuselage consists of stringer, skin and enclosure frame. Stringer and skin strong, good compression stability, bending moment caused by all the axial force by the upper and lower skin and stringer composed of the wall plate tension, pressure to bear. The fuselage structure has high torsional stiffness, thick skin and small local deformation, which is conducive to improving aerodynamic performance and is suitable for high-speed aircraft. This fuselage maximizes the carrying capacity of the stringer and skin while reducing structural weight. The modeling and main parameters of fuselage are shown in figure 1.10 and figure 1.11.

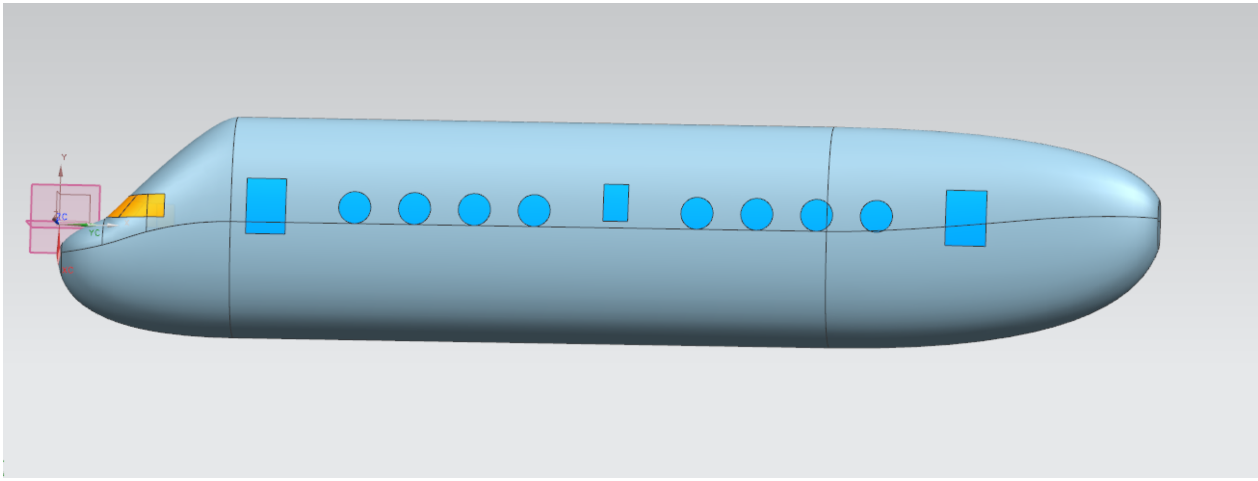


Figure 1.10 — The modeling of fuselage

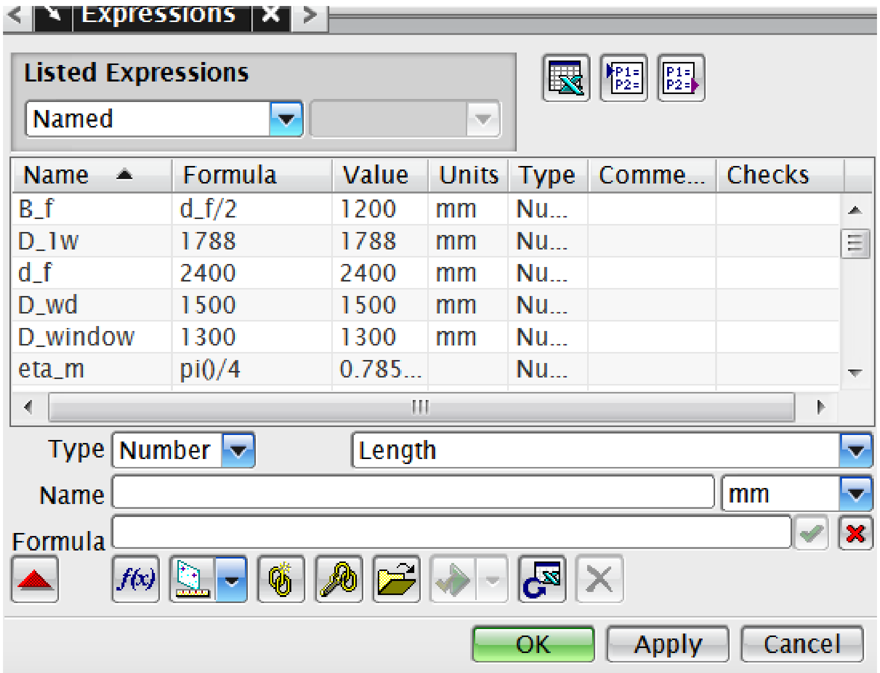


Figure 1.11 — The main parameters of fuselage

Since the aircraft's maximum Mach number is 0.84, the wing is designed to have a sweep angle of 34° in their initial state. This can delay the generation of shock waves on the wing surface, and even if they do occur, it helps to reduce shock intensity and reduce flight drag.

The airfoil is NACA-2214. The relative cambour of the airfoil is 2%, the maximum cambour position is 0.2 chord length, and the relative thickness is 14%. It has higher maximum lift coefficient and lower drag coefficient. The modeling and main parameters of wing is shown in figure 1.12 and figure 1.13.

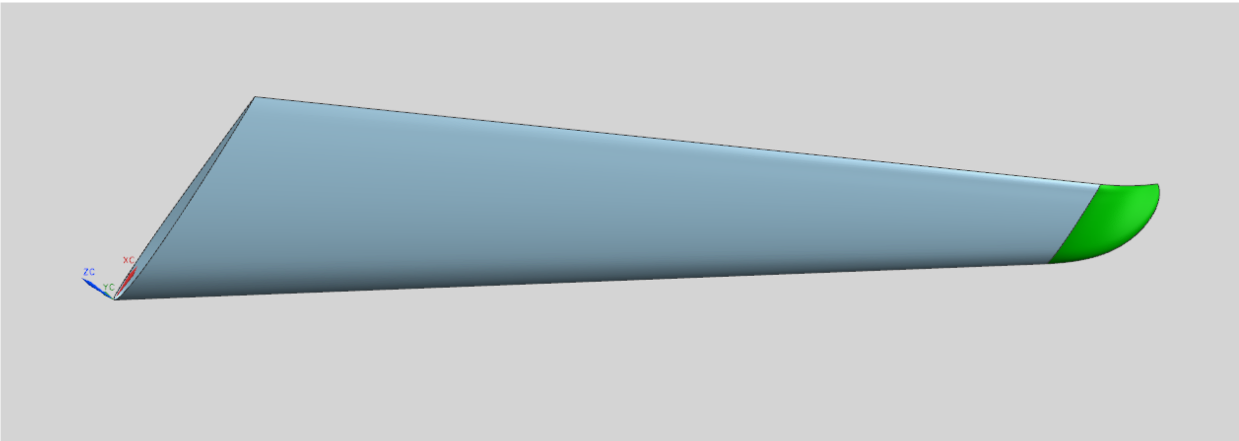
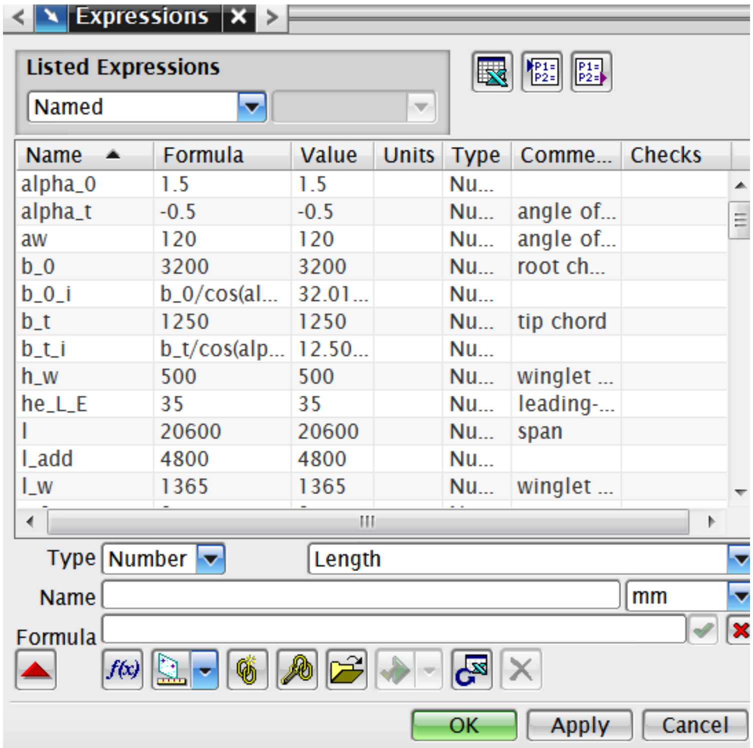


