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: ***«Design of Composite Wing Structure of the Light Aircraft»***

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Applicant of 2 Year 161F Group

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

Факультет *Літакобудування*
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Рівень вищої освіти *другий (магістерський)*
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(найменування)

ЗАТВЕРДЖУЮ
Завідувач кафедри

к.т.н., доц. Андрій ГУМЕННИЙ
"____" ____________202___року

ЗАВДАННЯ
НА КВАЛІФІКАЦІЙНУ РОБОТ

Ло Веньхан
(прізвище, ім’я та по батькові)

1.Тема кваліфікаційної роботи ***Design of Composite Wing Structure of the Light Aircraft (Проектування конструкції композитного крила легкого літака)***

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} = 1 person, n_{pass} = 1 person, V_{CR} = 260 km/h
H_{CR} = 2400 m, $L_{TO.}$ = 0.5 km, $L_{(mp=max)}$ = 1800 km

Зміст пояснювальної записки (перелік завдань, які потрібно розв’язати)

ABSTRACT
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2. ECONOMIC SECTION

- 2.1. Initial data.
- 2.2. Social Measures.
- 2.3. Management costs.
- 2.4. Depreciation of equipment.
- 2.5. Total cost of research.
- 2.6. Cost profit.
- 2.7. Wholesale price.
- 2.8. Results of calculations.
- 2.9. Conclusion of economic section.

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)

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1. КАЛЕНДАРНИЙ ПЛАН

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

Здобувач _____________ **Luo Wenhang**

(підпис) (ім'я та прізвище)

Керівник кваліфікаційної роботи

_____________ **Лілія БУЙВАЛ**

(підпис) (ім'я та прізвище)

ABSTRACT

The explanatory note contains: 77 p., 11 Tables, 50 Fig., 32 References

Object of research: light aircraft

The purpose of the work: Develop technical requirements specification, research statistical data analysis of scientists, who dealt with vehicle and what issues they claim and give the rationale for the choice of the scientific research of light aircraft area, by referring to factual data from the prototype, citing its advantages and strengths, it is clear that the advantages of the design of this project's light aircraft, based on theoretical experience, lie in its long range, compact size, good stability, and the proposal of a plan to increase the strength of the wing structure by using composite material layers.

Research methods: statistical, analytical methods, 3D modelling methods.

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

1. Designed light aircraft belongs to normal category aeroplanes, it complies with Certification Specifications for Normal Category Aero planes (CS-23). Implementing rules for the airworthiness and environmental certification of aircraft and related products, parts and appliances, as well as for the certification of design and production organizations (recast) Text with EEA relevance.

2. The propeller provides the required power, it can accommodate 2 passengers including the pilot, and carry a certain weight of luggage, and it is designed for vacation travel and business flights.

3. Designed light aircraft the expected maximum flight altitude is 3500 m, and a maximum range of 1800 km, a maximum speed of 290 km/h, a cruise speed of 260 km/h, the cruising altitude is 2400 m, the stall speed is 75 km/h, the takeoff run distance is 500 m.

3. Siemens NX is used to create the model of the whole designed light aircraft.

4. Composite materials have been widely used in aerospace and aviation structures due to their high specific strength and stiffness. In particular, they have significantly increased their application in various new aircraft developed in recent years. Composite laminates are the most commonly used structural form in composite materials, offering excellent structural design capabilities. Honeycomb sandwich structures, as a special form of structure, have found extensive applications in aerospace and aviation structures. They have become two important composite material structural forms in this field along with composite laminates, in this light aircraft, composite materials are used in wing design and gave a proposal.

4. Unidirectional carbon fiber/epoxy prepreg and Nomex aramid honeycomb, model NRH-3-48 are chosen to use in the wing structure, the beam is I beam, and different parts have the different schemes to lay up. The concrete scheme and data is shown in the project.

LIGHT AIRCRAFT, COMPOSIT MATERIAL, 3D MODELLING, CARBON, ANALYSIS, SANDWICH PANEL.

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 term "Light Aircraft" is commonly used to refer to small airplanes with a maximum takeoff weight of 5670kg or less, which are often used in the private and business aviation sectors. These aircraft are typically powered by a single or twin-engine turboprop or small jet engines.

Light aircraft can be used for a variety of purposes, including private flying, commercial flying, flight training, cargo transportation, sightseeing tours, and more. They are often able to take off and land on shorter runways, and fly at lower altitudes.

This project takes the Cessna Light aircrafts as prototypes, based on its overall design, aerodynamic requirements to design a relatively cheap and useful new light aircraft. At the same time, according to the theory and methods of composite aircraft structure design, a new structural design scheme is proposed for the wing structure of this aircraft, aiming to meet the strength requirements of the wing structure while reducing weight.

1.1.2 Aircraft Assignment

After conducting a data investigation and analyzing the similarities and differences of several types of Cessna light aircraft, the parameters and characteristics of the designed light aircraft are determined.

The range of an aircraft is determined by several factors, including its fuel capacity, fuel efficiency, weight, and environmental conditions. The farthest distance between two cities in Ukraine is from Kharkiv to Lviv, the straight-line distance is 880 kilometers, need to be within the maximum range of the light aircraft.

The use of composite materials to manufacture lightweight aircraft wings has the characteristics including: Lightweight; High strength; High corrosion resistance; Good plasticity; Low maintenance costs. Therefore, using composite materials to manufacture lightweight aircraft wings can improve the performance, safety, and lifespan of the aircraft, while reducing flight and maintenance costs, which is a very reasonable choice.

Selecting composite materials and performing laminate design requires consideration of the operating environment and requirements, and the selection of suitable composite materials based on desired performance and characteristics. Precise laminate design can ensure the performance, strength, and weight requirements of the aircraft wing. Finally, the performance and reliability of the design can be verified through simulation and testing. This project proposes one scheme of laminate design and didn’t do the last step.

1.1.3 The basis for the development

This aircraft is designed to carry 2 pilots/passengers, so this light aircraft meets the certification specifications for a normal aircraft [1]:

- International Standard Atmosphere.
- EASA CS-23: EASA CS-23 is a standard issued by the European Aviation Safety Agency, applicable to the design and manufacture of light aircraft (MTOM ≤ 5700 kg). The standard specifies the requirements for the structure, systems, materials, performance, flight testing, and certification of light aircraft.
- FAA Part 23: FAA Part 23 is a standard issued by the Federal Aviation Administration (FAA), similar to EASA CS-23, applicable to the design and manufacture of light aircraft (MTOM ≤ 5700 kg). The standard requires that the aircraft's structure, systems, materials, performance, flight testing, and certification meet the regulations.
- ASTM F44: ASTM F44 is a standard issued by the ASTM International organization in the United States, applicable to the design, manufacture, and maintenance of light aircraft. The standard covers multiple aspects, including materials, structures, systems, electrical and mechanical equipment, cockpit and cabin, environment, and ergonomics.
- ISO 23383: ISO 23383 is a standard issued by the International Organization for Standardization (ISO), applicable to the design, manufacture, and maintenance of light aircraft. The standard requires that light aircraft meet requirements in safety, reliability, environmental protection, and economy, to ensure high levels of safety and performance in use.
- MIL-STD-1797B: MIL-STD-1797B is a standard issued by the US Department of Defense, applicable to the design and manufacture of military light aircraft. The standard specifies the requirements for military light aircraft, including structure, systems, materials, performance, flight testing, and certification.
- SAE ARP 4754: SAE ARP 4754 is an international standard issued by the Society of Automotive Engineers (SAE), applicable to the development processes of aircraft and systems. The standard covers the entire aircraft development process, including system requirements, functional requirements, design, implementation, verification, validation, and certification. While it is not specific to light aircraft, it can be applied to the design of light aircraft as well.

1.1.4 Expected Operational Conditions

Environmental characteristics and factors:

a. Barometric pressure – for all range of flight altitudes in accordance with International Standard Atmosphere; on ground PH = +20 mm HG.

b. The temperature of ambient air (t_{AA}) is determined based on the temperature change with altitude, in accordance with International Standard Atmosphere. The temperature deviation from the mean value for different altitudes is represented by the lines "Minimum Arctic" and "Maximum Tropical". The aircraft and its systems must be able to operate within this temperature range, even after the aircraft has been on the ground at a t_{AA} as low as minus 60℃.

c. Relative humidity on the ground is of 98% at an ambient air which the temperature is +35℃.

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 (Runway):

- 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 conditions ($\mu \geq 0.6$), reduction of lateral wind component limiting values is allowable.

Operational factors:

a. Crew: pilot.

b. RWY type – A road with no obstacles and a width of not less than 8 meters.

c. No altitude and altitude restrictions for take-off and landing.

d. Condition of paved RWY:

- dry;
- wet;
- occupied by water;
- Covered with snow up to 10 cm thick.

e. Characteristics of unpaved RWY:

- with or without turf;
- loamy, clay;
- sandy, sandy-loam;
- rocky, crushed stone;
- chernozem soil.

f. 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 the daytime and in the night;
- above plain and mountain surface of local airlines;
- above water distanced from the ground for up to 30 min. of flight;
- range of geographical latitudes: up to 70°north and 55°south latitude.

g. 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).

h. Components and characteristics of ground facilities for flight provisions in accordance with civil aviation considering.

i. Maintenance periodicity is equal to 300 flight hours of the airframe.

j. Service life and lifetime for:

1. aircraft (up to discarding): Landings 50000; flight hours 40000; service life 25 years.
2. engine: assigned service life 20000; life to first overhaul and life between overhauls 6000.
3. vendor items – as a rule the vendor items' service life shall correspond to the aircraft service life or there shall be multiple repair periodicity.

The aircraft is pre-programmed to take off from normal airport runways and has a maximum range of 1800 km, a maximum speed of 290 km/h.

A cruise speed of 260 km/h. The stall speed depends on several factors, such as aircraft weight, payload, altitude, temperature, and more. Generally, the stall speed (with flaps up) at maximum takeoff weight (MTOW) is around 75 km/h.

The minimum standard for takeoff is determined based on aircraft performance and the regulations of the departure airport, taking into account various factors. In general, the minimum standard for takeoff should meet the following requirements: a cloud ceiling of 300 meters or more to ensure that the pilot can maintain visual contact during takeoff; a visibility of 1800 meters or more to ensure that the pilot can see the takeoff path and the surrounding environment; and a takeoff run distance that depends on factors such as aircraft weight, altitude, temperature, wind direction, etc., Generally, the takeoff run distance is 500m.

The Minimum Descent Altitude/Height (MDA/H) refers to the altitude/height calculated by rounding up the Overrunning Obstacle Altitude/Overrunning Obstacle Height (OCA/OCH) determined by a specific instrument approach procedure for non-precision approaches using certain navigation equipment. This calculation takes into consideration various factors, and the final result is rounded up and then an additional 5 meters is added.

1.1.5 General Requirements

This light aircraft the expected maximum flight altitude is 3500 m, and a maximum range of 1800 km, a maximum speed of 290 km/h, a cruise speed of 260 km/h, the cruising altitude is 2400 m, the stall speed is 75 km/h, the takeoff run distance is 500 m.

The aircraft, its engines, equipment and other parts, and operational publications shall meet the following requirements:

Engine – AMC & GM to CS-23.903(a)(1):

AMC & GM refer to Acceptable Means of Compliance and Guidance Material, respectively. They are provided by the European Union Aviation Safety Agency (EASA) to assist in meeting the regulatory requirements set out in the CS-23 certification specifications.

CS-23.903(a)(1) is a specific section within the CS-23 regulations that pertains to the design requirements for flight instruments. It requires that flight instruments be designed to function properly under all foreseeable flight and ground conditions, with appropriate markings and indications that are clear and easily readable.

AMC & GM to CS-23.903(a)(1) provide guidance on how to comply with this regulation, including recommendations on the design and testing of flight instruments. The AMC provides specific methods for showing compliance with the regulation, while the GM provides additional information and best practices for meeting the requirements.

Specific aircraft structural requirements:

a. The design and manufacture of aircraft should follow the principle of "fail safe structure".

b. The weight layout and center of gravity of the aircraft shall ensure operation with all and a small number of passengers in all possible versions of loading operation according to the instructions for loading and center of gravity without the use of ballast. The ground tail center of gravity limit is not less than 5% of the MAC.

Wing design requirements [2]:

a. Proof of strength: Limit load tests of control surfaces are required. These tests must include the horn or fitting to which the control system is attached; In structural analyses, rigging loads due to wire bracing must be accounted for in a rational or conservative manner.

b. Installation: Movable surfaces must be installed so that there is no interference between any surfaces, their bracing or adjacent fixed structure, when one surface is held in its most critical clearance positions and the others are operated through their full movement; If an adjustable stabilizer is used, it must have stops that will limit its range of travel to that allowing safe flight and landing.

c. Hinges: Control surface hinges, except ball and roller bearing hinges, must have a factor of safety of not less than 6·67 with respect to the ultimate bearing strength of the softest material used as a bearing; For ball or roller bearing hinges, the approved rating of the bearing may not be exceeded.

Wing flap design requirements:

a. Wing flap controls: Each wing flap control must be designed so that, when the flap has been placed in any position upon which compliance with the performance requirements of CS-23 is based, the flap will not move from that position unless the control is adjusted or is moved by the automatic operation of a flap load limiting device; The rate of movement of the flaps in response to the operation of the pilot’s control or automatic device must give satisfactory flight and performance characteristics under steady or changing conditions of airspeed, engine power and attitude; If compliance with CS 23.145 (b) (3) necessitates wing flap retraction to positions that are not fully retracted, the wing flap control lever settings corresponding to those positions must be positively located such that a definite change of direction of movement of the lever is necessary to select settings beyond those settings.

b. Wing flap position indicator: Flap installations with only the retracted and fully extended position, unless a direct operating mechanism provides a sense of “feel” and position (such as when a mechanical linkage is employed; or the flap position is readily determined without seriously detracting from other piloting duties under any flight condition, day or night; and flap installation with intermediate flap positions if any flap position other than retracted or fully extended is used to show compliance with the performance requirements of CS-23; and the flap installation does not meet the requirements of sub-paragraph (a) (1) .

c. Flap interconnection: The main wing flaps and related movable surfaces as a system must be synchronized by a mechanical interconnection between the movable flap surfaces that is independent of the flap drive system or by an approved equivalent means and be designed so that the occurrence of any failure of the flap system that would result in an unsafe flight characteristic of the aero plane is extremely improbable; or the aero plane must be shown to have safe flight characteristics with any combination of extreme positions of individual movable surfaces (mechanically interconnected surfaces are to be considered as a single surface); If an interconnection is used in twin-engine aero planes, it must be designed to account for the unsymmetrical loads resulting from flight with the engine on one side of the plane of symmetry inoperative and the remaining engine at take-off power. For single-engine aero planes and twin-engine aero planes with no slipstream effects on the flaps, it may be assumed that 100% of the critical air load acts on one side and 70% on the other.

Functional system and equipment requirements:

a. Aircraft control system: While using the braking devices in control systems and wing high-lift devices it is necessary to provide stability of brake torques of these devices during operation.

b. Aircraft hydraulic system and landing gear:

- Purity of operating fluid of hydraulic system should be ensured by onboard filtration means.
- The filters should be designed to provide operation with contamination monitoring on pre-failure condition.
- Provision should be made to exclude the possibility of air lock formation in hydraulic system suction lines after long-time parking of an aircraft, or to ensure their fast removal.
- When the system loses operating fluid, it is necessary to prevent contamination of hydraulic system by main pressurizing source.

- Hydraulic system control units activated during system operation should be located near the appropriate measuring equipment.
- Provision should be made in the design of a hydraulic system to remove filtering elements and components with the service life less than the life time of airframe without operating fluid drain.
- The visual monitoring of filling of capacities used to collect the drainage leakages should be provided.
- While towing with disabled APU the wheel braking should be provided.

c. Air Conditioning System (ACS):

- During all phases of flight, the air temperature in the aircraft cabin should be determined and maintained within the range of 17 to 25 degrees Celsius. Assuming ground preparation, the specified temperature value should be reached within 20 minutes of takeoff.
- The operation of the ACS within the operating parameters and conditions should not result in a change in chamber pressure.
- Cabin airflow at the head level of the seated passenger should not exceed 0.4m/s.
- The operation of the ACS in normal operating mode should not depend on the operation of other systems using the same source of compressed air as the ACS.

d. Power plant:

- The engines typically used in light aircraft include piston engines and light turboprop engines. Piston engines can be diesel, gasoline or aviation gasoline engines, while light turboprop engines are turbine engines with a combustion chamber located inside the engine.
- For piston engines, high reliability, low maintenance costs, and good durability are generally required. The output power and torque of the engine need to be adapted to the design and operation of the aircraft, while also having quick response capabilities to ensure the safety of the aircraft in emergencies. In addition, for piston engines, certain fuel efficiency requirements must be met to reduce flight costs.
- For light turboprop engines, high power output, reliability and durability are required, while also being lightweight and easy to maintain. Due to the high compression ratio and fuel injection technology adopted by turboprop engines, higher fuel efficiency can be obtained, which is also an advantage when applied to light aircraft. Turboprop engines also have the advantages of

quick response and high automation, making them suitable for flight missions that require quick takeoff and frequent speed changes.

- For this project, a two-seater aircraft with a low speed, piston engines are more appropriate.

e. Electrical equipment:

- The variable-frequency 115/200 V AC system shall be used as primary electrical system. Electronic converters shall be used as 115/200 AC constant-frequency 400 Hz electric power source.
- The 27 V DC system should be used as secondary electrical system. Rectifiers shall be used as electric power source.
- Built in test should be provided in generation systems to ensure: troubleshooting ensuring accuracy to LRU with an index of monitoring entirety and coefficient of resolution of fault location not less than 0.95; Operation of all electrical system units (including generator) up to safe failure.
- The electric power nickel-cadmium storage batteries and converters should be used as emergency sources.
- Monitoring of critical state of storage batteries should be provided.
- Quality of power in all systems should correspond to State Standard ГОСТ 19705-89.

f. External lights:

The following lights should be installed on an aircraft:

- Navigational lights with halogen lights (single);
- Fuselage light beacons of white radiation chromaticity with reduction of light intensity to 10 % from maximum value;
- Landing and taxi lights;
- Illumination flood lights of stabilizer structure for visual definition of character of an ice formation.
- The navigation lights serviceability warning system and beacons should be provided on aircraft.

g. Communications:

The aircraft communication equipment should ensure fulfillment of following tasks:

- Two-way communication within the limits of direct radio visibility with traffic control service of each airfields of take-off or landing and in which control area the aircraft is located;
- Two-way communication at any moment of flight at least with one ground radio station for lines equipped so to ensure continuous radio communication in microwave frequency band;
- Reception of meteorological reports or special notices transmitted by meteorological services or air control services of aerodromes on flight course at any moment of flight;
- Operative communication between all crew members at any moment of flight;
- Warning of passengers in flight;
- Communication with ground-based systems for cases when the aircraft landed outside of aerodrome or supply of signal to lead search and rescue services.

h. Fire protection:

The design means ensuring survivability of power plant structure in case of fire on engine, and also wing protection against flame exposure should be developed.

Environmental protection requirements:

a. Noise: Light aircraft must comply with noise level restrictions, which limit the amount of noise they can generate during takeoff, landing, and flight. The noise level limits are typically established by national or regional aviation authorities.

b. Emissions: Light aircraft must meet certain emissions standards, which limit the number of pollutants they can release into the atmosphere. This includes requirements for exhaust emissions and fuel consumption.

c. Fuel: Light aircraft must use aviation fuel that meets certain quality standards, to ensure that it does not contain harmful substances that could damage the engine or pollute the environment.

d. Waste: Light aircraft operators must dispose of waste materials such as oil, hydraulic fluids, and cleaning solvents in an environmentally responsible manner, in accordance with local regulations.

e. Wildlife protection: Light aircraft operators must take steps to minimize the impact of their operations on wildlife, including avoiding flying over protected areas, minimizing noise and disturbances, and complying with any wildlife-related regulations.

Requirements for maintainability, testability, technical diagnosis and repair means:

a. General Requirements for Maintenance and Overhaul Systems (MO). Maintenance means should support the airworthiness of the aircraft throughout its useful life. Scheduled maintenance procedures should be based on MSG-3 technology.

- The MO system, reliability, testability and maintainability shall ensure that the annual average flight time of each aircraft under the expected operating conditions is not less than 2500-3000 flight hours.
- The overall specific operational man-hours of the aircraft MO shall not exceed 8 hours of flight time, including the man-hours for troubleshooting, corrective actions and replacement parts for an average flight time of 1.5 hours.
- The overall specific operating time of the MO shall not exceed the flight time of 0.75 man-hours.
- The average engine replacement time should not exceed 2.5 hours.
- 25 minutes for any wheel change.
- Other operational technical data for testability, manufacturability and reliability shall comply with the "General requirements for operational technical data of CA aircraft" authorized by MAP-MCA from 15.12.90.
- The aircraft and its systems should be adapted to the operating conditions.
- Installation, removal, adjustment, maintenance of items should be simple enough to be carried out by the operator's staff.

b. Maintenance conditions:

- Periodic maintenance is implemented in stages, it is necessary to provide system maintenance and partition maintenance.
- Execute announcement operations, regularly replace items and operate maintenance should be regularly maintained.
- Failures should be eliminated during operational maintenance to ensure the flight effectiveness of the aircraft during maintenance. It is necessary to set all kinds of the operating constraints causing necessity of fulfillment off-schedule maintenance.

- Operative maintenance and troubleshooting should be ensured on open parking in all climatic zones.
- Organizational structure of aviation-engineering service, requirements for personnel, safety precautions and flammability control during MO are set according to effective "Instructions on Aeronautical Engineering Operation and Maintenance for Civil Aviation".
- Aircraft MO plan should be based on monitoring of a condition of systems, articles and equipment on scheduled kinds of MO by results of which the systems, articles and equipment are recovered.
- Periodic maintenance should comprise the base form of a constant maintenance procedures and additional procedures whose are aliquot to base form and executed on aircraft operating time (service life) counted off from the maintenance start time.
- Lists of failures and faults allowable for aircraft departure in flight or operation up to the next form of maintenance should be developed.
- PC-based technological ground operational complex (TGOC) should be created over the aircraft development to ensure information support function of maintenance in CA divisions. TGOC development should be conducted according to separate performance specification agreed with the Customer.
- While fulfilling operative maintenance forms the ground test means should not be applied.

Flight safety:

a. Doors, hatches and access doors shall be designed to be locked or sealed.

b. Wired communication between ground service and pilot should be provided.

c. Stall warning.

There must be a clear and distinctive stall warning, with the flaps and landing gear in any normal position, in straight and turning flight.

The stall warning may be furnished either through the inherent aerodynamic qualities of the aero plane or by a device that will give clearly distinguishable indications under expected conditions of flight. However, a visual stall warning device that requires the attention of the crew within the cockpit is not acceptable by itself.

During the stall tests required by CS 23.201 (b) and CS 23.203 (a) (1), the stall warning must begin at a speed exceeding the stalling speed by a margin of not less than 9.3 km/h (5 knots) and must continue until the stall occurs.

When following the procedures of CS 23.1585, the stall warning must not occur during a take-off with all engines operating, a take-off continued with one engine inoperative or during an approach to landing.

During the stall tests required by CS 23.203 (a) (2), the stall warning must begin sufficiently in advance of the stall for the stall to be averted by pilot action taken after the stall warning first occurs.

d. fall into the sea.

The wings and the cabin can make the flying car float on the water. The sealing system of the flying car can ensure that the sea water will not penetrate into the car. The oxygen system provides the oxygen needed by the driver and passengers in the car. The built-in satellite positioning system can send to the location for rescue.

Escape system:

The flying car is planned to be equipped with a parachute capable of carrying the entire aircraft, so that passengers and pilots can safely return to the ground in the event of an emergency.

Ice protection:

If certification with ice protection provisions is desired, compliance with the following requirements must be shown:

a. The recommended procedures for the use of the ice protection equipment must be set forth in the Aero plane Flight Manual or in approved manual material.
b. An analysis must be performed to establish, on the basis of the aero plane’s operational needs, the adequacy of the ice protection system for the various components of the aero plane. In addition, tests of the ice protection system must be conducted to demonstrate that the aero plane is capable of operating safely in continuous maximum and intermittent maximum icing conditions as described in AMC-1.
c. Compliance with all or portions may be accomplished by reference, where applicable because of similarity of the designs to analysis and tests performed for the type certification of a type certificated aircraft.
d. When monitoring of the external surfaces of the aero plane by the flight crew is required for proper operation of the ice protection equipment, external lighting must be provided which is adequate to enable the monitoring to be done at night.

Besides all of these, Ground facilities and repair equipment shall correspond to this performance specification; Simulators and training devices should be designed for aircraft according to individual performance specifications. The programs for training of flight and technical staff should be developed up to completion of certification tests; Processing and analysis of flight data using the ground personal computer shall be provided to control the correctness of maintaining of preset flight modes and the pilot technique, to evaluate the pilots' professional level, technical state of the aircraft, its equipment and functional systems in monitoring of operation conditions within life time limits.

1.2 Statistical data analysis

Data analysis refers to the use of appropriate statistical analysis methods to analyze a large amount of data collected, so as to extract useful information, form conclusions, and conduct more detailed research and summary of the data.

Data analysis helps people make judgments so they can take appropriate action. And data analysis is the process of organizing and destination data collection and analysis to make it information. This process is the supporting process of the quality management system.

Result of statistical analysis gives initial data for research investigated object and control intermediate stages to get result with high quality and as minimum as possible error.

1.2.1 Scientists and issues they claim

Light aircraft, also known as small airplanes, are aircraft with a maximum takeoff weight of 5600 kg or less [3]. They are typically powered by piston engines, turboprop engines, or small jet engines. Light aircraft are commonly used for a variety of purposes, including private flying, commercial flying, flight training, cargo transportation, sightseeing tours, and more.

One of the key benefits of light aircraft is their versatility. They are able to take off and land on shorter runways, which allows for increased access to more remote destinations. Additionally, they are typically able to fly at lower altitudes, providing a more immersive and enjoyable flying experience.

Due to their relatively low operating and maintenance costs, light aircraft are a popular choice for private pilots and small businesses. They are often used for personal travel, short-haul flights, and regional transportation.

Light aircraft come in various designs and configurations to suit different needs and preferences. Some common types include single-engine piston airplanes, turboprop airplanes, and small jet airplanes. The Cessna 172 and the Beechcraft Bonanza are examples of popular light aircraft models.

In terms of safety, light aircraft are subject to rigorous maintenance and inspection requirements, and are equipped with advanced avionics systems to assist pilots in navigation and communication. Pilots are required to undergo extensive training and licensing to ensure safe and responsible operation of these aircraft.

Overall, light aircraft play an important role in the aviation industry, providing efficient and accessible transportation options for private individuals and businesses alike.

Table 1.1 shows some light aircraft scientists and the aircraft they have designed.

Table 1.1 – Light aircraft scientists

Scientist	Status	Aircraft	Year of design
Clyde Cessna	Founder of Cessna Aircraft Company	Cessna Model A	1927
Walter Beech	Co-founder of Beech Aircraft Corporation	Beechcraft Model 17 "Stagger wing"	1932
William T. Piper	Founder of Piper Aircraft	Piper J-3 Cub	1937
Ted Smith	Founder of Aero Commander	Aero Commander 500	1948
Jim Bede	Founder of Bede Aircraft	Bede BD-5	1970
Roy Lopresti	Designer of the Grumman American AA-5 series	AA-5B Tiger	1974
Tom Hamilton	Founder of Glasair Aviation	Glasair I	1979
Chris Heintz	Designer of the Zenith Aircraft Company CH series	Zenith CH 701	1986
Richard VanGrunsven	Founder of Van's Aircraft	RV-6	1986
Dan Johnson	Founder of the Light Aircraft Manufacturers Association	Flight Design CT	1997

1.2.2 Light aircraft data analysis

As this project uses the Cessna Light aircraft as prototypes, the table 1.2 presents common light aircraft designed by Cessna in chronological order, with key parameters including seats, engine, max takeoff weight, service ceiling, cruise speed, range, geometrical parameters, type of airfoil, wing span, wing area, aspect ratio and so on, which laid the foundation for the subsequent design.

By referring to factual data from the prototype, citing its advantages and strengths, it is clear that the advantages of the design of this project's light aircraft, based on theoretical experience, lie in its long range, compact size, good stability, and the proposal of a plan to increase the strength of the wing structure by using composite material layers.

Table 1.2 – Common Cessna light aircraft models

Name	Main Parameter	Production	Year of manufacture
Cessna 150	Seats: 2; Engine: 1 x Continental O-200 100 hp; Max takeoff weight: 725 kg; Service ceiling: 4,267 m; Cruise speed: 137 km/h, Range: 776 km; Length: 7.32m, Height: 2.03m, Width: 1.07m; Type of airfoil: NACA2412, Aspect ratio: 6.12; Wing span: 8.23m, Wing area:13.6m^2;	23839	1958 – 1977
Cessna 172 Skyhawk	Seats: 4; Engine: 1 x Lycoming O-360 180 hp; Max takeoff weight: 1,111 kg; Service ceiling: 4,267 m; Cruise speed: 226 km/h, Range: 1,290 km; Length: 8.28m, Height: 2.72m, Width: 1.08m; Type of airfoil: NACA2412, Aspect ratio: 7.32; Wing span: 11m, Wing area:16.2m^2;	44000+	1955 – present
Cessna 182 Sky Lane	Seats: 4 – 6; Engine: 1 x Lycoming IO-540 230 hp; Max takeoff weight: 1,406 kg; Service ceiling: 5,517 m; Cruise speed: 268 km/h, Range: 1,500 km; Length: 8.84m, Height: 2.82m, Width: 1.1m; Type of airfoil: NACA2412, Aspect ratio: 7.87; Wing span: 10.9m, Wing area:16.2m^2;	23237	1956 – present
Cessna 177RG	Seats: 4; Engine: 1 x Pratt & Whitney Canada IO-360-A1B6, 200 horsepower 200 hp; Max takeoff weight: 1,225 kg; Service ceiling: 4,572 m Cruise speed: 200km/h, Range: 1389 km; Length: 8.56m, Height: 2.72m, Width: 1.8m; Type of airfoil: NACA 63A215, Aspect ratio: 8.16; Wing span: 10.97m, Wing area:15.5m^2.	1540	1971 – 1789

Cessna 150

The Cessna 150 [4] as shown in Figure 1.1 is a two-seat, high-wing, single-engine airplane that was designed and built by Cessna Aircraft Company from 1958 to 1977. The design process of the Cessna 150 involved many stages, from initial concept development to detailed design and testing, before it was put into production.

The Cessna 150 was developed to be a simple, reliable, and affordable airplane that could be used for a variety of purposes, such as flight training, personal transportation, and recreational flying. The design team at Cessna focused on creating an airplane that was easy to fly and maintain, while also meeting the requirements of the Federal Aviation Administration (FAA) for certification in the Normal category of the Civil Air Regulations (CAR).

The design process began with the selection of the Continental O-200 engine, which provided 100 horsepower and was well-suited for the airplane's intended use. The engine was mounted in a truss-type steel frame that formed the basis of the airplane's structure. The high-wing design allowed for excellent visibility and stability, while the wing struts provided additional support and reduced drag.

The fuselage of the Cessna 150 was designed to be spacious and comfortable for two people, with ample headroom and legroom. The cockpit was equipped with dual controls and a full complement of instruments for flight navigation and monitoring. The cabin was also designed with safety in mind, with features such as reinforced seats and a fuel system that was isolated from the cabin to reduce the risk of fire in the event of an accident.

The Cessna 150 was designed to be easy to maintain and repair, with a modular construction that allowed for easy access to the engine and other components. The airplane was also designed to be lightweight and fuel-efficient, with a maximum takeoff weight of 680kg and a cruise speed of 67km/h.

Figure 1.1 Overview of Cessna 150

Once the initial design was completed, the Cessna 150 underwent a series of rigorous tests to ensure that it met all of the FAA's requirements for certification.

These tests included ground testing, flight testing, and structural testing, as well as tests for stability and control, performance, and safety.

After the Cessna 150 was certified by the FAA, it went into production and became one of the most successful and popular light airplanes of all time, with over 23,000 units produced. The airplane's success was due in large part to its excellent design, which combined simplicity, reliability, and affordability with outstanding performance and versatility.

Overall, the design process of the Cessna 150 was a remarkable achievement in aviation engineering, demonstrating the ability of Cessna Aircraft Company to create a high-quality airplane that was both innovative and practical.

Figure 1.2 shows the three-view of Cessna 150 [5].

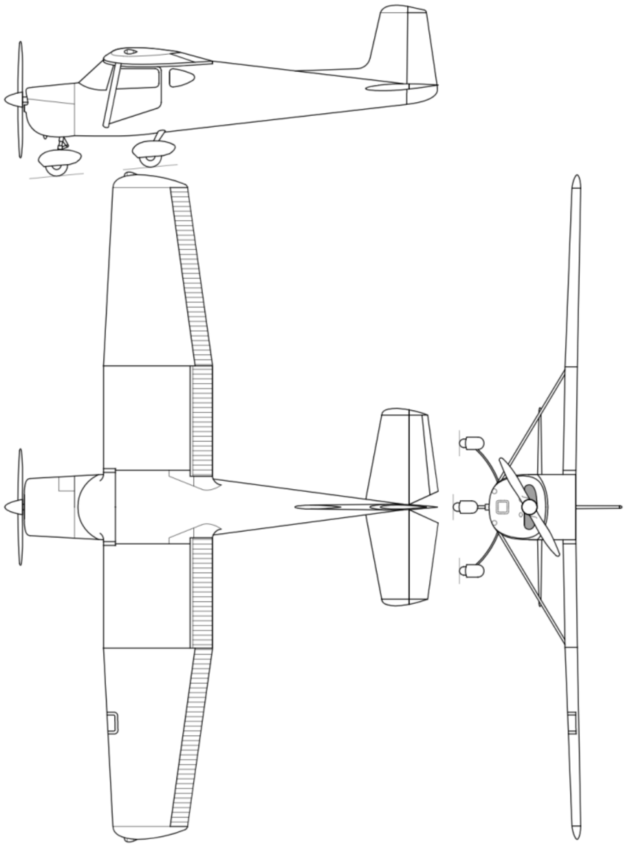

Figure 1.2 Three-view of Cessna 150

From Figure 1.2, We can see the sketch of the wing, following is the wing data analysis of Cessna 150 [6]:

The wing structure of Cessna 150 uses a variety of materials:

a. The wing spar and truss use high-strength aluminum alloy, which has the characteristics of lightweight, high strength, and corrosion resistance, and can improve the strength and rigidity of the wing.

b. The surface covering of the wing uses lightweight aviation-grade aluminum alloy sheet, which has high corrosion resistance and surface smoothness, and can reduce the surface resistance and wind resistance of the wing.

c. The leading edge and trailing edge of the wing are covered with fiberglass film, which has good wear resistance and corrosion resistance, and can effectively protect the leading edge and trailing edge of the wing, prolonging the service life of the wing.

There are some characters in the wing design:

a. High-wing design: The wings are attached to the top of the fuselage, which can increase lift and make the aircraft easier to take off and land. It also provides better visibility and a more comfortable flying experience.

b. Wing membrane design: The front and rear edges of the wings are smooth surfaces, reducing air resistance and improving the aircraft's speed and fuel efficiency.

c. Double-beam truss structure: The internal structure of the wing consists of two intersecting beams and a series of truss frames that form the skeleton of the wing. The skin panels cover the skeleton to form the shape of the wing. This structure can improve the strength and stiffness of the wing while reducing its weight, increasing the aircraft's payload and fuel efficiency.

d. Compact design: The wings of the Cessna 150 are relatively short, which can reduce drag and wind resistance, improving the aircraft's speed and fuel efficiency. Additionally, the compact design of the wings also makes the aircraft more agile and maneuverable.

Cessna 172 Skyhawk

The Cessna 172 Skyhawk [7] as shown in Figure 1.3 is a four-seat, single-engine, high-wing airplane that was designed and built by Cessna Aircraft Company. The design process of the Cessna 172 involved many stages, from initial concept development to detailed design and testing, before it was put into production.

Figure 1.3 Overview of Cessna 172 Skyhawk

The Cessna 172 was developed to be a reliable and easy-to-fly airplane that could be used for a variety of purposes, such as flight training, personal transportation, and recreational flying. The design team at Cessna focused on creating an airplane that was versatile, efficient, and affordable while meeting the requirements of the Federal Aviation Administration (FAA) for certification in the Normal category of the Civil Air Regulations (CAR).

The design process began with the selection of the Continental O-300 engine, which provided 145 horsepower and was well-suited for the airplane's intended use. The engine was mounted in a truss-type steel frame that formed the basis of the airplane's structure. The high-wing design allowed for excellent visibility and stability, while the wing struts provided additional support and reduced drag.

The fuselage of the Cessna 172 was designed to be spacious and comfortable for four people, with ample headroom and legroom. The cockpit was equipped with dual controls and a full complement of instruments for flight navigation and monitoring. The cabin was also designed with safety in mind, with features such as reinforced seats and a fuel system that was isolated from the cabin to reduce the risk of fire in the event of an accident.

The Cessna 172 was designed to be easy to maintain and repair, with a modular construction that allowed for easy access to the engine and other components. The airplane was also designed to be fuel-efficient, with a maximum takeoff weight of 2,550 pounds and a cruise speed of 122 knots.

Once the initial design was completed, the Cessna 172 underwent a series of rigorous tests to ensure that it met all of the FAA's requirements for certification. These tests included ground testing, flight testing, and structural testing, as well as tests for stability and control, performance, and safety.

After the Cessna 172 was certified by the FAA, it went into production and became one of the most successful and popular general aviation airplanes of all time, with over 44,000 units produced. The airplane's success was due in large part to its excellent design, which combined reliability, efficiency, and affordability with outstanding performance and versatility.

In the years since its introduction, the Cessna 172 has undergone many improvements and modifications, including engine upgrades, avionics upgrades, and structural improvements. The airplane has also been used for a variety of specialized purposes, such as aerial photography, air ambulance, and military training.

Overall, the design process of the Cessna 172 Skyhawk was a remarkable achievement in aviation engineering, demonstrating the ability of Cessna Aircraft Company to create a high-quality airplane that was both innovative and practical.

Figure 1.4 shows the three-view of Cessna 172 Skyhawk [8].

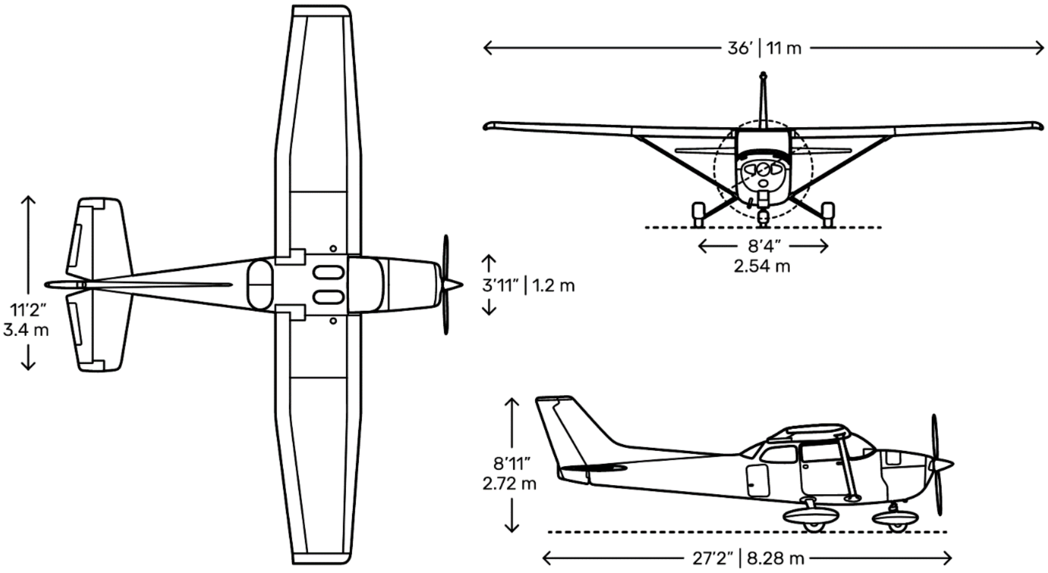

Figure 1.4 Three-view of Cessna 172 Skyhawk

From Figure 1.4, We can see the sketch of the wing, following is the wing data analysis of Cessna 172 Skyhawk [9]:

There are some similar materials to Cessna 150 used in the Wing:

a. Main spar and ribs: high-strength aluminum alloy. This material has the characteristics of lightweight, high strength, and corrosion resistance, which can improve the strength and stiffness of the wing.

b. Surface covering skin: aviation-grade aluminum alloy plate. This material has high corrosion resistance and surface smoothness, which can reduce the surface resistance and wind resistance of the wing.

c. Leading edge and trailing edge: fiberglass film. This material has good wear resistance and corrosion resistance, which can effectively protect the leading edge and trailing edge of the wing and prolong the service life of the wing.