Figure 1.12 — The modeling of wing



Name	Formula	Value	Units	Type	Comme...	Checks
alpha_0	1.5	1.5		Nu...		
alpha_t	-0.5	-0.5		Nu...	angle of...	
aw	120	120		Nu...	angle of...	
b_0	3200	3200		Nu...	root ch...	
b_0_i	b_0/cos(al...	32.01...		Nu...		
b_t	1250	1250		Nu...	tip chord	
b_t_i	b_t/cos(alp...	12.50...		Nu...		
h_w	500	500		Nu...	winglet ...	
he_L_E	35	35		Nu...	leading-...	
l	20600	20600		Nu...	span	
l_add	4800	4800		Nu...		
l_w	1365	1365		Nu...	winglet ...	

Figure 1.13 — The modeling of wing

The horizontal tail adopts the same shape as the wing and has similar aerodynamic characteristics as the wing. The structure of the single vertical tail is simple. Because the distance from the axis is very close, the single vertical tail does not need to carry out excessive reinforcement at the wing root, which reduces the weight of the structure, saves the cost, and is also very simple in production. The modeling and main parameters of horizontal tail are shown in figure 1.14 and figure 1.15.

The number of vertical tail is one, it adopts the form of single vertical tail. The modeling and main parameters of vertical tail is shown in figure 1.16 and figure 1.17.