There are also some similar characters to Cessna 150 in the wing design:

a. High-wing design: The wing is attached to the top of the fuselage, which enhances the lift and stability of the wing, making the aircraft easier to take off and land. This design also provides better visibility and a more comfortable flying experience.

b. Wing covering: The wing's leading and trailing edges are smooth curved surfaces that reduce drag and wind resistance, thereby increasing the aircraft's speed and fuel efficiency.

c. Double-beam truss structure: The wing's interior consists of two intersecting beams and a series of trusses that form the skeleton of the wing, which is then covered with skin panels to create the wing's outer shape. This structure improves the wing's strength and rigidity, while also reducing its weight, increasing the aircraft's payload capacity and fuel efficiency.

Cessna 182 Sky Lane

The Cessna 182 Sky Lane [10] as shown in Figure 1.5 is a four-seat, single-engine, high-wing aircraft designed and produced by the Cessna Aircraft Company. The design process involved multiple stages, from initial concept development to detailed design and testing, until production.

The design goal of the Cessna 182 Sky Lane was to create a high-performance aircraft suitable for both business and personal use in all weather conditions. The design team aimed to develop an efficient and comfortable aircraft that also met the normal category certification requirements set by the Federal Aviation Administration (FAA) under the Civil Air Regulations (CAR).

During the design process, the team chose the Continental O-470-R engine, which provides the aircraft with 230 horsepower. The engine is connected to the bottom of the aircraft, and the entire bottom structure is designed as a box beam to provide support. The high-wing design provides excellent visibility and stability, while the wing struts provide additional support and reduce drag.

The Cessna 182 Sky Lane's cabin is designed to be spacious and comfortable, accommodating four people with ample head and legroom. The cockpit is equipped with dual controls and a full set of instruments for flight navigation and monitoring. The cabin is also designed to be very safe, including reinforced seats and an independent fuel system to reduce the risk of fire in the event of an accident.

The design of the Cessna 182 Sky Lane is intended to be easy to maintain and repair, with a modular structure that allows easy access to the engine and other components. The aircraft is also designed to be fuel-efficient, with a maximum takeoff weight of 3100 pounds and a cruising speed of 140 knots.

Figure 1.5 Overview of Cessna 182 Sky Lane

After receiving FAA certification, the Cessna 182 Sky Lane began production and became one of the most successful and popular general aviation aircraft in history, with over 23,000 produced. The aircraft's success is attributed to its outstanding design, which combines reliability, efficiency, and economy with excellent performance and versatility.

Figure 1.6 shows the three-view of Cessna 182 Sky Lane [11].

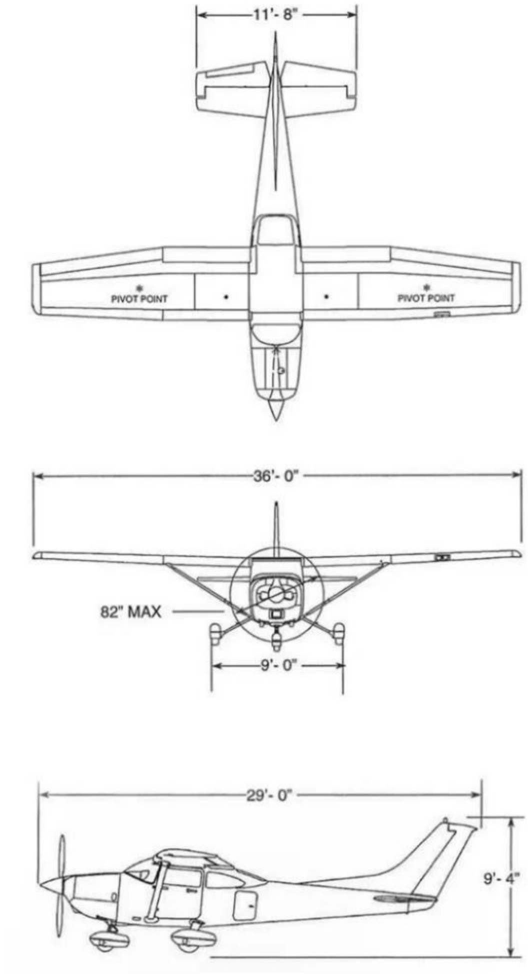

Figure 1.6 Three-view of Cessna 182 Sky Lane

From Figure 1.6, We can see the sketch of the wing, following is the wing data analysis of Cessna 182 Sky Lane [12]:

Cessna 182 Sky Lane's wings use a variety of different materials, including:

a. Main spars and ribs: high-strength aluminum alloy. This material has the characteristics of light weight, high strength, and corrosion resistance, which can improve the strength and stiffness of the wings.

b. Surface covering skin: aviation-grade aluminum alloy plate. This material has high corrosion resistance and surface smoothness, which can reduce the surface resistance and drag of the wings.

c. Leading and trailing edges: fiberglass film. This material has good wear resistance and corrosion resistance, which can effectively protect the leading and trailing edges of the wings and prolong the service life of the wings.

d. In addition, the wings of the Cessna 182 Sky Lane also use special materials commonly used in internal combustion engine aircraft, such as titanium alloy, composite materials, etc. These materials have the advantages of higher strength and light weight, which can further improve the performance and flight performance of the wings.

The design of Cessna 182 Sky Lane wing features:

a. High-wing configuration: Similar to the Cessna 172, the wing is mounted on the top of the fuselage, which increases lift and stability, making the aircraft easier to take off and land, and provides better visibility and a comfortable flying experience.

b. Wing membrane design: The leading and trailing edges of the wing are smooth surfaces, reducing drag and wind resistance, and improving the aircraft's speed and fuel efficiency.

c. Double-beam truss structure: Similar to the Cessna 172, the wing is constructed of two intersecting beams and a series of trusses, which form the skeleton of the wing. The skin panels are then overlaid on the skeleton to form the wing's shape. This structure increases the wing's strength and stiffness while reducing its weight, improving the aircraft's payload capacity and fuel efficiency.

d. Improved fuel system: The Cessna 182 Sky Lane introduced a dual fuel pump and fuel injection system, which improves fuel efficiency and reliability.

e. Adjustable flaps: The Cessna 182 Sky Lane features adjustable flaps, which can be adjusted to change the wing's lift and drag according to the flight requirements, improving the aircraft's performance and stability.

f. Large wing fuel tank: The Cessna 182 Sky Lane has a large fuel tank inside the wing, which can hold more fuel and extend flight time and distance.

Cessna 177 RG

The design process of the Cessna 177 RG [13] as shown in Figure 1.7 began in the late 1960s with the goal of creating a new model that would be more efficient and comfortable than the Cessna 172, while still maintaining its excellent flight characteristics. The design team at Cessna focused on creating a high-winged, single-engine aircraft that would offer better visibility and more room in the cabin for passengers and cargo.

To achieve these goals, the designers of the Cessna 177RG incorporated a number of new features into the aircraft's design. These included a wider fuselage, a larger cabin, and a more aerodynamic wing design. Additionally, the aircraft was equipped with a retractable landing gear, which helped to reduce drag and improve performance.

The Cessna 177RG was also equipped with a new engine, the Pratt & Whitney Canada IO-360, which provided increased power and efficiency compared to previous Cessna models. This, combined with the aircraft's other design features, resulted in a faster and more efficient aircraft that was well-suited for a variety of applications.

Figure 1.7 Overview of Cessna 177RG

After several years of design and testing, the Cessna 177RG was introduced to the market in 1971. It quickly became popular among pilots and aviation enthusiasts for its combination of speed, efficiency, and comfort. Although production of the Cessna 177RG ended in 1978, it remains a beloved aircraft among many pilots today.

Figure 1.8 shows the three-view of Cessna 177RG [14].

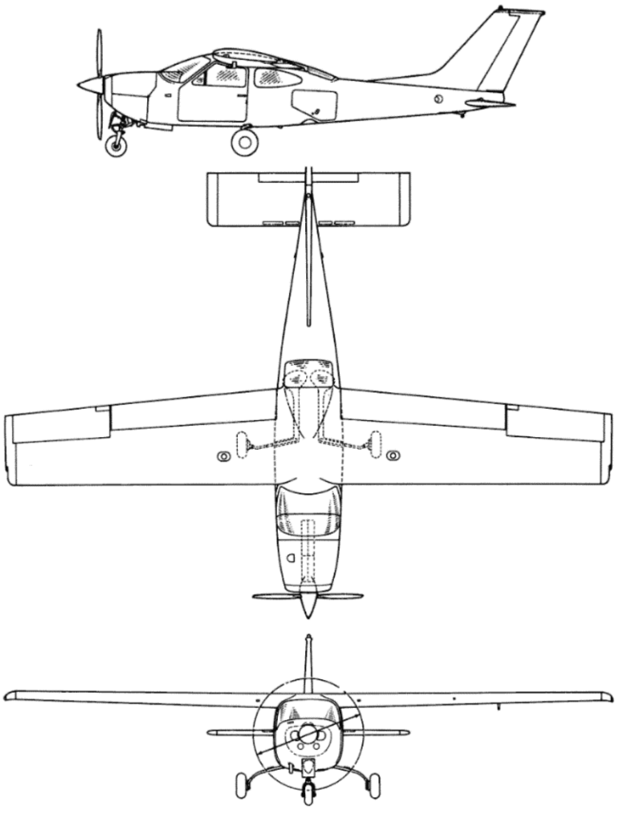

Figure 1.8 Three-view of Cessna 177RG

From Figure 1.8, We can see the sketch of the wing, following is the wing data analysis of Cessna 177RG [15]:

There are many similar materials to Cessna 182 Sky Lane used in the Wing, as described before.

Cessna 177RG is a high-wing aircraft with the following features in its wing design:

a. High-wing design: The wing of the Cessna 177RG is connected to the top of the fuselage, which improves the lift and stability of the wing, making the aircraft easier to take off and land, and providing better visibility and a comfortable flight experience.

b. Asymmetric airfoil: The wing of the Cessna 177RG uses an asymmetric airfoil, also known as the "Stefan wing," with a longer upper surface than the lower surface, which generates a larger lift coefficient and improves the lift performance of the aircraft.

c. Single main spar: The wing of the Cessna 177RG uses a single main spar, which reduces manufacturing costs while also reducing the weight of the wing, increasing the aircraft's payload and fuel efficiency.

d. Elastic suspension: The wing of the Cessna 177RG uses an elastic suspension system, which reduces the effect of wind on the wing, improving the stability and smoothness of the aircraft.

e. Adjustable flaps: The Cessna 177RG has adjustable flaps, which can adjust the angle of the flaps according to flight needs, thereby changing the lift and drag of the wing, improving the performance and stability of the aircraft.

f. Built-in fuel tanks: The wing of the Cessna 177RG has built-in fuel tanks, which can hold more fuel and extend the flight time and distance.

In summary, the high-wing design of the Cessna 177RG aims to improve the lift performance and stability of the aircraft while reducing the weight of the wing, increasing the aircraft's payload and fuel efficiency.

To sum up

These airplanes are all produced by Cessna Aircraft Company, and below are their similarities and differences:

Similarities:

- All of these airplanes are single-engine, small, and light aircraft suitable for personal and commercial use.
- They have similar wing shapes, with straight leading and trailing edges forming rectangular wings.
- They all use a vertical tail fin and horizontal stabilizer, giving them stable

flight characteristics.

- All of these airplanes are propeller-driven aircraft, using adjustable variable-pitch propellers.

Differences:

- The Cessna 150 is the smallest airplane, equipped with a 100-horsepower engine and can accommodate two passengers and a small amount of luggage. The Cessna 172 and Cessna 182 Sky Lane are larger, with the Cessna 172 equipped with a 180-horsepower engine and the Cessna 182 Sky Lane equipped with a 230-horsepower engine, capable of accommodating 4-6 passengers and more luggage.
- The Cessna 177 RG has a different wing shape, with a larger sweep angle and forward-swept leading edge. It also features retractable landing gear, giving it higher speed and better performance.
- The Cessna 182 Sky Lane and Cessna 177 RG can both use retractable landing gear.

As this project needs to study the internal wing structure design, the similarities and differences of the wing structure of these light aircraft are summarized as follows:

Similarities:

- All of these aircraft have a rectangular wing shape with a straight leading and trailing edge.
- Their wings use a metal frame structure and skin structure.
- They are all equipped with flaps and ailerons on the wings, which can increase lift and maneuverability at low speeds.

Differences:

- The wing area of the Cessna 150 is relatively small with a shorter wingspan.
- The wings of the Cessna 172 and Cessna 182 Sky Lane are larger in wing area and have a longer wingspan to accommodate more passengers and luggage.
- The Cessna 177 RG has a different wing shape with a larger sweep angle and forward-swept leading edge to improve flight performance.
- The wings of the Cessna 182 Sky Lane and Cessna 177 RG have wingtip fences to reduce the aircraft's drag and noise in the air.

1.2.3 Composite material data analysis

Composite materials are materials that are made by combining two or more different materials with the aim of achieving desirable properties not available in any of the constituent materials alone. The resulting composite material has superior physical, mechanical, and chemical properties than the individual materials [16].

Composite materials can be mainly divided into two categories: structural composite materials and functional composite materials.

Structural composite materials are materials used as load-bearing structures. They are typically composed of reinforcement elements that can withstand loads and matrix elements that connect the reinforcement elements to form an integral material while also transferring forces. The reinforcement elements can include various types of glass, ceramics, carbon fibers, polymers, metals, natural fibers, fabrics, whiskers, sheets, and particles, while the matrix elements can include polymers (resins), metals, ceramics, glass, carbon, and cement. Different combinations of reinforcement and matrix elements can be used to create a wide range of structural composite materials, which are typically named based on the type of matrix used, such as polymer matrix composites. The characteristics of structural composite materials are that they can be designed based on the requirements of the material's load-bearing function, and more importantly, they can be designed as composite structures, with reinforcement element layouts that can meet the necessary requirements while saving materials.

Functional composite materials are typically composed of functional and matrix elements. The matrix not only serves to form the integral material, but can also produce synergistic or enhanced functional effects. Functional composite materials refer to composite materials that provide physical properties beyond mechanical performance, such as electrical conductivity, superconductivity, semiconductor behavior, magnetism, piezoelectricity, damping, absorption, transmission, friction, shielding, flame retardancy, heat resistance, sound absorption, and thermal insulation. They are mainly composed of functional and reinforcement elements and matrix elements. The functional elements can be composed of one or more functional materials, and composite materials with multiple functions can be created by combining different functional elements. In addition, new functions can also be created as a result of the composite effect. The development of multi-functional composite materials is the future direction of functional composite materials.

The following composite materials are used in the manufacturing process of Cessna 182 Sky Lane and 177 RG [17]:

a. Carbon fiber reinforced composites: This material is made up of carbon fiber and resin and is used to manufacture components such as fuselage, wings, and tail.

b. Glass fiber reinforced composites: This material is made up of glass fiber and resin and is used to manufacture components such as fuselage, wings, and tail.

c. High-temperature resistant composites: This material is mainly used to manufacture high-temperature parts such as engine compartments, exhaust pipes, and turbochargers. It is made up of carbon fiber, ceramics, and resin.

d. Carbon fiber/epoxy resin prepreg: This material is mainly used to manufacture large components such as fuselage and wings. It is made up of carbon fiber and epoxy resin and has advantages such as high strength, light weight, and corrosion resistance.

The composition of these composite materials may vary slightly depending on factors such as production batches, manufacturing processes, and specification requirements.

There are some common composite materials used in the aircraft:

- Carbon Fiber Reinforced Polymer (CFRP): Made up of carbon fibers and resin, used for manufacturing aircraft body, wings, and tail components.
- Glass Fiber Reinforced Polymer (GFRP): Made up of glass fibers and resin, used for manufacturing aircraft body, wings, and tail components.
- High Temperature Resistant Composite Material: Made up of carbon fibers, ceramics, and resin, used for manufacturing high-temperature components like engine compartments, exhaust pipes, and turbochargers.
- Carbon Fiber/Epoxy Resin Prepreg: Made up of carbon fibers and epoxy resin, with advantages of high strength, lightweight, and corrosion resistance, used for manufacturing large aircraft parts like body and wings.
- Polyimide (PI) [18]: Known for high strength, heat resistance, and corrosion resistance, used for manufacturing aircraft engine compartments and wings.
- Polyamide (PA) [19]: Known for high strength and rigidity, used for manufacturing aircraft engine compartments and wings.
- Phenolic Resin (PF) [20]: Known for excellent heat resistance and flame-retardant properties, used for manufacturing interior, partitions, and support structures.
- Polystyrene Foam (PS) [21]: Known for lightweight and sound insulation properties, used for manufacturing partitions, insulation materials, and decorative materials.
- Polycarbonate (PC) [22]: Known for high strength and transparency, used for manufacturing aircraft cockpit glass and portholes.
- Aluminum Alloy Composite Material (AMC): Made up of aluminum alloy and fiber-reinforced materials, known for high strength and corrosion resistance, used for manufacturing aircraft body, wings, and tail components.

1.2.4 Composite honeycomb sandwich structure in aircraft

The sandwich structure emerged in the early 1940s, a process of just over 60 years. It was first used by Professor H. J. G. Hirsch in the UK when designing aircraft structures, and later adopted by the Chance Vought aircraft manufacturing company in the US for producing wings [23]. The sandwich structure typically consists of thin sheets as facings and thicker, low-density materials as the core, which are bonded together. Examples of materials used include metallic honeycomb cores, aramid fiber (such as Kevlar), glass fiber, and carbon fiber reinforced thermoset resin honeycomb cores [24-25]. Figure 1.9 shows the composition of a honeycomb sandwich structure, and Figure 1.10 shows a full-height honeycomb sandwich structure beam. The sandwich structure greatly improves the stress conditions of structural components and has many outstanding properties, such as thermal insulation, high bending stiffness-to-mass ratio, and is therefore widely used in aircraft structures.

The domestically-produced Z-9 helicopter is composed of over 280 sandwich structure components made of Nomex honeycomb. The amount of Nomex honeycomb used in a single helicopter reaches 260 square meters, accounting for around 80% of the total coverage area, making it one of the largest users of honeycomb sandwich structures in China. A more novel honeycomb sandwich material was used for the floor of the Boeing 747, with carbon fiber reinforced plastic as the facing and a honeycomb core made of nylon. Its lifespan can reach 20,000 hours, and the weight saved by the aircraft is enough to accommodate seven additional passengers. The maximum weight reduction achieved by using sandwich structure composite materials for aircraft structural components is 35%.

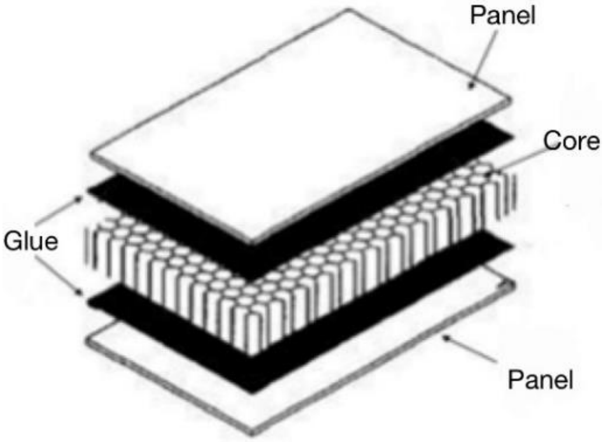

Figure 1.9 Honeycomb sandwich structure composition [26]

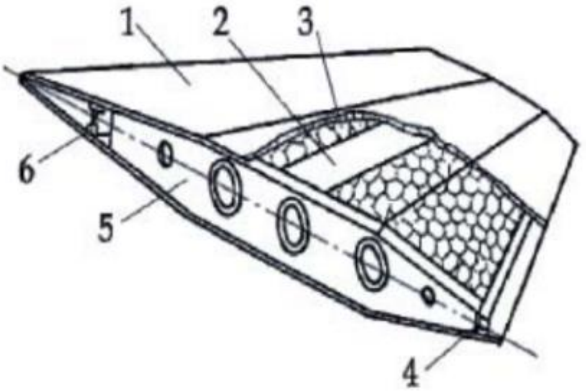

Figure 1.10 Honeycomb sandwich structure wing [27]

1-panel, 2-spar, 3-hoeycomb core, 4-behind wall, 5-rib, 6-front wall

1.3 Determining the parameters of two-seater light aircraft

In this chapter, the main focus is on the determination of the various initial parameters of the aircraft and the calculation of the relevant parameters.

1.3.1 Basic technical requirements

After collecting statistics, tactical and technical requirements can be determined.

Designed airplane with a capacity of up to 2 pilots/passengers and a range of L = 1800 km, the runway length L_{TO} = 0.5 km is assigned, then the cruising altitude H_{CR} = 2.40 km, and cruising speed V_{CR} = 260 km/h. There is recorder in table 1.3.

Table 1.3 – Tactical-Technical data of designed airplane

$M_{H = 2.4\ km}$	M_{max}	$L_{(mp=max)}$, km	$m_{p.}$, kg	$L_{TO.}$, km	H_{max}, km	V_{CR}, km/h	n_{crew}	n_{pass}	H_{CR}, km
0.21	*0.23*	*1800*	*160*	*0.5*	*3.5*	*260*	*1*	*1*	*2.40*

According to the analysis and processing of statistical data, Table 1.4 lists the basic geometric parameters for aircraft.

Table 1.4 – Main geometric parameters of designed airplane

λ	$\chi_{1/4}$	η	$\bar{c}$	$\bar{b}_{fl}$	δ_{fl}, degree	$\bar{S}_{al}$	λ_F	D_F, m	L_F, m
7.87	*3.21*	*1.75*	*0.15*	*0.25*	*25*	*0.07*	*5.67*	*1.46*	*8.29*
$\bar{S}_{HT}$	$\bar{S}_{VT}$	λ_{HT}	λ_{VT}	$\chi_{1/4\ HT}$	$\chi_{1/4\ VT}$	$\bar{c}_{HT}$	$\bar{c}_{VT}$	η_{HT}	η_{VT}
0.22	*0.20*	*3*	*1.06*	*15*	*40*	*0.1*	*0.1*	*2.0*	*2.0*

1.3.2 Take-off weight estimation

The take-off weight of the aircraft in the zero approximation is determined by the formula (1.1):

$$m_o = \frac{m_p + m_{OI.CE}}{1 - \left(\bar{m}_S + \bar{m}_{PP} + \bar{m}_{EQ} + \bar{m}_F\right)}, \qquad (1.1)$$

where m_p – mass of payload, which calculated:

$$m_p = m_{lug} \cdot n_{pas} = 120 \cdot 1 = 120 \text{ kg},$$

m_{lug} – mass of luggage, which calculated:

$$\mathrm{m_{lug}} = \mathrm{m_{pas}} + \mathrm{m_{bag}} = 90 + 30 = 120 \text{ kg};$$

$\mathrm{m_{OI.CE}}$ – mass of operational items and equipment:

$$\mathrm{m_{OI.CE}} = (\mathrm{m_{crew}} \cdot \mathrm{n_{crew}}) + \Delta \mathrm{m_{OI}} = (80 \cdot 1) + 50 = 130 \text{ kg},$$

here $\mathrm{m_{crew}}$ – the mass of service load and crew, it is assumed that the average weight of each crew member is 80 kg; $\Delta m_{OI} = 50\, kg$ – which includes food, water, training related equipment, medication, and other supplies.

$\overline{\mathrm{m}}_{\mathrm{S}}$ – Is the relative mass of the aircraft structure, which includes the relative mass of the wing, fuselage, tail, and landing gear. For light aircraft $\mathrm{m_S} = 0.34$.

$\overline{\mathrm{m}}_{\mathrm{PP}}$ – 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 $\overline{\mathrm{m}}_{\mathrm{PP}} = 0.10$.

$\overline{\mathrm{m}}_{F}$ – The relative mass of fuel, which is found by empirical formula:

$$\overline{\mathrm{m}}_{F} = a + \mathrm{b}\,\frac{L}{V_{CR}} = \mathrm{a} + \mathrm{b} \cdot \mathrm{t}_{p} = 0.05 + (0.04 \cdot 6.923) = 0.327,$$

where L – is the flight range L= 1800 km, V – is the speed, V= 260 km/h, tp – the estimated flight time, tp = 6.923 hours, a = 0.05, b = 0.04.

$\overline{\mathrm{m}}_{\mathrm{EQ}}$ – 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 subsonic heavy aircraft $\overline{\mathrm{m}}_{\mathrm{EQ}} = 0.10$.

The take-off mass of the aircraft in the zero approximation is calculated by formula (1.1):

$$m_o = \frac{120 + 130}{1 - (0.34 + 0.10 + 0.327 + 0.1)} = 1879.7 \approx 1880 \text{ kg}.$$

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: $m_S = 0.28 \cdot m_o = 0.28 \cdot 1880 \approx 526$ kg.

Wing weight: $m_W = 0.377 \cdot m_S = 0.377 \cdot 526 \approx 198$ kg.

Fuselage mass: $m_F = 0.367 \cdot m_S = 0.367 \cdot 526 \approx 193$ kg.

Tail weight: $m_{TU} = 0.073 \cdot m_S = 0.073 \cdot\ 526 \approx 38.5$ kg.
Fuel mass: $m_{fuel} = 0.259 \cdot m_S = 0.259 \cdot\ 526 \approx 132$ kg.
Landing gear weight: $m_{LG} = 0.183 \cdot m_S = 0.183 \cdot 526 \approx 96.5$ kg.
Mass of the power plant: $m_{PP} = 0.10 \cdot m_0 = 0.10 \cdot 1880 = 188$ kg.
Mass of the equipment: $m_{EQ} = 0.10 \cdot m_0 = 0.10 \cdot 1880\ =\ 188$ kg.

In the table 1.5 are indicated the masses of the airplane parts and units.

Table 1.5 – Masses of the airplane parts and units

m_o, kg	$m_{OI.CE}$, kg	m_p, kg	m_{CR}, kg	m_{fuel}, kg	m_{PP}, kg	m_{EQ}, kg	m_S, kg			
							m_W, kg	m_F, kg	m_{TU}, kg	m_{LG}, kg
1880	130	120	80	132	188	188	198	193	38.5	96.5
							526			

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 takeoff is p_0= $80 daN/m^2$, wing area can be determined by formula (1.2):

$$S = \frac{m_0 \cdot\ g}{10 \cdot p_0} = \frac{1166 \cdot\ 9.8}{10 \cdot 80} \approx 14.28\, m^2. \tag{1.2}$$

Wingspan can be determined by formula (1.3):

$$l = \sqrt{\lambda S} =\ \sqrt{7.87 \cdot 14.28} \approx 10.6\, m. \tag{1.3}$$

The root b_0 and tip b_t chord of the wing can be determined by formula (1.4):

$$b_0 = \frac{S}{l} \cdot \frac{2 \cdot \eta}{\eta + 1} = \frac{14.28}{10.6} \cdot \frac{2 \cdot 1.75}{1.75 + 1} = 1.71\, m; \tag{1.4}$$

$$b_t = \frac{b_0}{\eta} = \frac{1.71}{1.75} = 0.98\, m.$$

The mean aerodynamic chord is calculated by formula (1.5):

$$b_{MAC} = \frac{2}{3} \cdot b_0 \cdot \frac{\eta^2 + \eta + 1}{(\eta + 1) \cdot \eta} = \frac{2}{3} \cdot 1.71 \cdot \frac{1.75^2 + 1.75 + 1}{(1.75 + 1) \cdot 1.75} = 1.37 m. \tag{1.5}$$

Determine the coordinate MAC on the wingspan by formula (1.6):

$$Z_A = \frac{l}{6} \cdot \frac{\eta + 2}{\eta + 1} = \frac{10.6}{6} \cdot \frac{1.75 + 2}{1.75 + 1} = 2.41\, m. \tag{1.6}$$

Coordinate of MAC nose along an OX axis is calculated by formula (1.7):

$$X_{MAC} = \frac{l}{6} \cdot \frac{\eta + 2}{\eta + 1} \cdot tg\chi_{LE},\ X_{MAC} = \frac{10.6}{6} \cdot \frac{1.75 + 2}{1.75 + 1} \cdot 0.091 = 0.22m, \tag{1.7}$$

where χ_{LE} – Sweep angle of wing leading edge is calculated by formula (1.8):

$$tgx_{LE} = tgx + \frac{\eta - 1}{\lambda(\eta + 1)} = tg3.21^{\circ} + \frac{1.75 - 1}{7.87(1.75 + 1)} = 0.091. \tag{1.8}$$

Determination of fuselage parameters

Fuselage diameter:

$$D_F = 1.46 \text{ m}.$$

Fuselage length is calculated by formula (1.9):

$$L_F = \lambda_F \cdot D_F = 5.67 \cdot 1.46 = 8.29\, m. \tag{1.9}$$

Determination of tail unit parameters

According to the textbook “DEVELOPMENT OF A PILOT PROJECT OF AN AIRCRAFT”, we can achieve the information:

HT: $\overline{S}_{HT} = S_{HT}/S_{cr} = 0.15\ldots0.3$; $\overline{S}_{EL} = S_{EL}/S_{HT} = 0.2\ldots0.4$;

$\lambda_{HT} = 2\ldots3.5$; $\eta_{HT} = 2\ldots3.5$; $\chi_{HT} = 0\ldots60^{\circ}$.

VT: $\overline{S}_{VT} = S_{VT}/S_{cr} = 0.08\ldots0.20$; $\overline{S}_{RUD} = S_{RUD}/S_{VT} = 0.2\ldots0.45$;

$\lambda_{VT} = 0.8\ldots1.2$; $\eta_{VT} = 2\ldots3.5$; $\chi_{VT} = 0\ldots60^{\circ}$.

Thus we select:

$\overline{S}_{HT} = 0.22$; $\overline{S}_{EL} = 0.3$; $\lambda_{HT} = 3$; $\eta_{HT} = 2$; $\chi_{HT} = 15^{\circ}$.

$\overline{S}_{VT} = 0.20$; $\overline{S}_{RUD} = 0.25$; $\lambda_{VT} = 1.06$; $\eta_{VT} = 2$; $\chi_{VT} = 40^{\circ}$.

The horizontal tail is determined according by formula (1.10):

$$S_{HT} = \bar{S}_{HT} \cdot S = 0.22 \cdot 14.28 = 3.14\, m^2. \tag{1.10}$$

Length of horizontal tail is determined by formula (1.11):

$$L_{HT} = \sqrt{\lambda_{HT} \cdot S_{HT}} = \sqrt{3 \cdot 3.14} = 3.07\, m. \tag{1.11}$$

Chords of horizontal tail unit is calculated by formula (1.12):

$$b_{0\,HT} = \left(\frac{S_{HT}}{L_{HT}}\right) \cdot \frac{2\eta_{HT}}{\eta_{HT}+1} = \left(\frac{3.14}{3.07}\right) \cdot \frac{2 \cdot 2}{2+1} = 1.36\,m. \qquad (1.12)$$

$$b_{t\,HT} = \frac{b_{OHT}}{\eta_{HT}} = \frac{1.36}{2} = 0.68\,m.$$

Mean aerodynamic chord of horizontal tail is calculated by formula (1.13):

$$b_{MAC\,HT} = \frac{2}{3} \cdot b_{0\,HT} \cdot \frac{{\eta_{HT}}^2 + \eta_{HT} + 1}{\eta_{HT} \cdot (\eta_{HT}+1)} = \frac{2}{3} \cdot 1.36 \cdot \frac{2^2+2+1}{2 \cdot (2+1)} = 1.06\,m. \quad (11.13)$$

$$Z_{MAC\,HT} = \frac{L_{HT}}{6} \cdot \frac{\eta_{HT}+2}{\eta_{HT}+1} = \frac{3.07}{6} \cdot \frac{2+2}{2+1} = 0.68m.$$

Horizontal distance between wing to MAC is calculated by formula (1.14):

$$X_{MAC\,HT} = \frac{l}{6} \cdot \frac{\eta+2}{\eta+1} \cdot tg\chi_{LE}, X_{MAC\,HT} = 2.36 \cdot 0.38 = 0.90\,m, \qquad (1.14)$$

where χ_{LE} – sweep angle of wing leading edge is calculated by formula (1.15):

$$tg\chi_{LE} = tg\chi + \frac{\eta_{HT}-1}{\lambda_{HT}(\eta_{HT}+1)} = tg15^\circ \; + \frac{2-1}{3 \cdot (2+1)} = 0.38 \qquad (1.15)$$

The vertical tail is determined by formula (1.16):

$$S_{VT} = \bar{S}_{VT} \cdot S = 0.20 \cdot 14.28 = 2.87\,m^2. \qquad (1.16)$$

Height of vertical tail is determined by formula (1.17):

$$h_{VT} = \sqrt{\lambda_{VT} \cdot S_{VT}} = \sqrt{1.06 \cdot 2.87} = 1.74\,m. \qquad (1.17)$$

Chords of vertical tail unit is determined by formula (1.18):

$$b_{0\,VT} = \left(\frac{S_{VT}}{h_{VT}}\right) \cdot \frac{2\eta_{VT}}{\eta_{VT}+1} = \left(\frac{2.87}{1.74}\right) \cdot \frac{2 \cdot 2}{2+1} = 2.21m. \qquad (1.18)$$

$$b_{t\,VT} = \frac{b_{0\,VT}}{\eta_{VT}} = \frac{2.21}{2} = 1.11m.$$

Mean aerodynamic chord of vertical tail unit is determined by formula (1.19):

$$b_{MAC\,VT} = \frac{2}{3} \cdot b_{0\,VT} \cdot \frac{{\eta_{VT}}^2 + \eta_{VT} + 1}{\eta_{VT} \cdot (\eta_{VT}+1)} = \frac{2}{3} \cdot 2.21 \cdot \frac{2^2+2+1}{2 \cdot (2+1)} = 1.72\,m. \qquad (1.19)$$

$$Y_{MAC\,VT} = \frac{h_{VT}}{6} \cdot \frac{\eta_{VT}+2}{\eta_{VT}+1} = \frac{1.73}{6} \cdot \frac{2+2}{2+1} = 0.38\,m.$$

Vertical distance between wing root to MAC root is determined by formula (1.20):

$$X_{AVT} = Y_{AVT} \cdot tg\chi_{VT} = 0.38 \cdot tg40^{\circ} \ = 0.32\,m. \tag{1.20}$$

Determine landing gear parameters

The size of offset e is more often determined in share of wing MAC is determined by formula (1.21):

$$e = (0.15 \ldots 0.20)b_{MAC} = 0.15 \cdot 1.37 = 0.21\,m. \tag{1.21}$$

Angle of main wheels setoff γ should be higher than angle of a touch down by a tail part is determined by formula (1.22):

$$\gamma = \varphi + (1 \ldots 2)^{\circ}, \tag{1.22}$$

φ is angle of touchdown by the tail part is determined by formula (1.23):

$$\varphi = \alpha_{max} - \alpha_{wi} - \Psi, \tag{1.23}$$

where $\alpha_w = (0 \ldots 4)^{\circ} = 0^{\circ}$ (angle between a wing chord and longitudinal axis of fuselage), $\Psi = 0$ (static ground (parking angle) in zero approximation it is $\Psi = 0$)

$$\varphi = 10 - 0 - 0 = 10^{\circ}.$$

$$\gamma = 10 + 2 = 12^{\circ}$$

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 take-off. But the lesser the *e* reduce γ.

Wheel base of the landing gear *b* it depends of fuselage length as formula (1.24):

$$b = (0.30 \ldots 0.40)L_F = 0.40 \cdot 8.29 = 3.32\,m. \tag{1.24}$$

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 as formula (1.25):

$$a = (0.88 \ldots 0.94)b = 0.90 \cdot 3.32 = 2.988\,m; \tag{1.25}$$

$$e = (0.12 \ldots 0.06)b = 0.07 \cdot 3.32 = 0.232\,m.$$

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.

On the basis of track of landing gear should be in such limits as formula (1.26):

$$2H < B < 15\,m \tag{1.26}$$

Calculation of H as formula (1.27):

$$H = \frac{e}{tan\gamma} = \frac{0.232}{tan12^{\circ}} = 1.09\,m. \tag{1.27}$$

$$2 \cdot H = 2 \cdot (1.09) = 2.18 < 2.3 < 15\,m.$$

Chosen size of track is satisfied condition, therefore B = 2.3m.

1.3.5 Development of the layout and center-of-gravity

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:

Distance from the center of mass to the nose of the wing MAC:

$$X_{C.G.} = (0.26 \dots 0.30) \cdot b_{MAC} = \ 0.26 \cdot b_{MAC} = 0.26 \cdot 1.37 = 0.36;$$

Distance form one-fourth of the MAC horizontal tail to the center of mass:

For the swept wing χ (30…. 60)° $L_{HS} = (2.5 \dots 3.5) \cdot b_{MAC} = 3.4 \cdot 1.37 = 4.66\,m.$

1.4 Modeling design of the whole aircraft

Aircraft overall design refers to the process of considering multiple design schemes and subsystems comprehensively to meet the comprehensive requirements of performance, safety, economy, and comfort. This project use Siemens NX to determine the model of designed aircraft.

Siemens NX is a 3D computer-aided design (CAD) and computer-aided manufacturing (CAM) software developed by Siemens Digital Industries Software. The software is aimed at providing complete solutions for engineers, designers, and manufacturers to design and manufacture complex products.

Siemens NX has extensive features, including modeling, assembly, drafting, engineering analysis, robot programming, toolpath generation, work holding simulation, and 3D printing. It also offers advanced features such as surface modeling, reverse engineering, simulation, and virtual reality.

In summary, Siemens NX is a powerful and comprehensive CAD/CAM software that can be used in various industries, including aerospace, automotive, mechanical, electronics, medical, etc. It provides complete design and manufacturing solutions for engineers, designers, and manufacturers.

The aerodynamic configuration of the aircraft is usually considered a system of relative positioning of lift surfaces (wing and stabilizers), their relative dimensions and shapes. The principal characteristic of the aircraft configuration is method of aircraft longitudinal trim. Longitudinal trimming of the aircraft is compensation of all force moments acting on the aircraft relatively to the aircraft longitudinal axis. Depending on stabilizers arrangements relatively to wings, four basic (balancing) configurations are distinguished [28]:

Figure 1.11 a

Figure 1.11 b

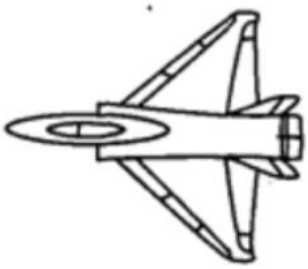

Figure 1.11 c

The “normal" configuration (Figure 1.11 a) – when the horizontal stabilizer is located behind the wing;

The “canard" configuration (Figure 1.11 b) – the horizontal stabilizer is located ahead of a wing;

The "tailless" configuration (Figure 1.11 c) – the aerodynamic configuration includes only one load bearing surface wing.

The “normal" classical configuration applies to most modern aircraft, and also in this aircraft:

Advantages of the "normal" configuration:

1. Wing is in the pure, undisturbed airflow and is not shadowed by stabilizers;
2. Nose section of a fuselage is short and does not create destabilizing moment relatively to the vertical axis; this allows to reduce area and mass of vertical stabilizer;
3. Crew has better observation of the front semi-sphere.

1.4.1 Unite of the aircraft computer modeling

Fuselage modeling

Use the generalized theoretical drawing of the fuselage and three view pictures of the prototype aircraft, thus draw the sketches of the aircraft as the following pictures shown.

During drawing the sketches, it is important to check all the elements are restricted, pay attention to all parameters that concerned. Determined the original point and distinct different point of view, step by step do fuselage modeling.