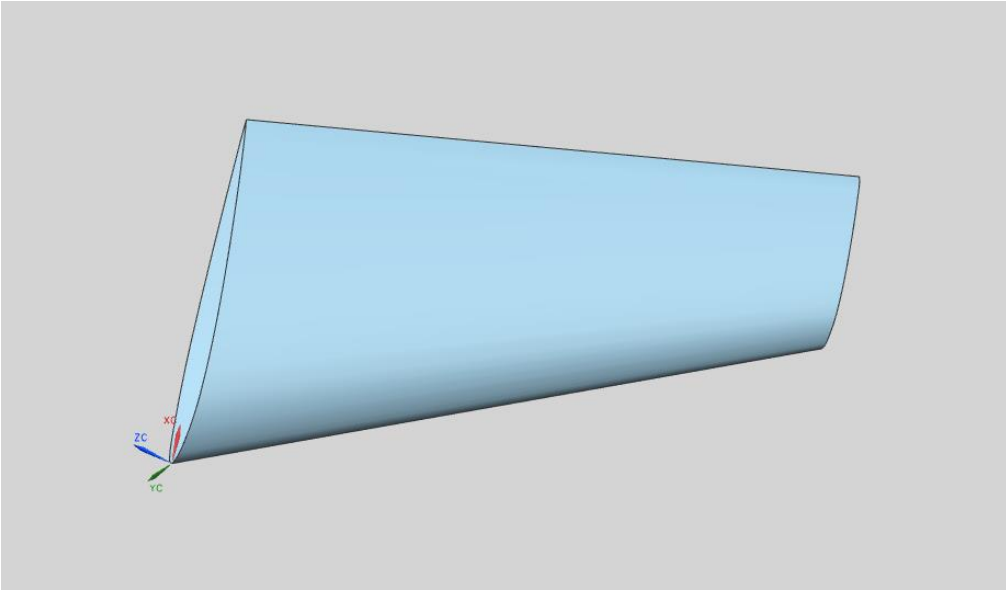


Figure 1.14 — The modeling of horizontal tail

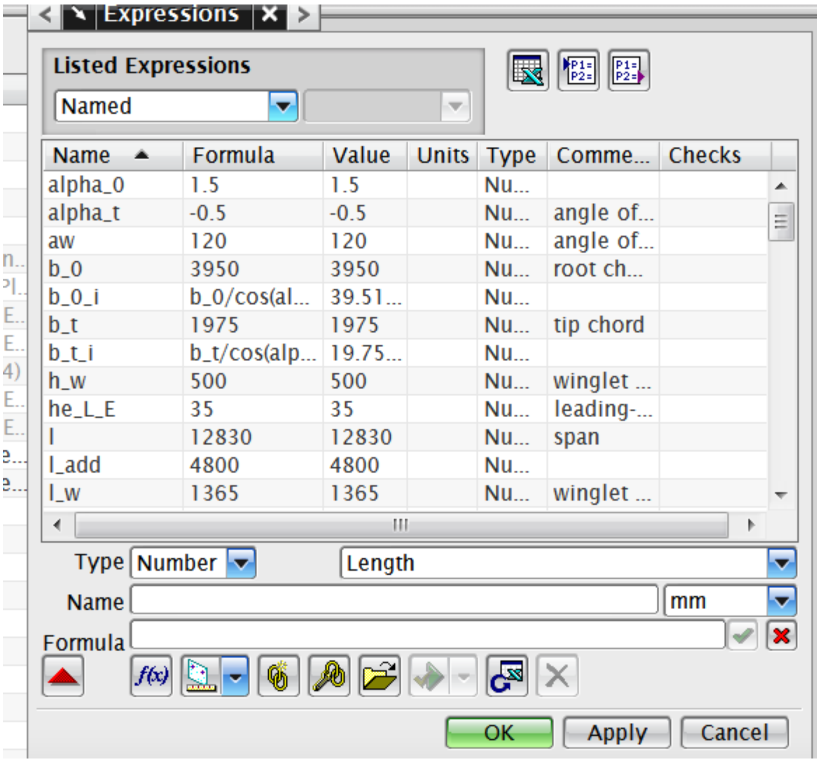


Figure 1.15 — The main parameters of horizontal tail

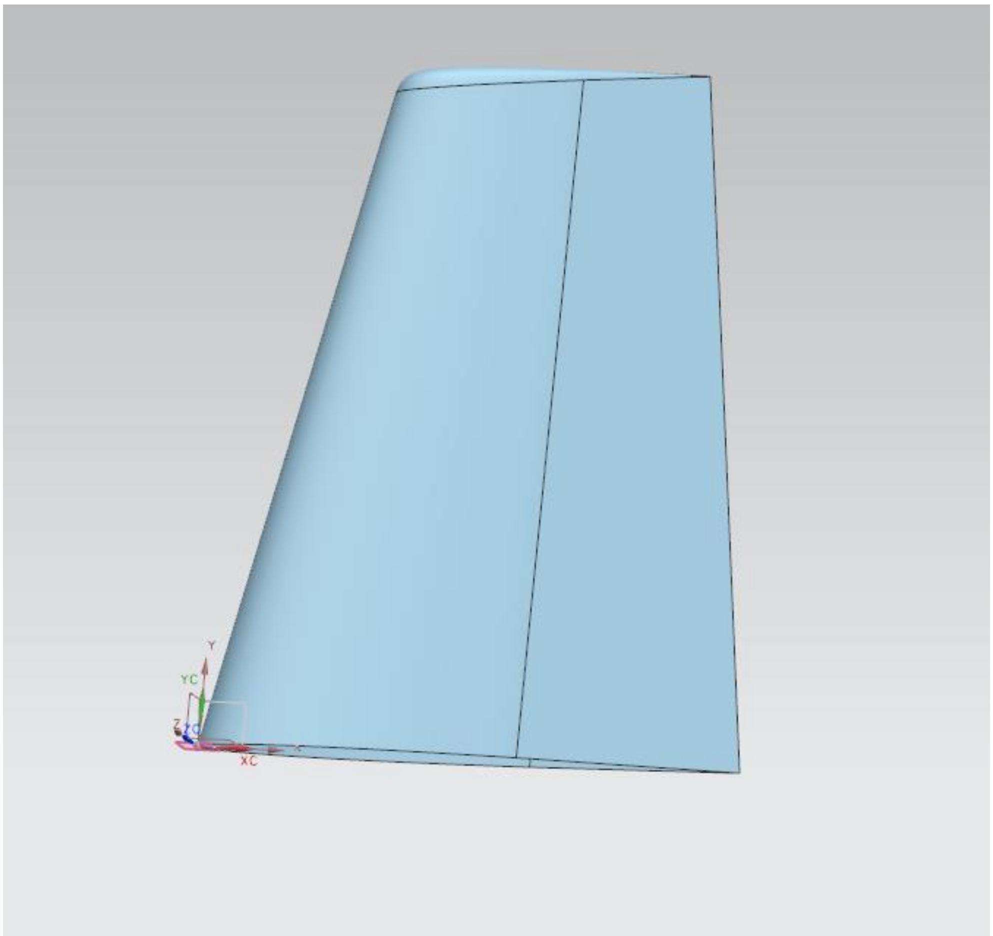


Figure 1.16 — The modeling of vertical tail

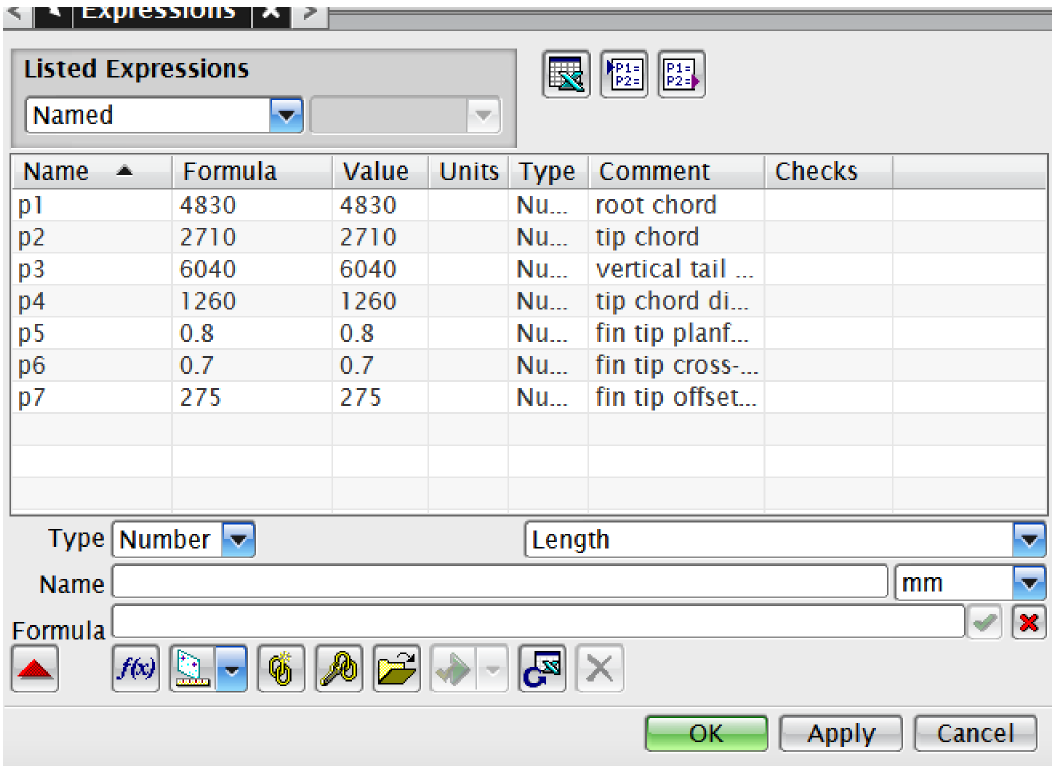


Figure 1.17 — The main parameters of vertical tail

The assembly diagram of the aircraft is shown in figure 1.18 and figure 1.19.