Since the fuselage is symmetrical, from the beginning is to create half model of the aircraft, such method will also be used in wing and horizontal stabilizer.

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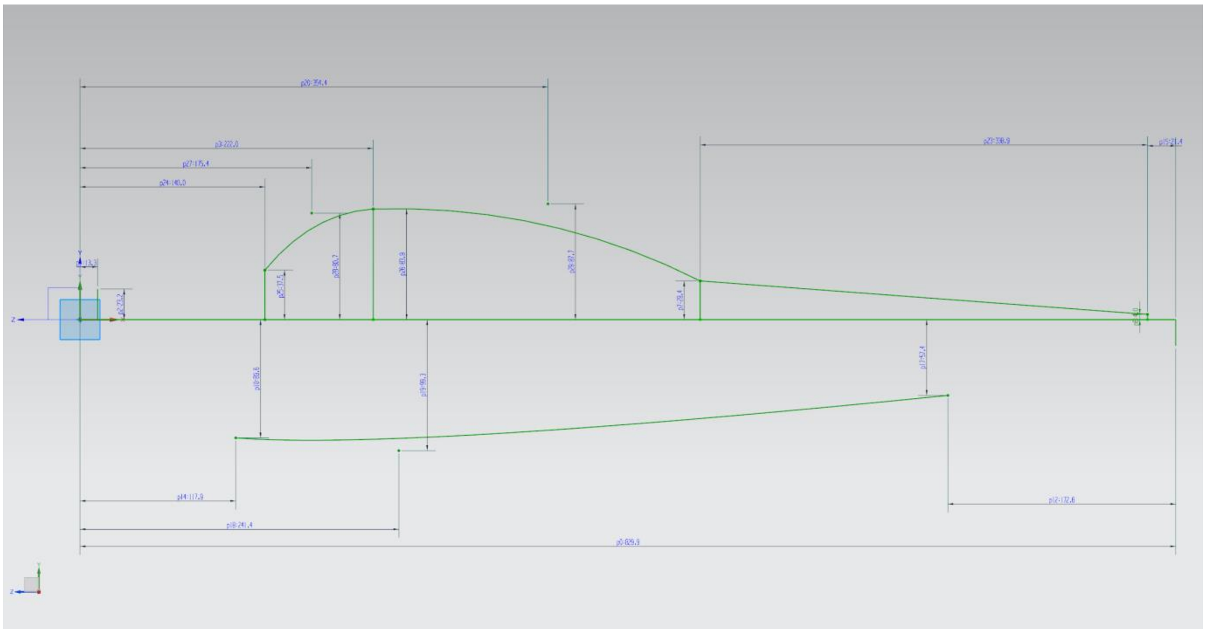


Figure 1.12 Sketch of the fuselage from left view

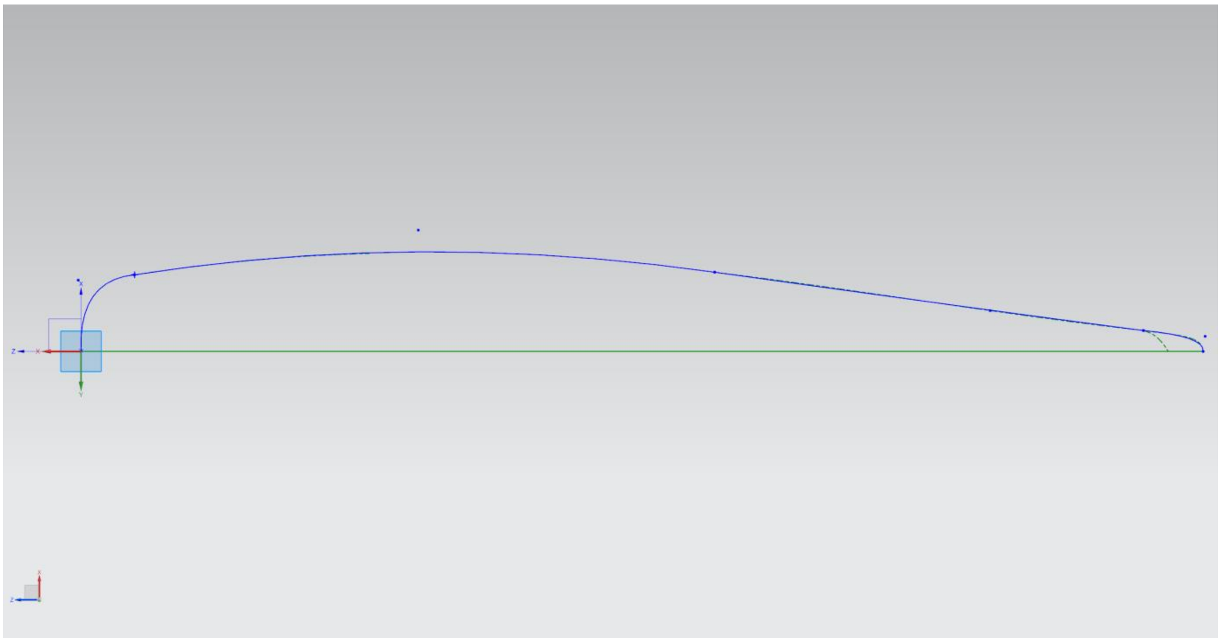


Figure 1.13 Sketch of the fuselage from bottom view

Figure 1.12 is the first step, shows the centerline sketch of the fuselage from left view, it’s in the XC – YC plane named «S»;

Figure 1.13 is the second step, shows a projection of the fuselage centerline onto a horizontal plane. It’s a sketch in the ZC – YC plane named «MW».

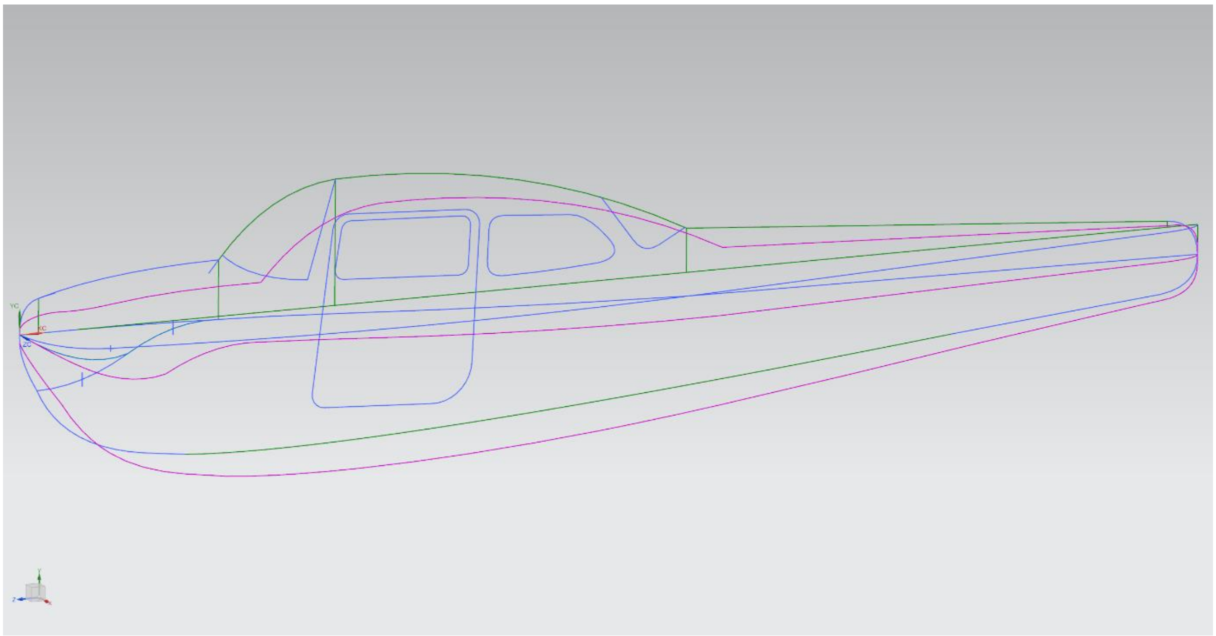


Figure 1.14 Sketch of the whole fuselage through projection

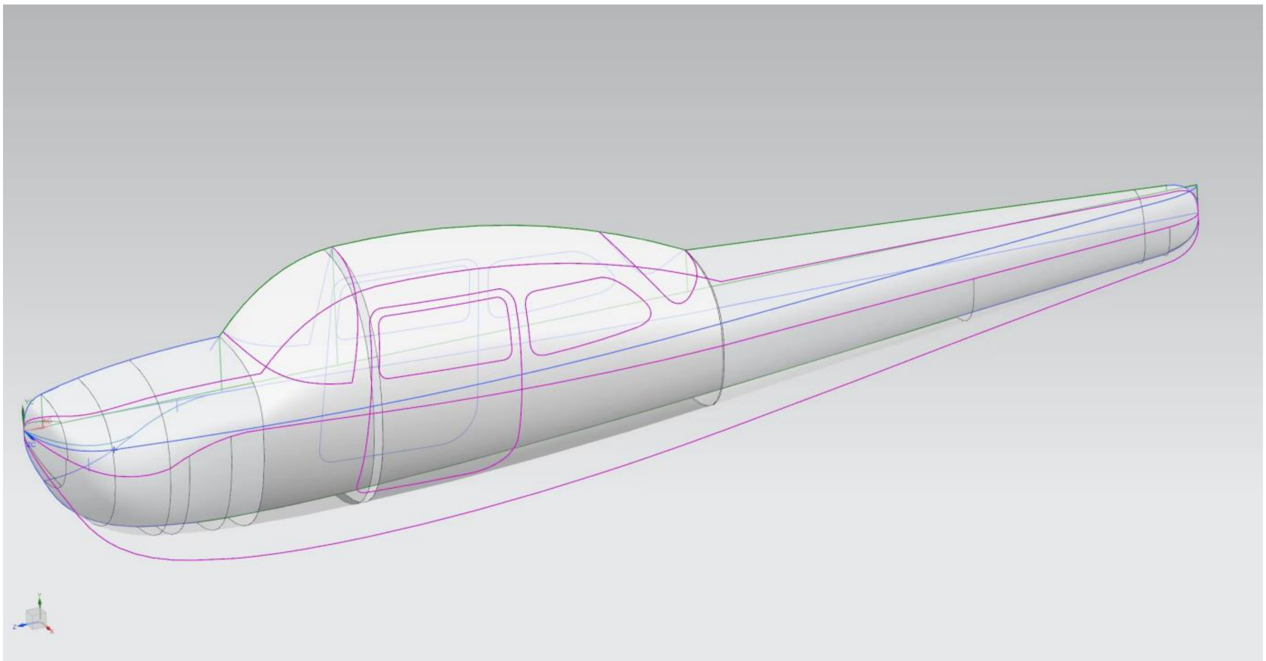


Figure 1.15 Create the surface of the fuselage and other parts (70% Transparency)

Figure 1.14 is the third step, it is including a sketch of the top and bottom guide curves of the surface of the nose of the fuselage located in the plane of symmetry of the fuselage XC – YC named «PN», and a projection of the center line of the nose of the fuselage on the plane of symmetry of the aircraft, which is a sketch in the XC – YC plane with the name «S_MN», and a sketch of the upper and lower guide curves of the surface of the tail part of the fuselage located in the plane of symmetry of the fuselage XC – YC with the name «P_H», and a projection of the centerline of the tail part of the fuselage on the plane of symmetry of the aircraft, which is a sketch in the XC – YC plane with the name «S_MH», and a sketch of the projection of the cockpit glazing on the plane of symmetry of the fuselage XC – YC with the name «P_WS».

Figure 1.15 is the fourth step, using various Surface generation commands such as Mesh Surface, Through Curve Mesh, Packed Surface, N-edge Surface and so on. At the same time, crate window and door by Parting Plane.

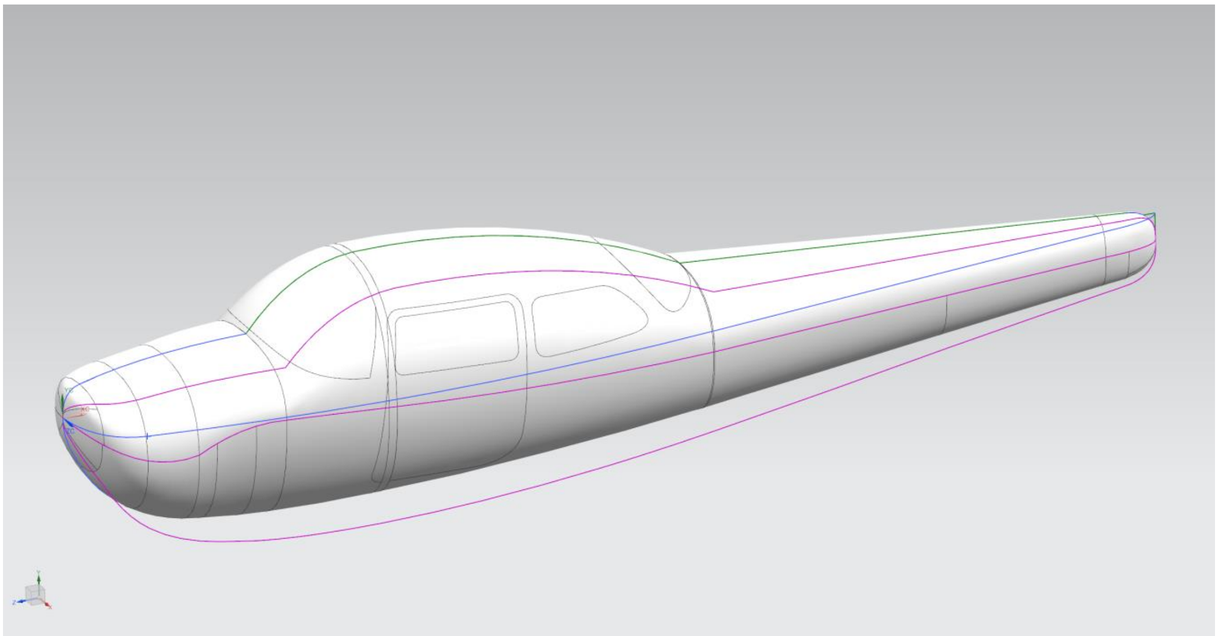
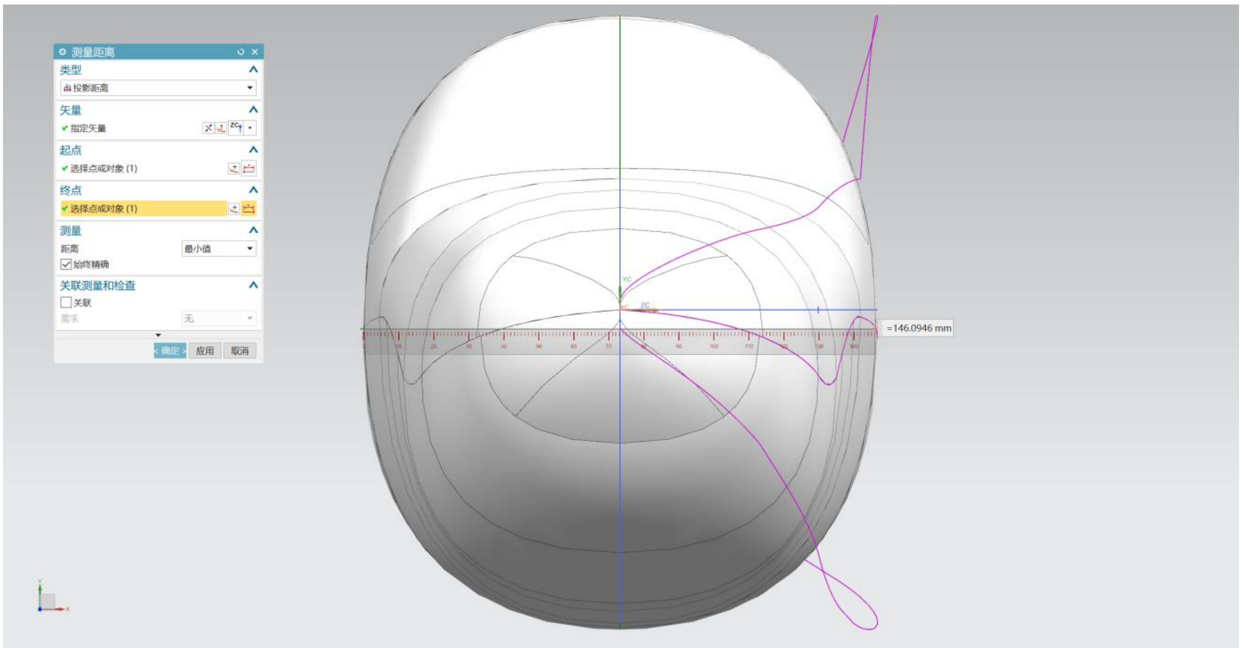


Figure 1.16. The mirror order of the fuselage modeling

Figure 1.16 is the last step of the fuselage modeling; we can gain the fuselage model after doing this step.

Figure 1.17 is to use the software to measure the width of the fuselage, which is correspond with the calculated result.



测量距离

类型

投影距离

矢量

✓ 指定矢量

起点

✓ 选择点或对象 (1)

终点

✓ 选择点或对象 (1)

测量

距离 最小值

☑ 始终精确

关联测量和检查

☐ 关联

需求 无

≈146.0946 mm

Figure 1.17 Measure the width of the fuselage

Wing modeling

In this aircraft we choose NACA four-digit airfoil, NACA2412 is fit for this aircraft. The concrete statistic is shown as the following:

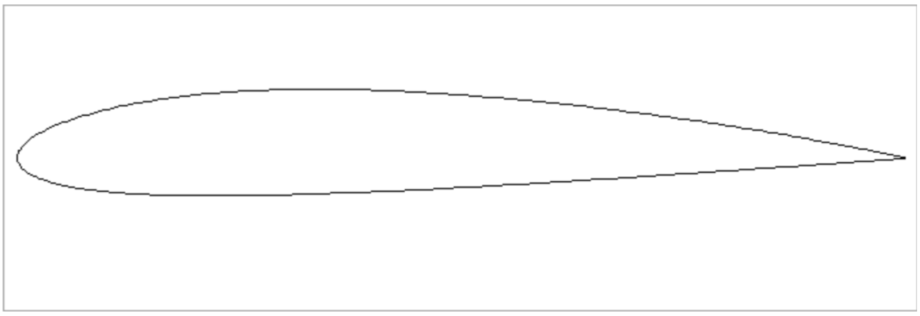


Figure 1.18 The shape of airfoil used in the wing

1. Maximum curvature (thickness) 12.01% at 29.9% wing chord;
2. The maximum surface is 1.92% at 41.7% wing chord;
3. Airfoil leading edge radius 1.5475%;
4. Airfoil trailing edge thickness 0.2514%.

Profili is a flight vehicle design and modeling software developed by Italian engineer and computer scientist Andrea Tarozzi. The software was launched in 1995 and has since been continuously improved and updated, becoming a widely used tool in the industry.

Profili offers a wide range of features, including flight vehicle modeling, wing profile design, aerodynamic analysis, 3D visualization, and more. It also allows users to export their design and analysis results in multiple formats for integration with other software.

The software's advantages include:

- Comprehensive functionality: Profili covers all aspects of flight vehicle design and modeling, catering to the needs of different users.
- User-friendly interface: The software has a clear and concise interface design, making it easy to use and allowing users to quickly get started with modeling and analysis.
- High precision: Profili employs advanced computational methods and algorithms to provide high-precision analysis and simulation results.
- High customizability: Users can customize various parameters and options of the software according to their needs, meeting different design and analysis requirements.

In summary, Profili is a comprehensive, high-precision, and highly customizable flight vehicle design and modeling software suitable for various users, such as flight vehicle designers, researchers, engineers, and more.

There are two methods to create the wing sketch, one is inputting parameters, which is shown in the Figure 1.19, other is use the Profili, we choose the second method in this project.

Apply to this aircraft, we choose NACA2412, and according to calculations above, we can gain the chord root=1.71m and chord tip=0.98m. Thus use Profili, we can gain 2D of airfoils. Check the 2D file by AutoCAD, then use NX to open the 2D file, use the 2D file during modeling.

Figure 1.20 is an example to use the Profili to determine the concrete airfoil that we want by input the value of wing chord.