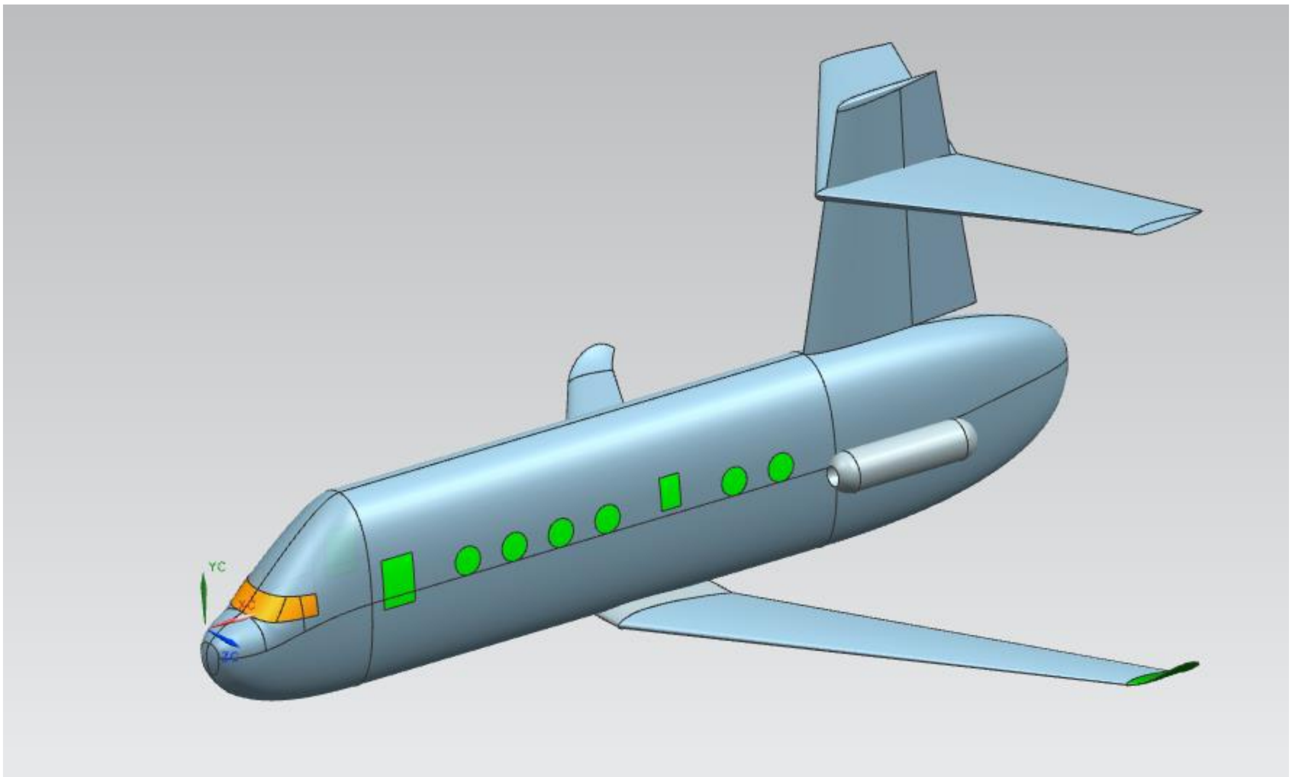


Figure 1.18 — The assembly diagram of the aircraft

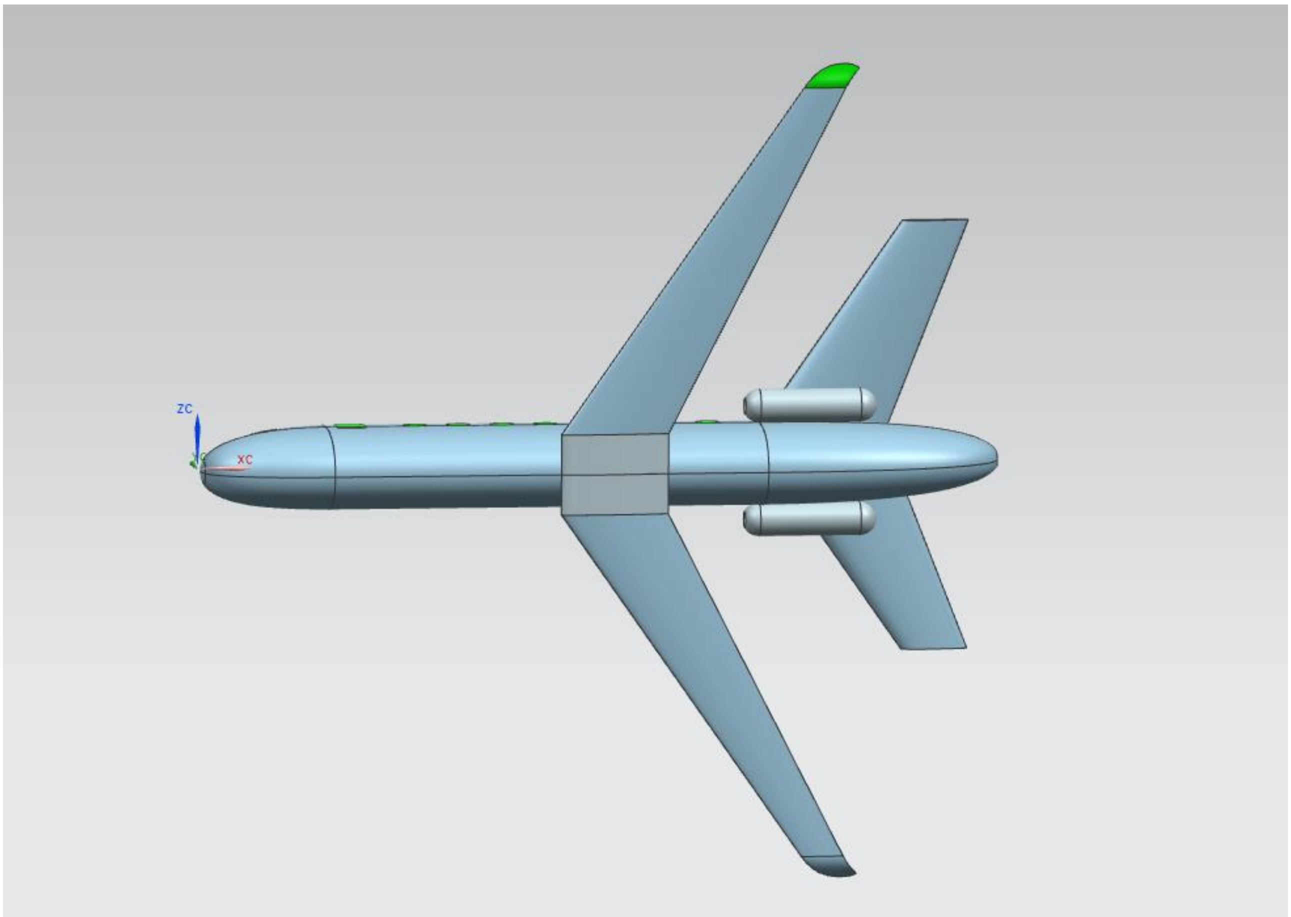


Figure 1.19 — The assembly diagram of the aircraft bottom

The tail adopts the form of T-tail, that is, the horizontal tail is at the top of the vertical tail. When flying at a small Angle of attack, the horizontal tail will not be affected by the wake of the aircraft wing, and the horizontal tail has high control efficiency. Therefore, even if the area of the horizontal tail is small, it can meet the needs of aircraft control and effectively reduce the weight of the aircraft. Because the horizontal tail fin position is relatively high, so you can do the tail door in the tail opening, convenient loading and unloading of cargo, take the plane is more comfortable.

The wing is mounted on the underside of the fuselage in the form of lower wings. Because the plane is designed to be a manned, long-range business jet, the aircraft adopts lower wing, the engine is shielded by the wing, the comfort is better, and the engine is close to the ground, the maintenance is more convenient. The wing's lower position also serves as an escape route to provide a safer escape point for passengers, at the same time emergency landing on the fuselage can also play a protective role. Because the mechanical structure of the lower wing is simple and the design resistance is small, the wind noise will be relatively small. This design concept has become a natural noise reduction processing method, which can also give passengers a better ride experience. The lower-wing design provides the aircraft with less drag and more lift, which also reduces energy consumption. In addition, the lower wing has good takeoff and landing performance, and it is easy to arrange the landing gear position. The wing of the lower wing aircraft is close to the ground, and the landing gear can be shorter. The distance between the two landing gear is wider, which increases the stability of the landing gear. The landing gear is easy to retract, reducing weight. At the same time the lower wing spars under the floor, easy to arrange the cabin.

The engine is powered by two turbofans. This kind of engine has large thrust, high propulsion efficiency, low noise and high fuel efficiency, which can meet the range requirements of aircraft. The engine is installed at the tail of the fuselage, the aerodynamic interference to the wing is small, the lift-drag ratio of the wing is increased, the lifting device can be installed on the wing to the maximum extent, and the directional torque of a single engine when it stops is small, the cabin noise of most cabin sections is low, the jet flow is higher from the ground, reduce the runway damage, take-off and landing, the engine is not easy to breathe foreign matter.

1.5 Integrated design of the wing of the designed aircraft

The wing adopts adaptive camber trailing edge design. Because of the relatively small load on the trailing edge of the wing, it is structurally easy to achieve, and brings great aerodynamic benefits. The wing curvature can not only change along the chord, but also along the spanwise differential change to realize the adaptive torsion of the wing.

The trailing edge of the wing is made into a flexible mechanism, and the trailing edge of the wing is controlled by a driving system to produce upward and downward displacement, so as to adapt to different flight conditions.

For the optimization method, morphing 3-point optimizations with varying lift coefficient and morphing 7-point optimizations with varying lift coefficient, Mach number, and altitude were considered [13]. Finally, comparison those results using a smaller morphing device and a configuration with a higher aspect ratio.

Needs to optimize the coupling of high-fidelity aerodynamic and structural models in a trade-off between structural weight and aerodynamic robustness is important in adaptive wing investigation. Adding adaptive deforming trailing edge device can reduce the weight of structure obviously morphing technology is more effective for higher aspect ratio wings.

Concept of aircraft design:

From the history of the aircraft industry, the idea of adaptive wings was developed by the Wright brothers through their concept of wing warping. In 1903, the Wright Brothers invented the first motorized aircraft, which had fixed biplanes and used lever control to control the geometric characteristics (torsion, bending, etc.) of the middle and lower wings of the biplanes to achieve small amplitude deformation of the wings. This aircraft was the first motorized aircraft and the earliest variable-wing aircraft [14].

In 1985, NASA's Ames-Dryden Laboratory, the U.S. Air Force, and Boeing launched the concept of a mission adaptive wing [15]. Mission adaptive wing can deflect the leading edge or trailing edge of the wing by actuators according to different flight conditions and requirements, so that the wing can achieve an ideal bending shape. The leading and trailing edge of the wing should be smooth, no gaps, completely closed [16].

In the 1980s, the United States selected AFTI/F -111 as the mission adaptive wing-carrying verification platform and conducted 59 flight tests. The wings have a drag reduction of about 7 percent in the designed cruise condition and more than 20 percent in the off-design condition. However, there are also drive system mechanical structure complex, large additional mass and other shortcomings [16].

In 1995, With support from DARPA, Northrop Grumman companies are beginning to research and test the concept of smart wings. In a 16% scale model of an F A -18 aircraft, the hinge-free fairing trailing edge control surface is driven by a shape-memory alloy (SMA) and the wing torsion is driven by a SMA torsion tube. The adaptive wing can increase the rolling moment by 8% ~ 15% and lift by 5% ~ 8% , and the pressure distribution is improved. It was later modified on a 30% full-wingspan scaled model of a Northrop Grumman Company unmanned combat aircraft, and verified that in the wide Mach range, high-frequency band drive systems were developed and validated for roll performance using hinge-free, smooth airfoils, and shape-memory alloy (SMA) driven leading and trailing edge control surfaces, the frequency and angle of deformation control are increased (deflection is 25°) [16].

Sridhar Kota et al put forward the concept of compliant mechanism, which is a flexible structure with single point drive and controllable deformation to transfer motion or force through elastic deformation of each metal component unit. This mechanism can magnify the movement displacement and energy of the smart material, and transfer it to the leading and trailing edges of the wing, so that the leading and trailing edges of the wing can deflect continuously and seamlessly [16].

R. Vos et al studied the method of deformation of trailing edge of wing model by piezoelectric material. This post-buckling variant of the wing to enhance the rolling control force, compared with the traditional motor-driven scheme, this scheme can reduce the weight of 3.5% [16].

Heinze and Karpe studied an aeroelastic lever, which is composed of a trailing-edge flap adjuster driven by a piezoelectric chip, to amplify the aeroelastic effect and deflect the trailing-edge flap. When the tab is deflected -2.5° can produce a 4.6° rotation angle of the flap, which verifies the feasibility of the piezoelectric material to provide the aeroelastic shape control driving force for the control surface [16].

Sofla has developed a range of shape-memory alloy flexible structures. The utility model has the advantages that the aerodynamic characteristics can be enhanced, and the wing has very strong mission adaptive ability. The downside is that the shape-memory alloy cools slowly [16].

In 2006, Joel A. Hetrick et al. published research on flexible wings for mission compliance. The advantage is that the trailing edge structure of the wing is light in weight, requires only a small driving force, can produce deformation, has sufficient strength and stiffness [16].

Yokozeki et al. investigated a flexible seamless aileron structure driven by a corrugated plate structure. The advantage is that it provides better aerodynamic performance, but the disadvantage is that the trailing edge structure can only be deflected downward and is only used to improve the lift characteristics of the aircraft during low-speed flight tasks such as take-off and landing [16].

In 2011, Qiu Tao et al. designed a flexible skin structure, which is composed of flexible honeycomb and elastic adhesive film. The advantage of the flexible skin structure is that it has better deformation ability in the face and certain bearing capacity outside the face [16].

In 2013, Benjamin K. S. et al. proposed a fishbone active variable camber wing structure. This structure can provide a large up and down deflection deformation capacity, and the drive capacity requirements are low [16].

In 2014, Chen Yijin studied three kinds of deformable skins and structures from experimental and theoretical perspectives: deformable skins with negative Poisson's ratio effect, actively deformable skins and variable stiffness tube structures that can be inserted into skins; Two kinds of internal wing support structures, negative Poisson's ratio honeycomb and zero Poisson's ratio honeycomb, are studied from finite element, theoretical and experimental perspectives. A composite wing skin structure is designed and studied, and its deformation and bearing capacity are verified by experiments [16].

In 2015, James J. of the U.S. Air Force Research Laboratory published the results of a variable camber compliant wing using a compliant mechanism and advanced manufacturing techniques. The advantages of this structure are light weight, small driving energy and low cost [16].

Adaptive wings can be divided into three main categories according to the way they deform: in-plane deformation, out-of-plane deformation and airfoil thickness deformation. The in-plane deformation includes the wing with variable spread length and wing with variable sweep Angle; Out of plane deformation includes torsional wing, folding wing, variable camber wing and variable tip wing [14]. This paper mainly studies the concept design of adaptive trailing edge variable camber wing.

The key techniques of adaptive wing design are:

(1) Design a wing skin with strong bearing capacity and deformation capacity;

(2) A new driving system is designed to drive the wing to produce deformation, which has the characteristics of light weight, high efficiency, quick response, low energy consumption and easy control;

(3) A cooperative control system can realize the coordination and synchronization of multiple drivers;

(4) Design a sensor system that can sense the surrounding environment and the structural state of the wing [18].

With conventional wings, the efficiency of the rudder is affected by gaps between the ailerons, flaps and the main wing. The adaptive wing adjusts the flow and aerodynamic loads on the airfoil by changing the geometry of the aircraft wing or using flow control method in flight, so that the aerodynamic forces in the whole flight envelope are close to the optimal state, so as to improve flight performance and reduce structural weight. Thus, the adaptive wing has no conventional ailerons, flaps, slats, or spoilers. The wing itself can bend flexibly to the desired position according to the mission requirements. The adaptive wing has no gaps in the rudder and no hinge torque.

The adaptive trailing edge variable camber wing designed in this paper mainly includes the main wing, ribs, spars, stringers, flexible trailing edge part, upper and lower surface skin. The space distribution model of the wing is shown in the figure 1.20.

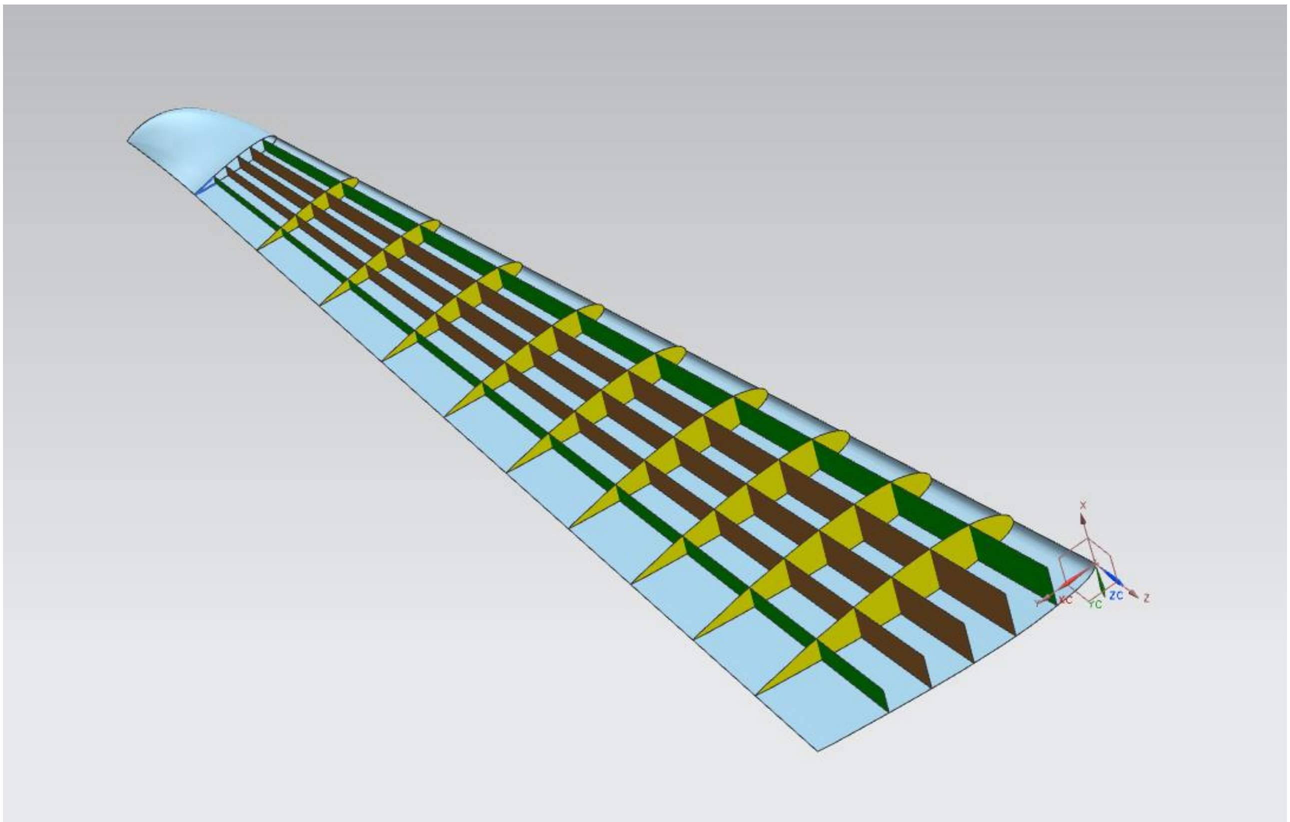


Figure 1.20 — The space distribution model of the wing

1. Flexible trailing edge part (Adaptive support structure)

The flexible trailing edge section is attached to the rear spar of the wing, and the internal structure is chiral honeycomb core. Actuate through a SMA(Shape Memory Alloys) actuator, varying from -30° to +30° . The flexible trailing edge section and hipotetical mechanism are shown in figure 1.21 and figure 1.22.