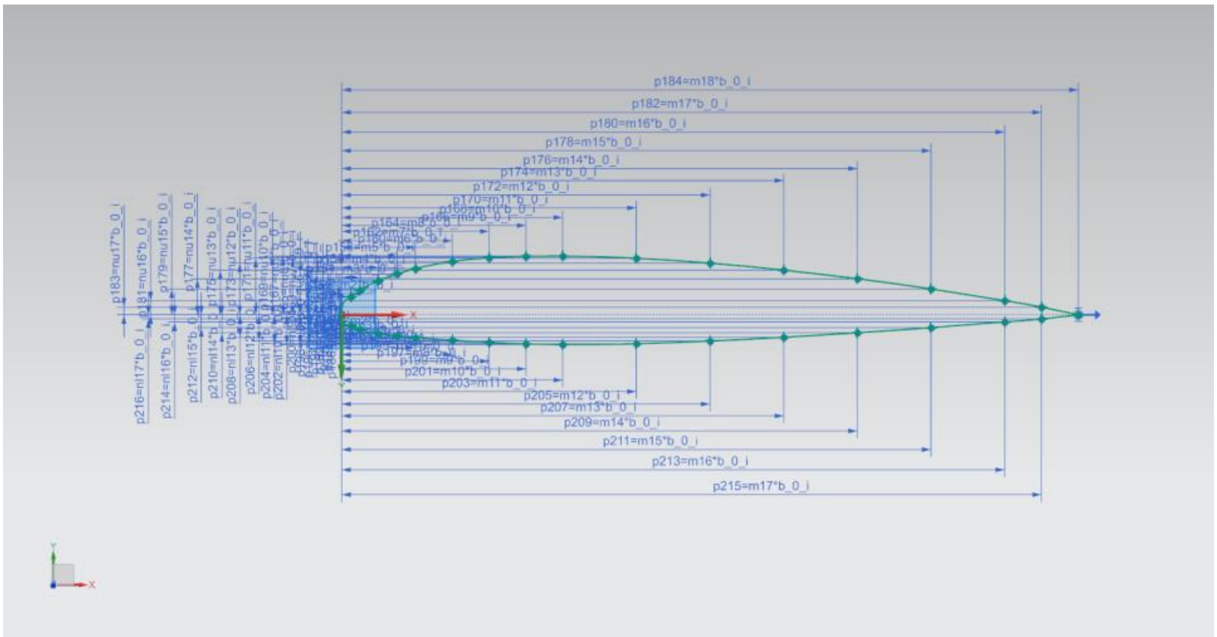


Figure 1.19 Sketch of wing root airfoil by inputting parameters


开始打印 start printing

Name of airfoil

翼型名字： NACA 2412

Chord length of the wing

翼弦 171 mm

☐ 蒙皮厚度 Skin thickness

上表面蒙皮厚度： 1.50 mm Upper Skin thickness

下表面蒙皮厚度： 1.50 mm Lowe Skin thickness

绘图线的宽度： 0.20 mm Width of the drawing line

☑ 绘出弦线 Draw the chord line

☑ 数学插补 Mathematical interpolation

☑ 显示翼型名称及相关信息. Show the name and the information related

Figure 1.20 Determine the concrete airfoil by profili

Figure 1.20 have a certain scale to real objects which is 1:10, should be noticed.

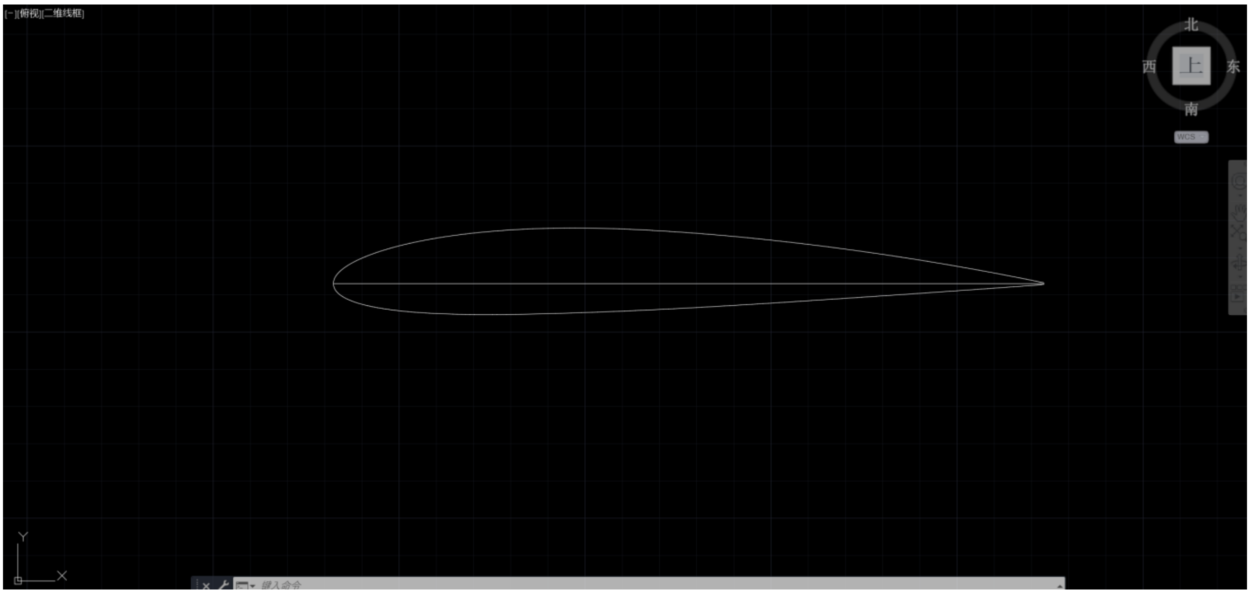


Figure 1.21 2D File of airfoil opened by AutoCAD

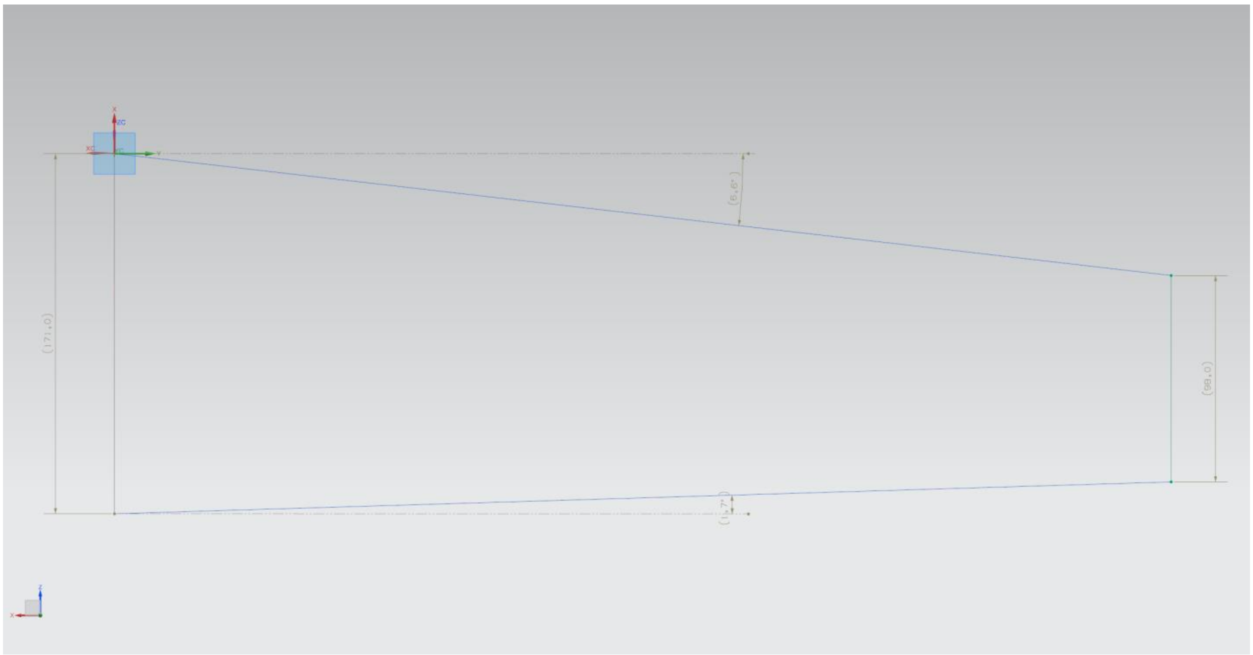


Figure 1.22 Sketch of the wing

In the Figure 1.21, when we use AutoCAD to open the 2D file of airfoil, we can measure the sacle of the airfoil to ensure if it is the right sketch that we want.

After input it in the Nx and make it right to its own place, then we can get the overview of the wing sketch as it is shown in the Figure 1.22 and Figure 1.23.

The angle of the trapezoidal wing of a light aircraft varies according to specific design requirements and usage scenarios. Generally, the leading-edge angle of the trapezoidal wing (i.e., the angle between the leading edge and the horizontal plane) is usually between 2° and 5°, and the trailing edge angle (i.e., the angle between the trailing edge and the horizontal plane) is usually between 0° and 2°.

This range is general, and in fact, it is also affected by the design requirements and usage scenarios of the aircraft. For example, if it is necessary to increase the lift coefficient of the aircraft, the leading-edge angle can be increased; if it is necessary to reduce drag, the trailing edge angle can be reduced. At the same time, it is also necessary to consider requirements such as aircraft stability and maneuverability. Therefore, the leading-edge angle of the trapezoidal wing of this light aircraft is 6.6° and the trailing edge angle is 1.7°.

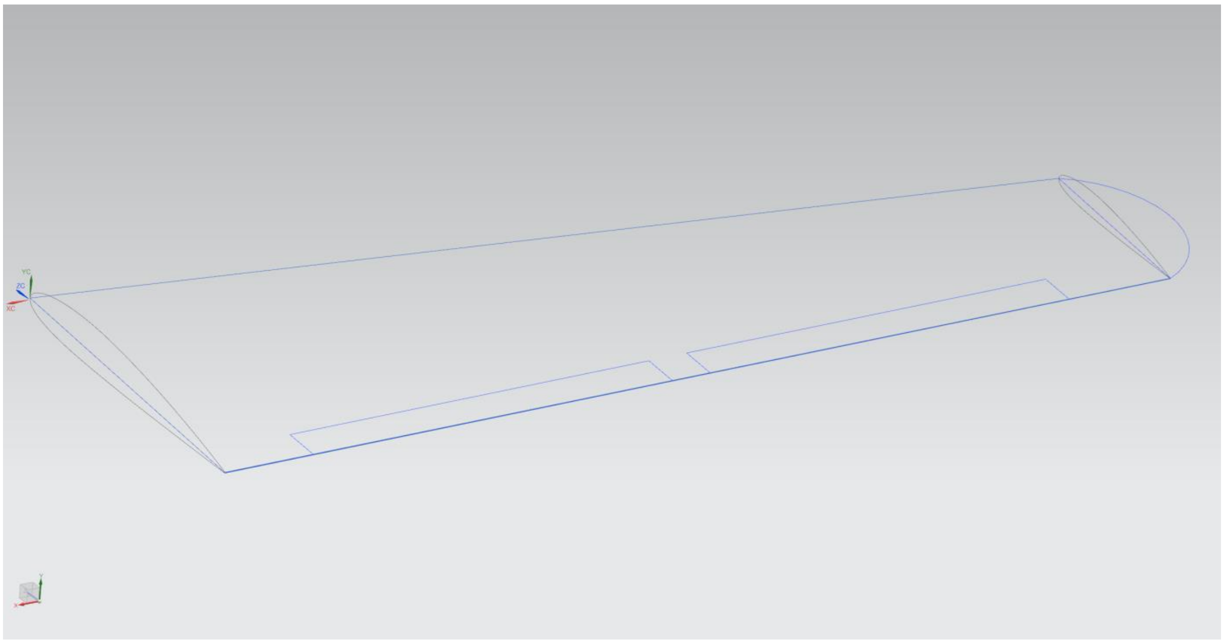


Figure 1.23 Overview of the wing sketch

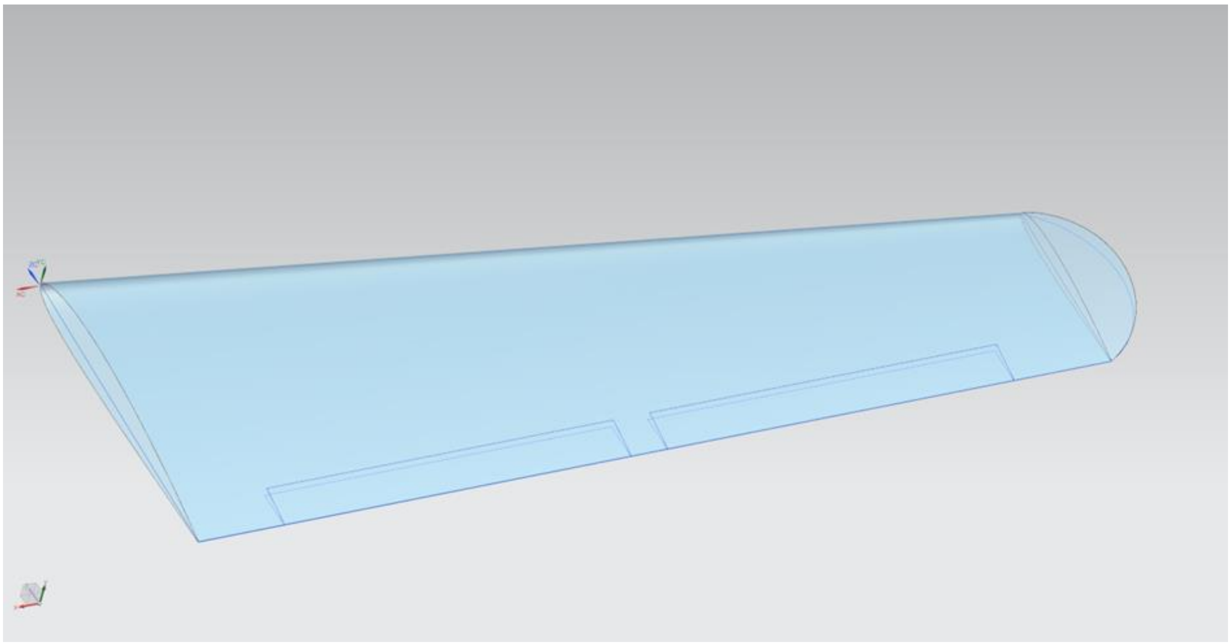


Figure 1.24 Create the surface of the wing

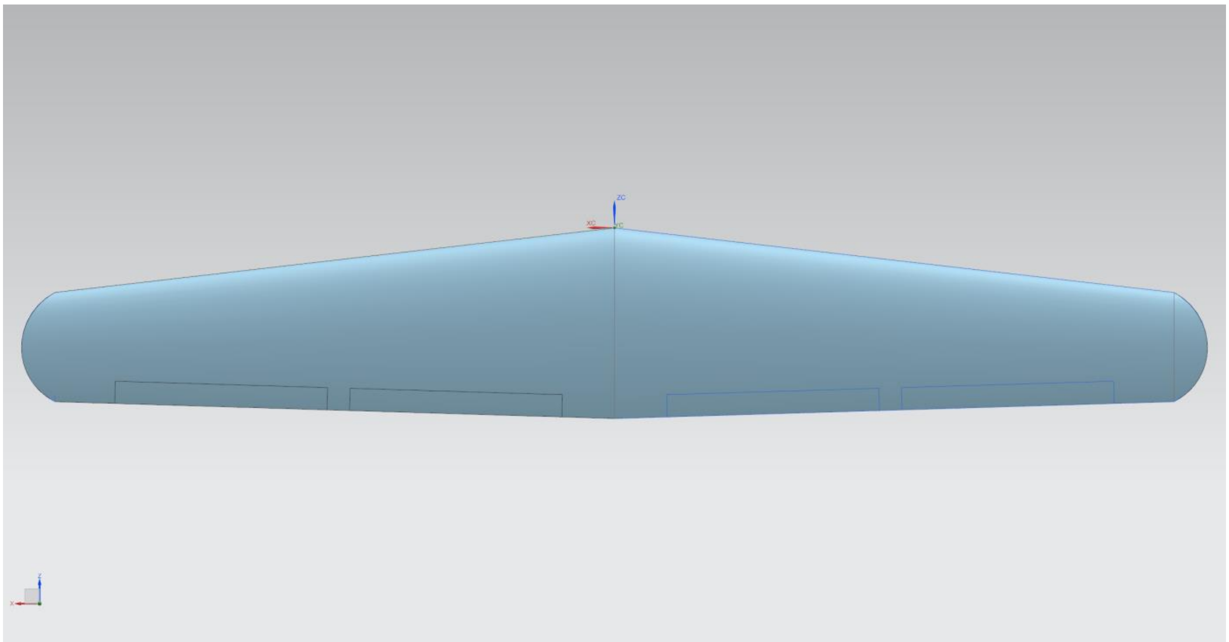
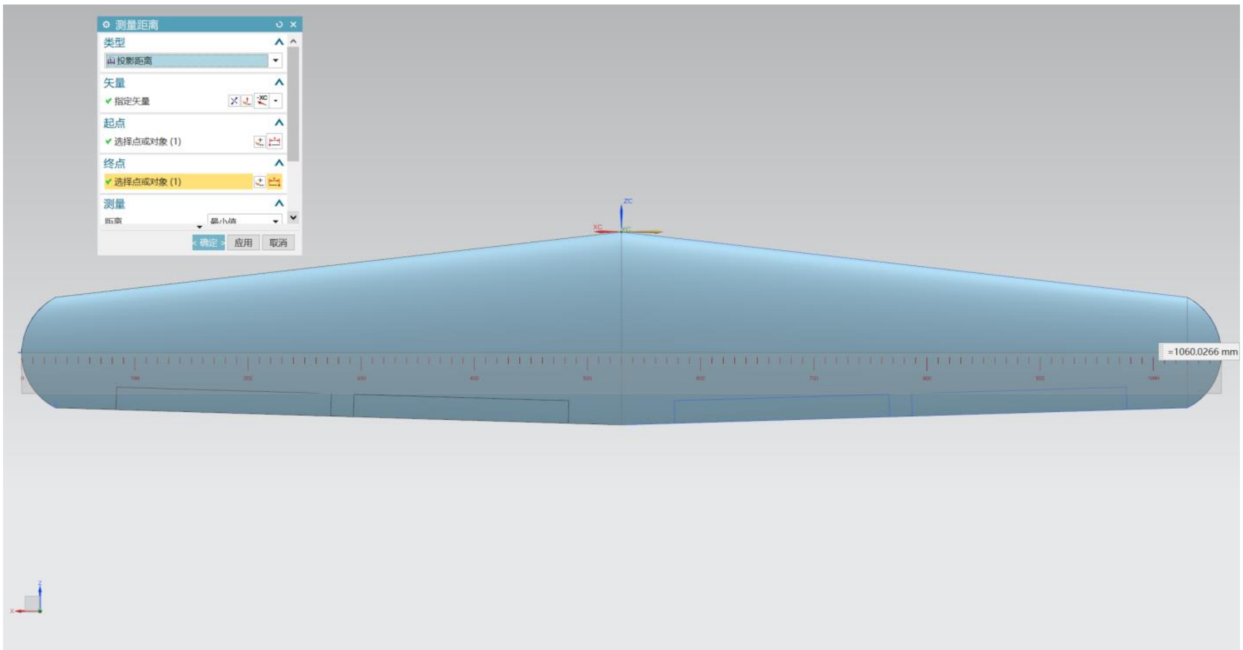


Figure 1.25 The mirror order of the wing modeling



类型: 投影距离

矢量: ✓ 指定矢量

起点: ✓ 选择点或对象 (1)

终点: ✓ 选择点或对象 (1)

测量: 距离 最小值

Figure 1.26 The measure order of the wing modeling

From Figure 1.26 the measure result we can determine the length of the wing is 10.6m, which is correspond with the calculated result.

Horizontal tail modeling

The light aircraft uses a symmetrical airfoil for its horizontal tail, typically employing the NACA 0010 airfoil. This airfoil has a thickness of only 10%, which is thinner than the airfoil used for the vertical tail, reducing drag and increasing the aircraft's speed. Additionally, this airfoil has good lift and drag characteristics, ensuring the aircraft remains stable in the air. Figure 1.28 shows the shape of airfoil use in the horizontal tail.