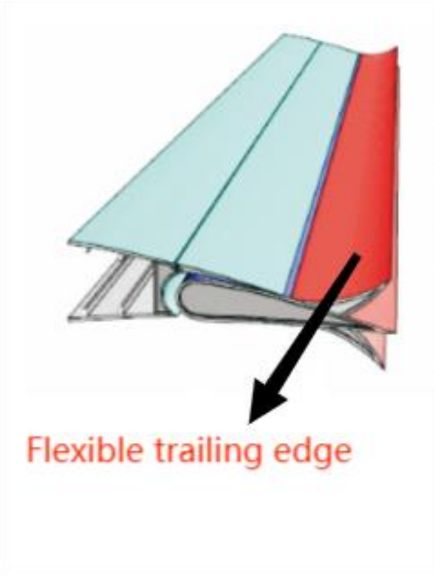


Figure 1.21 — Flexible trailing edge section

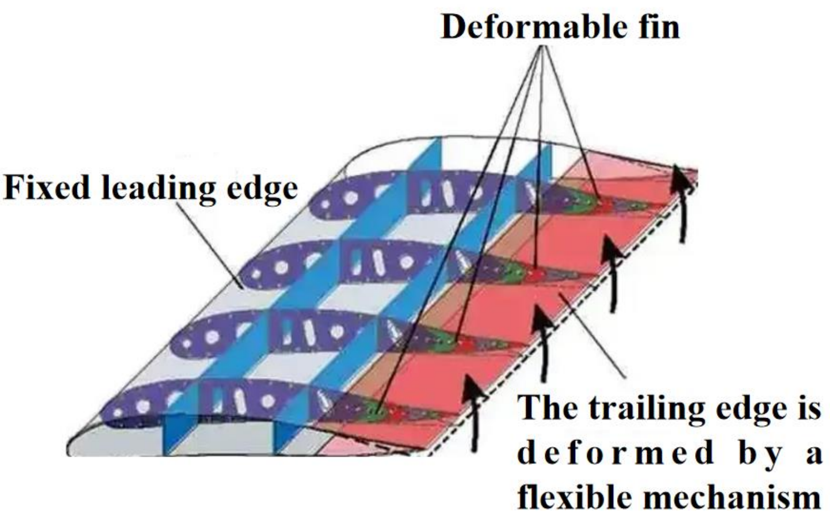


Figure 1.22 — Hipotetical mechanism

Flexible mechanism is a kind of flexible mechanism which generates a lot of mechanical motion through the elastic deformation of part or all of its flexible members. Compared with the traditional mechanism, the flexible mechanism has fewer components, can be designed as a whole, no need for assembly, no friction, wear and transmission clearance, small invalid stroke, can achieve high precision movement, improve reliability, and easy to manufacture. Therefore, if the flexible mechanism is applied in the adaptive wing, the performance of the aircraft can be improved better. Large deformation of structure is realized by small denaturation.

SMA has memory effect. It is an alloy material that can completely eliminate the deformation occurred at the previous temperature and restore the original shape before deformation at a certain temperature. It has much larger deformation recovery ability than common metals, so it is more suitable to be used as the driver of the flexible trailing edge.

The sandwich layer of honeycomb core structure is a series of hexagonal holes made of metal material, glass fiber or composite material. The upper and lower sides of the sandwich layer are then bonded (or brazed) on the thinner plate. Honeycomb sandwich structure of light weight, rigidity, good stability, high strength, sound insulation, heat insulation performance. The honeycomb core structure is shown in figure 1.23.

Figure 1.23 — The honeycomb core structure

According to the research of Qiu Kepeng et al, the traditional honeycomb structure Poisson's ratio is generally positive, while the chiral honeycomb core structure can be negative, and different negative Poisson's ratios can be realized. By optimizing the parameters, the chiral honeycomb core structure can simultaneously achieve deformation, light weight, high specific stiffness and negative Poisson's ratio characteristics.

Chiral honeycomb core can reduce the weight of morphing aircraft structure, because of its high stiffness-to-weight ratio and designable nonlinear features. It also has low in-plane stiffness and high out-plane flexural bending stiffness, and can withstand large global displacement under limited strain. So the flexible trailing edge part choose chiral boneycomb core by these features.

According to the research of Qiu Kepeng et al, the chiral honeycomb structure should be optimized. First need to determine the optimal formula for the chiral hexagon honeycomb. Then need to optimize the design according to the specified Young's modulus and Poisson's ratio. The optimization model is established and verified by experiments. Hexagonal chiral honeycomb core is special material and structure with negative Poisson’s ratio. The nonlinearity of hexagonal chiral honeycomb core structure directly affects its effective elastic properties. Finally need to study new chiral honeycomb core with specified elastic modulus and Poisson's ratio under large deformation conditions [19].

2. Upper and lower surface skin

For the skin material, a new type of intelligent material — SMP (Shape Memory Polymer) is used. SMP is similar to SMA. SMP is a material that can change stiffness.

In its initial state, SMP is a high modulus rigid material. Under certain conditions, after external stimulation, it will reduce its stiffness and become a low modulus flexible material. When this external stimulus is removed, the SMP returns to its original state of high modulus rigid material. If this stimulus is applied again, the SMP will again be a low modulus flexible material. Because of this special memory effect, SMPS are more suitable as the upper and lower skin materials for the flexible part of the trailing edge. External stimulation can be generated by electric heating, so that the stiffness of the skin material changes, and then produces deformation according to the demand.

3. Rotatable wing ribs

In 1998, H.P.Monner. Et al. proposed the concept of "rotatable wing rib" mechanism. The trailing edge of the wing equipped with the ribs can achieve seamless and continuous flexible deflection. The rib is composed of multiple subunits, similar to a connecting rod mechanism. Then, S.Ricci etc. according to the concept of "rotatable wing rib" mechanism, designed a wing-box model with a chord length of 0.6m and a span of 1.5m with four single-acting deflectable trailing-edge ribs arranged equally along the span of the wing. And then, Yang Zhichun et al. also carried out a detailed motion analysis of the rotatable wing rib. The rotatable rib is composed of more subunits, each subunits is composed of two plate and one within plate, its structure as shown in figure 1.24.

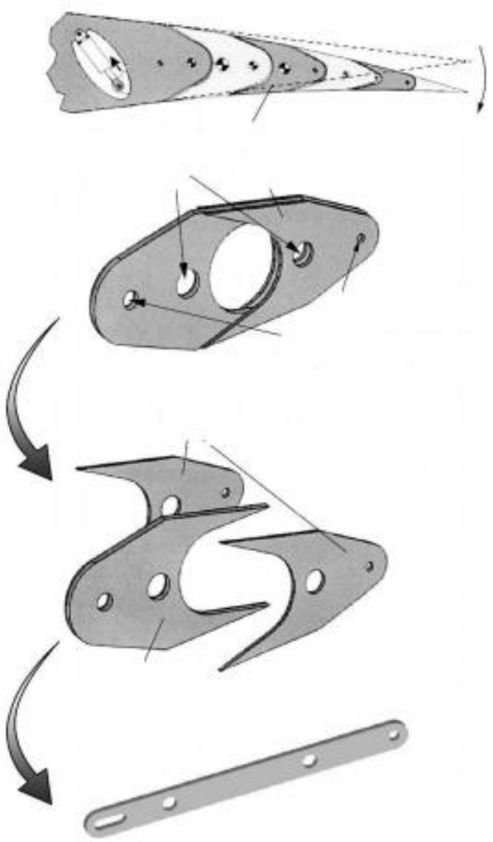


Figure 1.24 — Concept of rotating rib [20]

Therefore, the flexible part of the trailing edge of the wing can be deformed better by using the rotatable wing ribs.

2 Economic Section

2.1 Specific advantages of the designed aircraft and evaluation of possible sales in the markets

2.1.1 Characteristics of tactical and technical indicators of the designed aircraft as a commodity product

The main indicators affecting economic indicators are:

—takeoff mass of the aircraft: m= 36909 kg;

—payload: $m_{p.load} = 2400kg$

—number of crew members: $n_{crew} = 2$

—board flight-navigation equipment and automatic control of the aircraft provides: automatic and director control (functions"Autopilot" and "Flight Director") in the following flight control modes:

1) automatic speed control (function "Automatic traction") in the modes;

2) category III automatic approach;

3) speed protection functionality to withstand flight speed in the VLS – VMS range, calculated for the current configuration and flight conditions;

– an indication and recording system allows on-board data to be recorded on a standard memory card of the PCMCIA type (removable memory card) and automated wireless transmission of aircraft data on the ground to calculate production and forecast the remaining aircraft resource;

—required run length 1600 m;

—maximum flight Mach number and cruising flight speed M=0.84 and $V_{cruise} = 980km/h.$

—weight of aircraft structure $m_{str} = 10335kg.$

2.1.2 Characteristics of aircraft, which affect the competitive advantages and economic costs in its design in the research and development bureau (RDB) and in serial production at the production aviation enterprise

Designing aircraft has a lower required mass of fuel for the flight with a maximum payload on the flight range L=7000m, $m_{fuel} = 15132kg.$

Seat class and range directly reflect the market positioning of civil aircraft products. For aircraft of the same seat class, shorter design range can reduce the weight and operating cost of the aircraft to some extent, and improve the economy and competitiveness of the aircraft. The longer range allows for better route adaptability and operational flexibility, reducing fleet types and maintenance costs. All factors should be considered comprehensively in the selection of aircraft types, and trade-offs should be made in the economy, adaptability, reliability and other aspects of aircraft. Seat class and range are generally determined according to market demand obtained from market research.

Fuel consumption also has a great impact on aircraft operating economy, and fuel consumption is determined by a variety of factors. In the same positioning and meet the same safety premise, the higher the technical level, the higher the fuel efficiency. When the takeoff weight and commercial load of the aircraft are fixed, the higher the fuel efficiency of the aircraft, the longer the range of the aircraft. Flight speed and cruise lift-drag ratio also affect fuel consumption, and therefore the economy of operating costs.

Other factors that affect the economy of operating cost are maximum take-off weight and maximum zero fuel weight and unused aircraft weight. The maximum take-off weight determines the thrust requirements of the aircraft engine, and the maximum zero oil weight determines the maximum commercial load capacity of the aircraft. The use of empty weight is called invalid load. When the maximum take-off weight is certain, the use of empty weight increases, while the commercial load capacity decreases.

The cost of aircraft design and the positioning of aircraft products often affect the selling price of aircraft, and the difference in the selling price of aircraft will also affect the economy of aircraft operating costs.

Some operating costs of aircraft are relatively fixed and do not vary with the number of flights, such as the annual depreciation cost of the aircraft, insurance costs and crew salaries. These costs are basically fixed in size each year, and the more flights a year, the lower the operating cost is amortized into a single flight. If the reliability of the aircraft is poor, the annual number of flights is significantly lower than the normal design state, but the frequency and cost of maintenance and repair are significantly increased. These factors will significantly affect the operating cost of the aircraft, especially the aircraft manufacturer should ensure and improve the reliability of the aircraft as much as possible during the research and development period.

2.1.3 Sales market analysis based on marketing research and determining the volume of possible sales of finished aircraft

Factors that affect aircraft sales often depend on the type of aircraft, the price of the aircraft and the capabilities of the aircraft.

According to market research, global business jet sales by all companies were around 560 in 2018 and 620 in 2019, an increase of around 12 percent. Therefore, according to the current global market demand for business jets, it is estimated that the annual sales volume of the aircraft designed in this paper is about 50.

2.2 Initial data for the calculation of technical and economic indicators of the efficiency of the aircraft

2.2.1 Data for calculating costs in the experimental design bureau

To calculate the costs in the experimental design bureau, mass characteristics of the aircraft structure and equipment, speed characteristics, namely the maximum flight Mach number and maximum flight speed will be needed.

Mass characteristics of the aircraft taken from this note:

—weight of aircraft structure $m_{str} = 10335kg$.

—weight of equipment $m_{OI.CE} = 1660kg$.

Speed characteristics taken from this note:

$M_{max} = 0.84$, $V_{max} = 1029km/h$.

2.2.2 Data for calculating the costs of mass production of aircraft

To calculate the cost of mass production, you will need the relative masses of the structure of the parts of the aircraft, the annual output of the aircraft, the mass of the crew, power plants, and airframe.

Wing weight: $m_W = 4093kg$

Fuselage mass: $m_F = 3628kg$

Tail weight: $m_{TU} = 713kg$

Landing gear weight: $m_{LG} = 1901kg$

Fuel mass: $m_f = 15132kg$

Mass of the power plant: $m_{PP} = 3691kg$

Mass of the equipment: $m_{EQ} = 3691kg$

Empty mass: $m_{empty} = 17717kg$

Annual aircraft production N=50 machines

The number of engines installed on one aircraft is two.

2.3 Calculation of the main technical and economic indicators of aircraft design and production

2.3.1 Cost calculation research and development bureau

Preliminary consolidated costs are calculated at the stage of the preliminary design. Therefore, separate costs for preliminary design, sketch and working design should be combined in one stage - design. Costs at the stage of transferring documentation to the serial plant can be neglected. Therefore, the calculation of costs is carried out in the following stages:

– designing;
– production of prototypes;
– testing of prototypes and adjustment of design drawings of the aircraft.

The total cost of designing the structure of the aircraft is determined by the formula

$$\mathrm{C_{design}} = 1.5 \cdot m_{\mathrm{gl}}^{0.1} \cdot \frac{\mathrm{G_{(M+1)}}}{a^{(\mathrm{M+1})}},$$

where $\mathrm{C_{design}}$—design costs, thousand dollars;

m_{gl}—mass of glider with equipment, including mass of service load and crew without mass of power plants, tonns, $m_{gl} = 10335kg$=10.335t;

$\mathrm{G_{(M+1)}}$—gamma function = 0.94261;

M is the maximum speed of the designed aircraft in Mach numbers, M=0.84;

a—parameter that determines the probable deviation or the degree of scattering of a random variable, a = 0.0117.

$$C_{\text{design}} = 6403.13 \text{ th. dol.}$$

The cost of manufacturing prototypes of the aircraft

$$C_{manuf} = 1.5 \cdot m_{gl}^{1.237} \cdot V_{max}^{0.699} \cdot N^{0.2} \ ,$$

where C_{manuf}—the cost of manufacturing a prototype, thousand dollars; V_{max}—maximum speed of the aircraft = 1029km / h;

N is the serial number of the prototype.

$$C_{man1} = 3439.25 \ th.dol$$

$$C_{man2} = 3950.66 \ th.dol$$

$$C_{man} = C_{man1} + C_{man2} = 3439.25 + 3950.66 = 7389.91 \ th.dol$$

Costs for aircraft testing and adjustment of aircraft design drawings

$$C_{\text{test}} = 1.5 \cdot n_{\text{fs}} \cdot К_{\text{lr}}[1 + 0.01(n_{\text{fs}} - 1)] \ ,$$

where n_{fs}—the number of flight samples = 1;

$К_{\text{lr}}$—coefficient depending on the length of the runway = 0.80.

$$C_{test} = 12000 \ th.dol$$

Total costs of RDB and the price of the aircraft project:

$$C_{RDB} = C_{\text{design}} + C_{manuf} + C_{\text{test}} = 25793.04 \text{ th.dol}$$

$$Price_{RDB} = C_{RDB} + Profit_{RDB}$$

where $Profit_{RDB}$—the planned profit of RDB (15… 20%).