The concrete statistic is shown as the following:

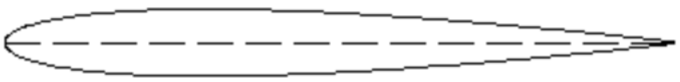


Figure 1.27 The shape of airfoil used in the horizontal tail

1. Maximum camber of 10.00% is located at 29.7% chord.
2. Maximum thickness of 0.00% is located at 0.0% chord.
3. Leading edge radius of the airfoil is 1.0931%.
4. Trailing edge thickness of the airfoil is 0.2100%.

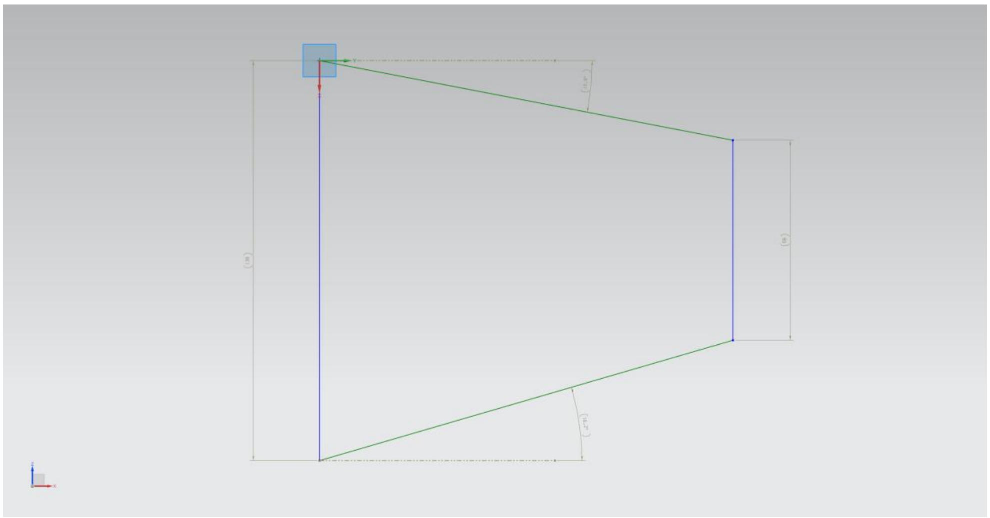


Figure 1.28 Sketch of the horizontal tail

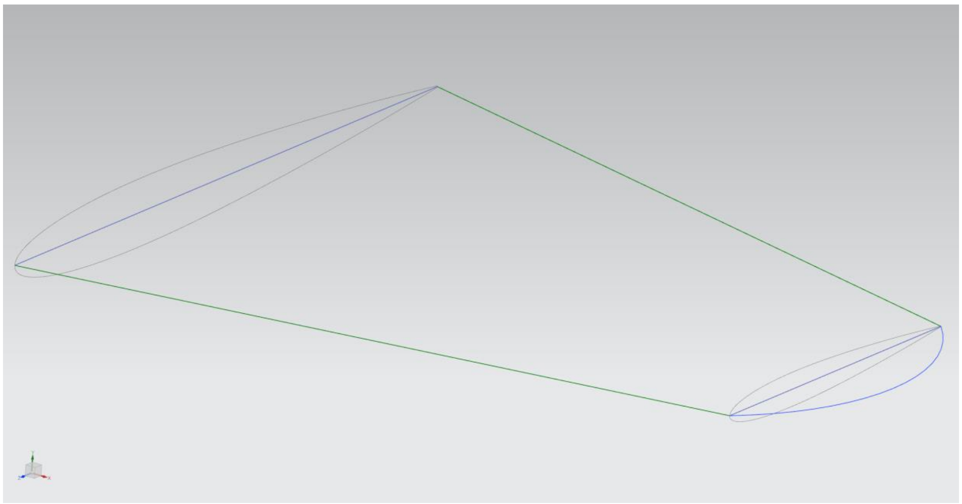


Figure 1.29 Overview of the horizontal tail sketch

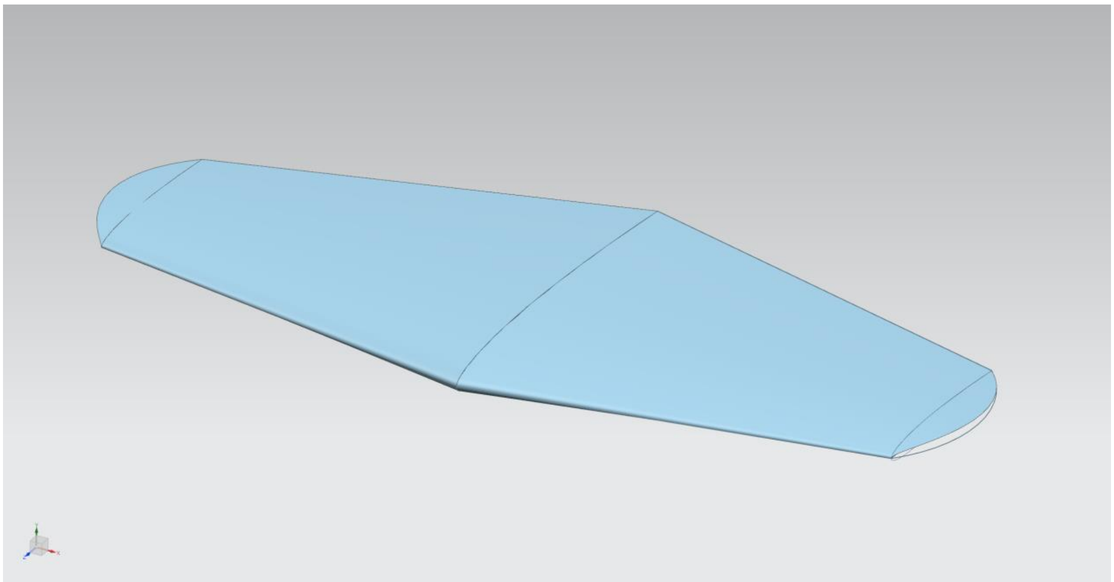
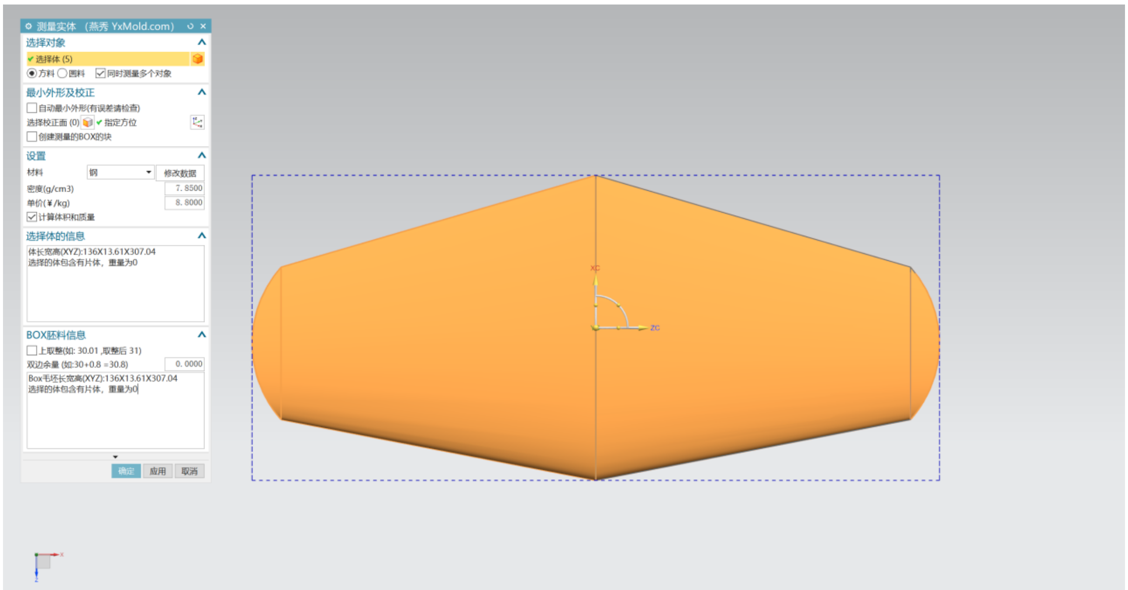


Figure 1.30 The mirror order of the horizontal tail modeling



测量实体（燕秀 YxMold.com）

选择对象

✓ 选择体 (5)

● 方料 ○ 圆料 ☑ 同时测量多个对象

最小外形及校正

☐ 自动最小外形(有误差请检查)

选择校正面 (0) ✓ 指定方位

☐ 创建测量的BOX的块

设置

材料 钢

密度(g/cm3) 7. 8500

单价(￥/kg) 8. 8000

☑ 计算体积和质量

选择体的信息

体长宽高(XYZ):136X13.61X307.04
选择的体包含有片体，重量为0

BOX胚料信息

☐ 上取整(如: 30.01 ,取整后 31)

双边余量 (如:30+0.8 =30.8) 0. 0000

Box毛坯长宽高(XYZ):136X13.61X307.04
选择的体包含有片体，重量为0

Figure 1.31 The measure order of the horizontal tail modeling

From Figure 1.31 the measure result we can determine the parameters of the horizontal tail is width 1.36m* height 0.13m* length 3.07m, which is correspond with the calculated result.

Vertical fin modeling

The initial dates of creating a parametric model of master-geometry surface of aircraft vertical tail are shown on Figure 1.32 and in this laboratory work they have the following names:

- p1 – root chord, mm;
- p2 – tip chord, mm;
- p3 – vertical tail height, mm;

- p4 – tip chord displacement, mm. This parameter can be replaced by leading-edge sweep angle in degrees;
- p5 – fin tip planform discriminant f;
- p6 – fin tip cross-section discriminant f;
- p7 – fin tip offset value, mm

The vertical tail surface is a ruled surface. It consists of two segments: fin with rudder and fin tip.

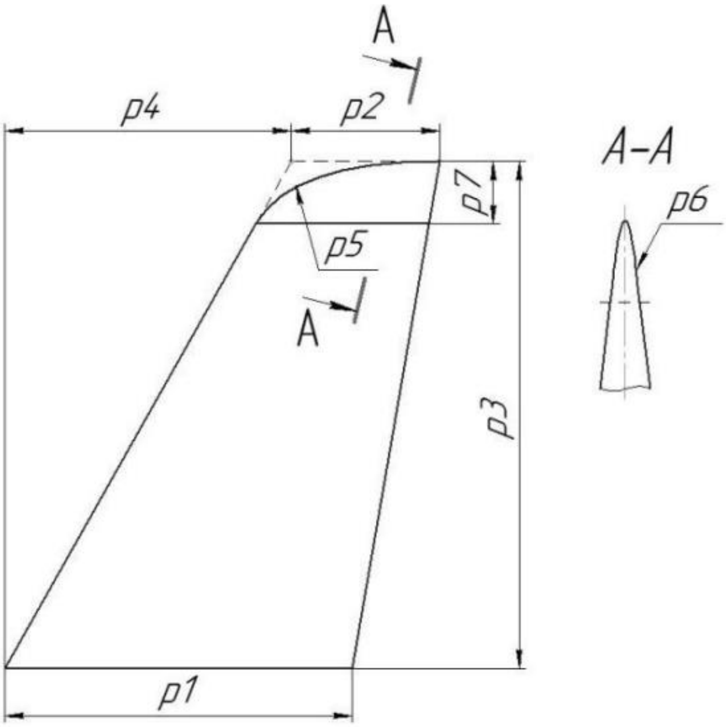


Figure 1.32 Changeable parameters of parametric model of master-geometry surface of aircraft vertical tail

In order to increase of VT efficiency, Dorsal fin installation (Figure 1.33) is used in this aircraft.

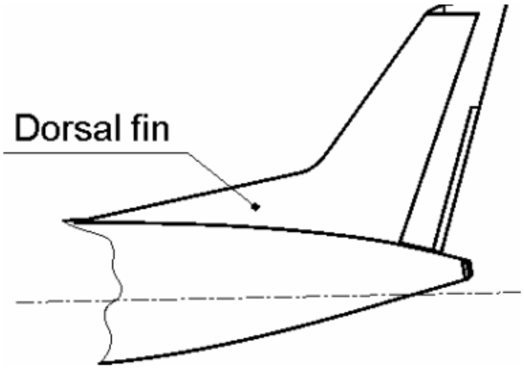


Figure 1.33 Dorsal fin

Figure 1.34 shows the first step of Vertical fin modeling is the sketch from the side view.

Figure 1.35 shows the second step of Vertical fin modeling is the sketch from the top view.

Figure 1.36 shows the third step of Vertical fin modeling is to create the surface.

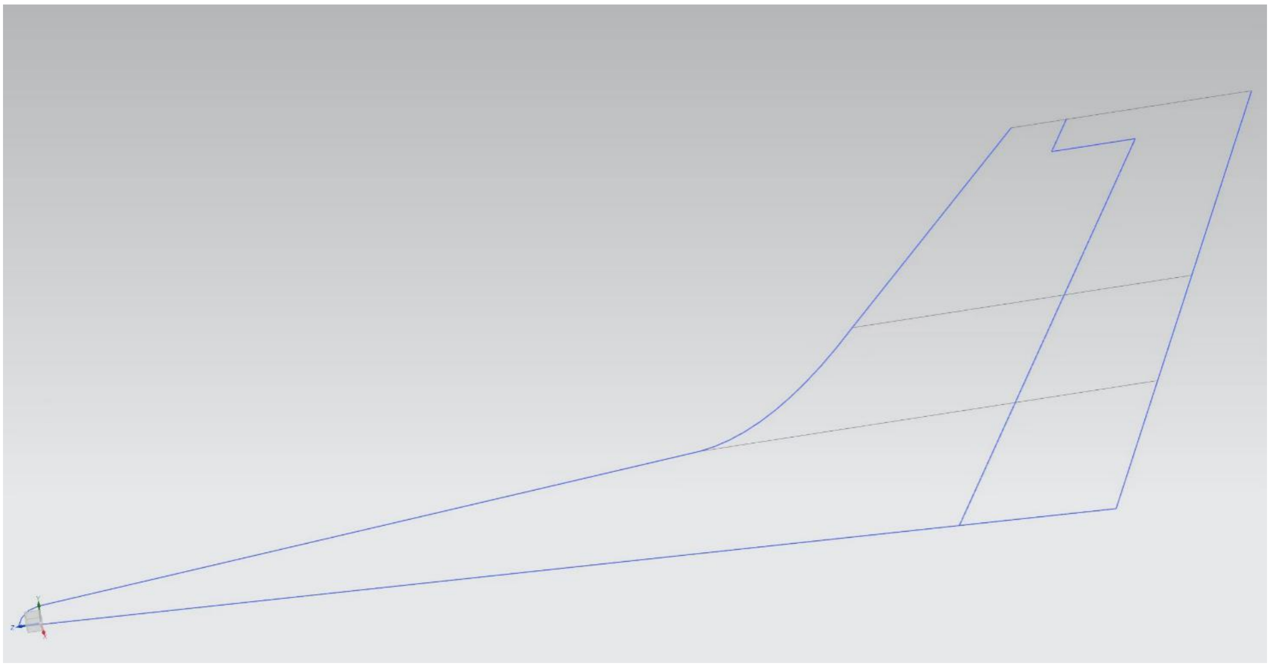


Figure 1.34 Sketch of the vertical fin from the side view

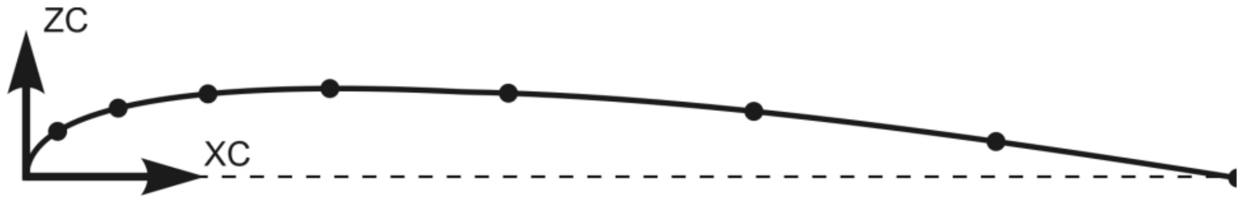


Figure 1.35 Creating NACA 0012 upper contour root airfoil

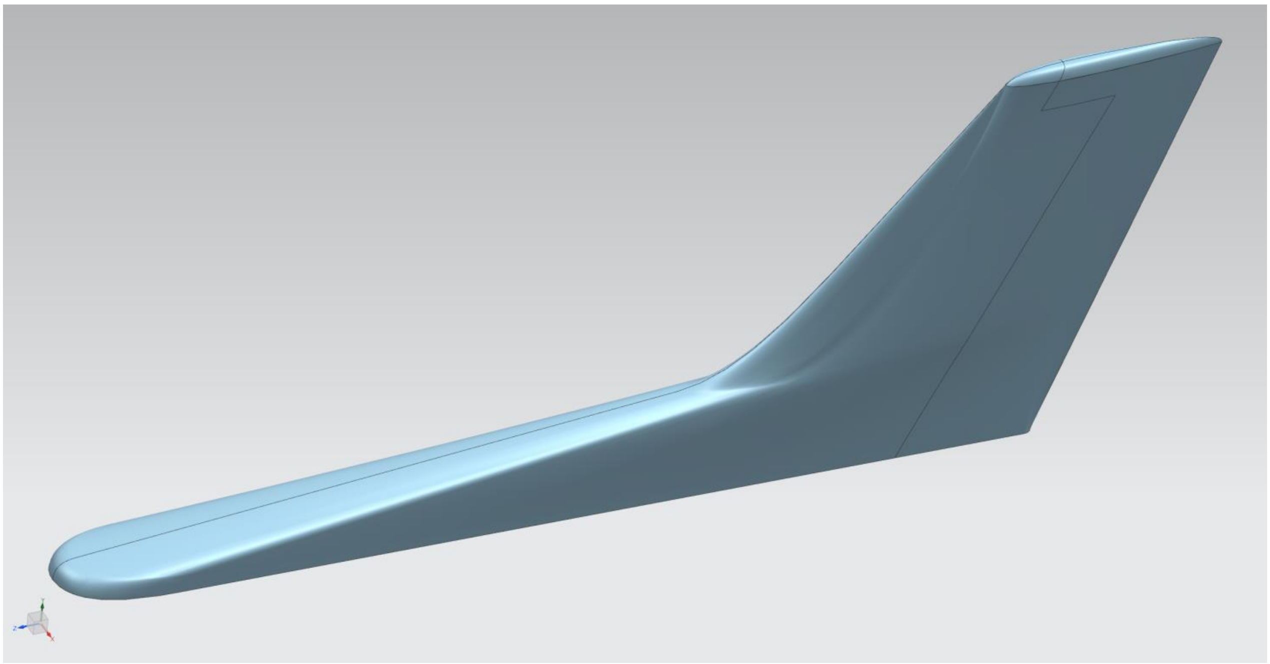
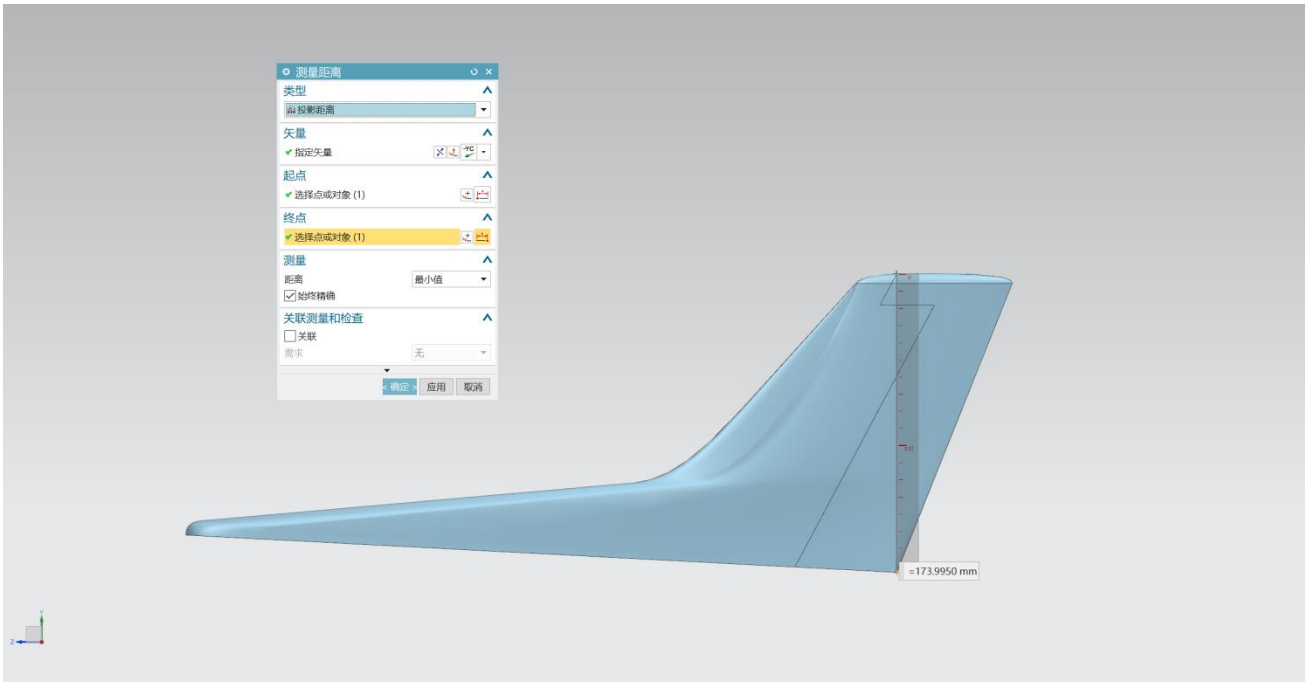