The sum of the cost of manufacturing two prototypes:

$$Price_{RDB} = C_{RDB} + Profit_{RDB} = 1.15 \times 25793.04 = 29662.00 \ th.dol$$

2.3.2 Calculation of costs for serial production of aircraft, profit and prices of aircraft and engine

Calculation of the production cost of the aircraft:

The production average cost, USD, of one aircraft from the annual program of production of N pieces is defined as follows:

$$C_{\text{Prod}} = C_{\text{M}} + C_{\text{pp}} + C_{\text{se}} + C_{\text{techn}} + C_{\text{W}} + C_{\text{UST}} + \text{GPC}$$

Costs of basic materials, raw materials and purchased semi-finished products (forgings, stampings, castings)

$$C_{\text{M}} = 1.95 \cdot 10^4 \cdot m_{gl}^{0.93} \cdot 0.9^{3.32 \lg N},$$

where N—the annual program of release of aircraft = 50.

$$C_M = 43556.09 \ th.dol$$

Cost of purchased products (on-board equipment and installations, control systems, etc.)

$$C_{\text{pp}} = 1.95 \cdot \left(2.37 \cdot V_{max} + 14.15 \cdot m_{\text{gl}} - 1280\right) \cdot N^{-0.09}.$$

$$C_{pp} = 1789.48 \ th.dol$$

The cost of manufacturing special technological equipment depends on the weight of the glider, the volume of production, maximum speed, the level of use of normalized equipment, the level of continuity of the glider design, the number of installed engines.

The labor intensity of manufacturing a technological set of equipment, million normo—h, Is determined by the formula:

$$\text{T}_s = (2.943 + 0.0775 \cdot m_{gl} - 2.58 \cdot 10^{-4} \cdot m_{gl}^2) \cdot 1.05^n,$$

where n is the number of aircraft engines = 2.

$$T_S = (2.943 + 0.0775 \cdot 10.335 - 2.58 \cdot 10^{-4} \cdot 10.335^2) \cdot 1.05^2 = 4.10$$

Total costs for the manufacture, repair and restoration of special technological equipment

$$\text{T}_{\text{se}} = \text{T}_s \cdot \text{K}_1 \cdot \text{K}_2 \cdot \text{K}_3 \cdot \text{K}_4 \cdot \text{K}_5 \ ,$$

where K_1 is the ratio that takes into account the volume of production:

$$\text{K}_1 = 2.27 \cdot 10^{-3} \cdot N + 0.64 = 0.7535;$$

K_2—coefficient that takes into account the level of application of normalized equipment =1.075;

K_3—coefficient that takes into account the level of continuity of the created structure =0.998;

K_4—coefficient that takes into account the manufacture of equipment backups, its repair and restoration =1.51;

K_5—factor that takes into account the type of aircraft vertical takeoff and landing with a horizontal position of the fuselage =1.12.

$$T_{se} = 4.10 \cdot 0.7535 \cdot 1.075 \cdot 0.998 \cdot 1.51 \cdot 1.12 = 5.61$$

The amount of costs for the manufacture of special equipment (total)

$$\text{Total C}_{\text{se}} = \text{T}_{se} \cdot \bar{\text{C}} = 5.61 \cdot 2.5 = 14.025\ mil.dol\ ,$$

where $\bar{\text{C}}$ - the cost of one standard time of production of special technological equipment, equal to 2.5 - 2.7 dollars.

Then the cost of manufacturing technological equipment per aircraft, thousand dollars, is calculated by the formula

$$C_{se} = \frac{Total\ C_{se}}{N_1 + N_2} = \frac{14.025}{50 + 50} = 0.14025\ th.dol$$

where Total C_{se}—the total cost of manufacturing special equipment, thousand dollars; N_1 , N_2—aircraft production programs for the first two years =50.

The amount of production costs is determined as a percentage of the cost of special technological equipment[2].

The cost of wages (basic and additional), accrued for the direct manufacture of the aircraft, USD, is determined by the expression

$$C_w = 1.5 \cdot 3.013 \cdot m_{gl}^{0.903} \cdot \text{M}^{0.42} \cdot \text{N}^{-0.32} \cdot \text{K}_{lp} = 99849.67\ th.dol$$

where K_{lp} is the coefficient that takes into account the increase in labor productivity of workers in the development of aircraft production for five years for fighters and seven years for passenger aircraft ($\text{K}_{lp} = \ 1.08^t$)

The amount of unified social tax (UST), which depend on the labor costs of workers of all categories of industrial personnel, according to the annual production program, USD:

$$C_{UST} = \frac{l_{UST} \cdot Cw \cdot \text{N}}{100} = 10983.46\ th.dol$$

where l_{UST} – Unified social tax (according to the current legislation are mandatory from 01.01.2016, the UST rate is 22% for all categories of payers).

General production costs (oveheads) for the organization of production and management of shops, sections, departments and other units of primary and secondary production, as well as costs for maintenance and operation of machinery and equipment are aggregated:

$$GPC = 1.92 \cdot 3.701 \cdot Cw \cdot \mathrm{N}^{-0.359} = 174195.41\ th.dol$$

The sum of all these costs will be the average production cost of the aircraft.

$$C_{Prod} = C_M + C_{pp} + C_{se} + C_W + C_{UST} + GPC = 330374.25 \text{ th.dol}$$

Calculation of the full cost of the aircraft:

Sales costs include costs associated with the sale of products (aircraft).

Expenditures on sales can be planned in the amount of 1.5% of production cost:

$$C_{SALES} = C_{prod} \cdot 0.015 = 330374.25 \cdot 0.015 = 4955.61 \text{ th.dol}$$

Then the estimated total average cost of the aircraft, which corresponds to the annual program of production of N aircraft:

$$C_{full} = C_{Prod} + C_{SALES} = 330374.25 + 4955.61 = 335329.86 \text{ th.dol}$$

Calculation of profit and price of the aircraft without engines:

The estimated profit from one aircraft without taking into account the cost of engines, dollars, can be determined using the coefficient of planned profit (profitability) as follows:

$$Profit_{aircraft} = \frac{R_{pr} \cdot C_{full}}{100} = \frac{0.15 \cdot 335329.86}{100} = 502.99\ th.dol$$

where R_{pr} – the profitability ratio recommended for sale in pure competition markets is 15… 25%.

Estimated average price of the aircraft without engines, USD:

$$Price_{aircraft} = C_{full} + Profit_{aircraft} + VAT = 336915.62\ th.dol$$

where VAT is the value added tax of the aircraft (state tax to the state budget).

Calculation of the cost of an aircraft with engines:

The cost of turboprop engine, USD, on average

$$Price_{engine} = 61.183 \cdot \mathrm{N_{Emax}} = 1640\ th.dol$$

where $\mathrm{N_{Emax}}$ - take-off power of one engine, kW.

The cost of series-developed engines for power plants of vertical takeoff and landing aircraft with a horizontal position of the fuselage are shown in table 3.6[2].

Then the estimated average price of one aircraft with engines

$$Price_{aircraft+engine} = Price_{aircraft} + Price_{engine} = 336915.62 + 1640 = 338555.62\ th.dol$$

2.3.3 Determining the break-even point of aircraft production

Break-even point — this is the volume of production and sales, when the company's income is equal to its costs, and the company has no profit or loss. Synonyms of the concept of break-even point are the point of profitability, the point of critical sales.

Break-even production can be determined in kind and in monetary terms. Natural expression is more acceptable for single-product production.

Thus, to calculate the planned value of production and sales, which corresponds to the break-even state of the enterprise, you need to know three values:

– wholesale price of goods;

– the amount of fixed costs (FC), ie such costs, the amount of which does not depend on the volume of production and sales and their changes (costs of equipment, its maintenance and operation, depreciation, administrative costs, social insurance, research, development, etc.);

– the amount of variable costs (VC), ie such costs that change their value due to changes in production and sales (costs of raw materials, wages of key production personnel, electricity, transportation, etc.).

Analytically break-even output (point) (annual critical release program) can be calculated by the formula

$$BEO = \frac{FCannual}{Price - VC} \approx 9$$

where FC_{annual} — annual fixed costs; Price- unit price of the product; VC — variable costs per unit of product.

Conclusions: to evaluate the economic efficiency of the proposed aircraft project.

2.4 Conclusions

It is expected to produce 50 aircraft a year at a total cost of around 338,555.62 th.dol. The break-even point for the aircraft is about 9. And the aircrafts are mainly used for trade between Ukraine and Beijing of China. Therefore, the aircraft can be sold to large-scale commercial companies to achieve a profit.

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