Figure 1.36 Create the surface of the vertical fin



测量距离

类型

投影距离

矢量

✓ 指定矢量

起点

✓ 选择点或对象 (1)

终点

✓ 选择点或对象 (1)

测量

距离 最小值

☑ 始终精确

关联测量和检查

☐ 关联

需求 无

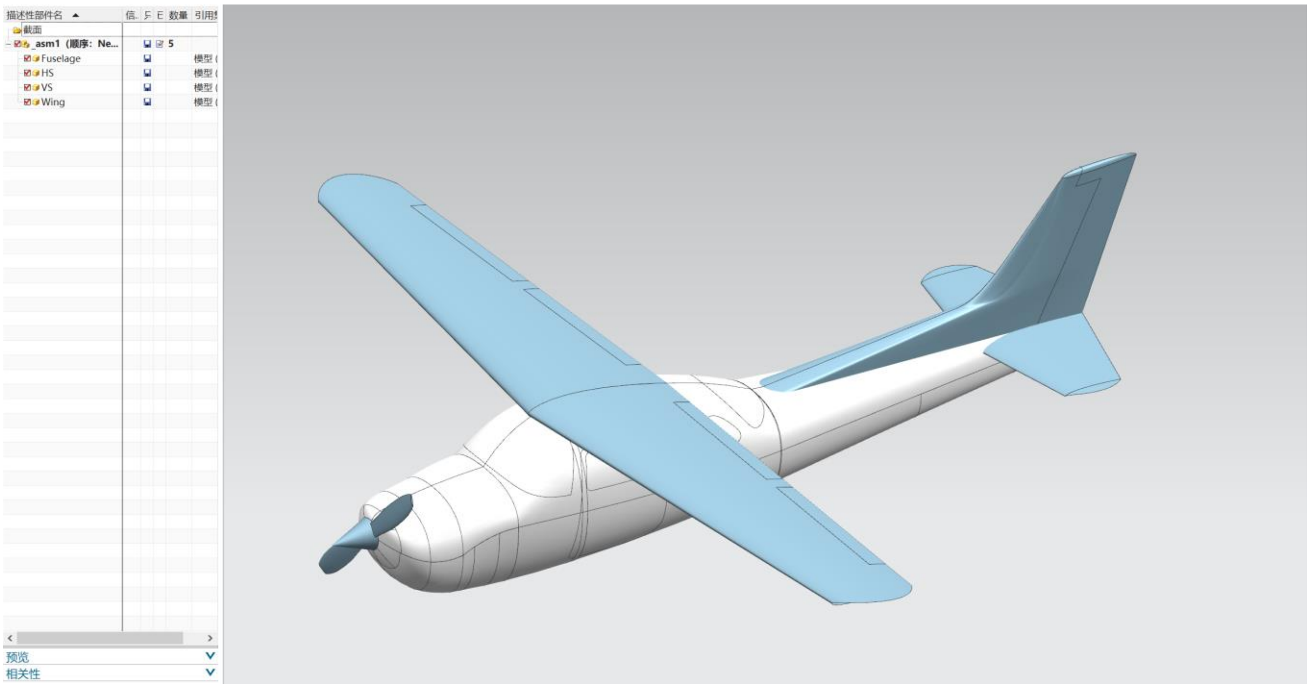
=173.9950 mm

Figure 1.37 Measure the height of the vertical fin

From Figure 1.37 the measure result we can determine the height of the vertical fin is 1.74m, which is correspond with the calculated result.

Assembly creating

Assemble all the parts of the aircraft, thus we can get the whole aircraft as is shown in the Figure 1.38.



描述性部件名	信.	𝘍	E	数量	引用集
截面					
☑ _asm1（顺序：Ne...				5	
☑ Fuselage					模型 (
☑ HS					模型 (
☑ VS					模型 (
☑ Wing					模型 (

Figure 1.38 Assembly of the whole aircraft

1.4.2 Three-dimensional parametric modeling

Three-view of the designed aircraft is created by Siemens NX after the modeling is done, this kind of file can be created by Siemens NX and imported by AutoCAD and then we can use AutoCAD to measure the detailed parameters. Figure 1.39 shows the three-view.

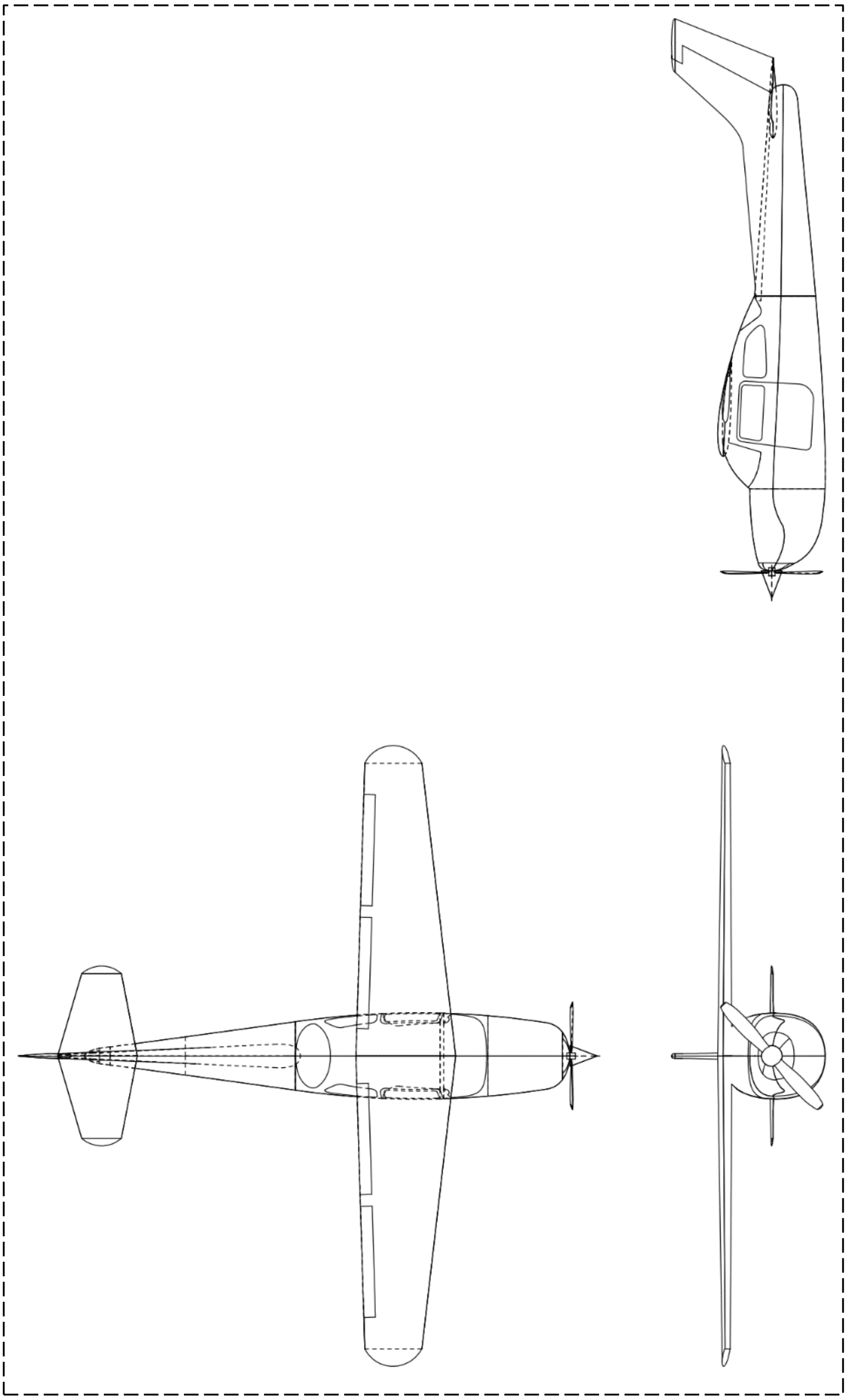


Figure 1.39 Three-views of the designed light aircraft

According to calculation, we can gain the main parameters as table1.6 describe:

Table 1.6 – Main parameters of designed aircraft

$L_{fuselage}$	$D_{fuselage}$	$H_{fuselage}$	S_{wing}	L_{wing}	$b_{t\ wing}$	$b_{o\ wing}$	
8.29m	1.46m	2.65m	14.28m^2	10.6m	0.98m	1.71m	
S_{ht}	$b_{t\ ht}$	$b_{0\ ht}$	L_{ht}	S_{vt}	$b_{t\ vt}$	$b_{0\ vt}$	H_{vt}
3.14m^2	0.68m	1.36m	3.07m	2.87m^2	1.11m	2.21m	1.74m

Figure 1.40 shows the assembly of the designed light aircraft.

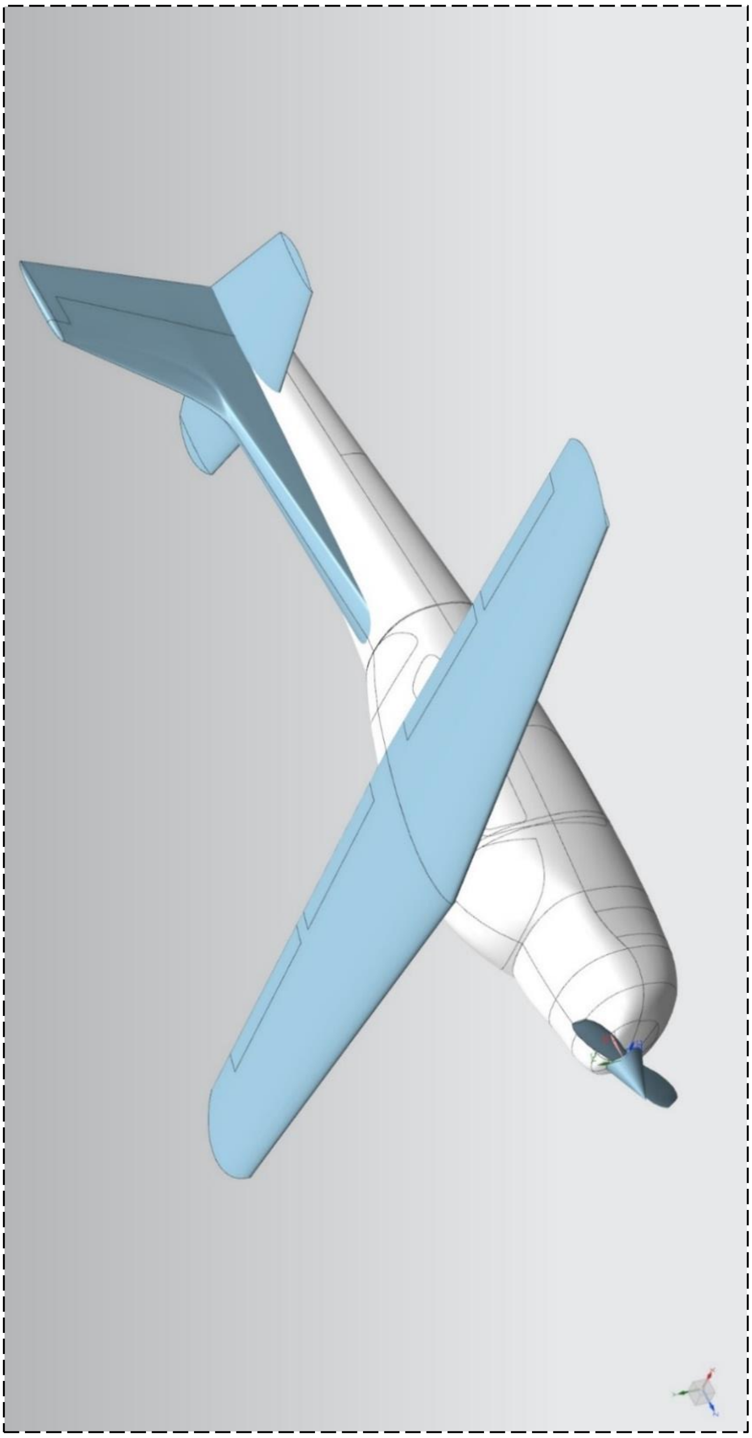


Figure 1.40 Assembly of the designed light aircraft

1.5 Application of composite sandwich panel structures in wing

1.5.1 Basic theory of composite laminates

The material properties of composite single-layer laminates

According to the mechanics of composite materials [29], the stiffness characteristics of a single-layer laminate in the axial direction are orthotropic. The expression for the stiffness matrix of a single-layer laminate in the axial direction is written as:

$$[Q] = \begin{bmatrix} Q_{11} & Q_{12} & 0 \\ Q_{12} & Q_{22} & 0 \\ 0 & 0 & Q_{66} \end{bmatrix} \tag{1.28}$$

Where:

$$Q_{11} = \frac{E_1}{1-\mu_{12}\mu_{21}},$$

$$Q_{12} = Q_{21} = \frac{\mu_{21}E_1}{1-\mu_{12}\mu_{21}} = \frac{\mu_{12}E_2}{1-\mu_{12}\mu_{21}},$$

$$Q_{22} = \frac{E_2}{1-\mu_{12}\mu_{21}},$$

$$Q_{66} = Q_{12}.$$

Among them, $Q_{11} \sim Q_{66}$ has four independent stiffness coefficients.

The off-axis characteristics of θ angle biased ply are anisotropic, and the expression for the stiffness matrix of the off-axis ply is:

$$[\bar{Q}] = \begin{bmatrix} \bar{Q}_{11} & \bar{Q}_{12} & \bar{Q}_{16} \\ \bar{Q}_{21} & \bar{Q}_{22} & \bar{Q}_{21} \\ \bar{Q}_{61} & \bar{Q}_{62} & \bar{Q}_{66} \end{bmatrix} \tag{1.29}$$

In the equation, $[\bar{Q}]$ is a symmetric matrix, so $\bar{Q}_{12} = \bar{Q}_{21}; \bar{Q}_{16} = \bar{Q}_{61}; \bar{Q}_{26} = \bar{Q}_{62}$. There is a linear transformation relationship between the off-axis stiffness matrix and the axial stiffness matrix, which is as follows:

$$[\bar{Q}] = T^T[Q]T \tag{1.30}$$

Where:

$$T = \begin{bmatrix} m^2 & n^2 & mn \\ n^2 & m^2 & -mn \\ -2mn & 2mn & m^2-n^2 \end{bmatrix}, m = cos\theta, n = sin\theta$$

According to equation (1.30), the off-axis stiffness coefficients can be seen as a linear combination of the axial stiffness coefficients, and the off-axis stiffness coefficients can be expressed as a linear combination of the axial stiffness coefficients as follows:

$$U_1 = \frac{1}{8}(3Q_{11} + 3Q_{22} + 2Q_{12} + 4Q_{66}) \tag{1.31}$$

$$U_2 = \frac{1}{2}(Q_{11} - Q_{22})$$

$$U_3 = \frac{1}{8}(Q_{11} + Q_{22} - 2Q_{12} - 4Q_{66})$$

$$U_4 = \frac{1}{8}(Q_{11} + Q_{22} + 6Q_{12} - 4Q_{66})$$

$$U_5 = \frac{1}{8}(Q_{11} + Q_{22} - 2Q_{12} + 4Q_{66}) = \frac{1}{2}(U_1 - U_2)$$

The expression shows that the single-layer off-axis stiffness with θ as a design variable forms the basis for performance-tailored design of laminated plates.

Stiffness characteristics of laminated composite plates

The physical equations of laminated composites can be represented according to the classical laminate plate theory [30]:

$$\left(\frac{N}{M}\right) = \begin{bmatrix} A & B \\ B & D \end{bmatrix} \left\{\frac{\varepsilon^0}{K}\right\} \tag{1.32}$$

Where:
$\{N\}$:The resultant force vector on the faces of a laminated composite plate;
$\{M\}$:The moment resultant matrix of the laminate plate;
$\{\varepsilon^0\}$: The matrix of in-plane strains in a laminated composite plate;
$\{K\}$: The curvature array of the mid-plane in a laminated plate;
$[A]$: The stiffness matrix for tensile loading of a laminated composite plate.

$$A_{ij} = \sum_{k=1}^{n} \left(\bar{Q}_{ij}\right)_{(k)} (Z_k - Z_{k-1}) = \sum_{k=1}^{n} \left(\bar{Q}_{ij}\right)_{(k)} h_k \tag{1.33}$$

z_k: The distance from the mid-plane to the interface between the k-th and (k + 1)-th layers is illustrated

h_k: The thickness of the kth ply in a laminate

$[B]$: The bending-coupling stiffness matrix of a laminated plate.

$$B_{ij} = \frac{1}{2}\sum_{ki=1}^{n}\left(\bar{Q}_{ij}\right)_{(k)}(Z_k^2 - Z_{k-1}^2) = \sum_{k=1}^{n}\left(\bar{Q}_{ij}\right)_{(k)}\bar{z}_k h_k \tag{1.34}$$

$\bar{z}_k$: The distance between the mid-plane of the k-th layer and the mid-plane of the layer in a laminated plate

$[D]$: The bending-torsion stiffness matrix of the laminated plate.

$$D_{ij} = \frac{1}{3}\sum_{k=1}^{n}\left(\bar{Q}_{ij}\right)_{(k)}(Z_k^3 - Z_{k-1}^3) = \sum_{k=1}^{n}\left(\bar{Q}_{ij}\right)_{(k)}\left(\frac{h_k^3}{12} + \bar{z}_k{}^2 h_k\right) \tag{1.35}$$

The analysis of the laminated plate yields three stiffness matrices: [A], [B], and [D] is as following:

1. By selecting the angles, percentage, and sequence of each individual layer in the laminated panel, the desired stiffness and strength performance of the laminated panel can be achieved;
2. Coupling phenomenon is unique to laminated panels and forms the basis for aerodynamic elasticity tailoring design of aircraft wing surfaces.

In the design of composite laminates for aircraft, symmetric balanced laminates are generally preferred, as they can eliminate the coupling between in-plane and out-of-plane deformations in laminates, while providing higher in-plane shear stiffness and out-of-plane bending stiffness, and are easy to form. For special design requirements of laminates, such as meeting specific aeroelastic needs, symmetric unbalanced laminates can be considered when symmetric balanced laminates cannot meet the design requirements. However, non-symmetric laminates are generally less considered due to the presence of significant coupling deformations, which make them less suitable for forming. Therefore, this project will focus on the research of laminate structures that meet the requirements of structural strength, stiffness, and stability by selecting appropriate laminate angles, thicknesses, and stacking sequences under the condition of symmetric balanced laminates.

1.5.2 Honeycomb sandwich panel core equivalent calculation

Due to the fact that when using engineering software for numerical calculations, the material properties of the honeycomb core in a sandwich structure cannot be directly given. It needs to be calculated equivalently as a single-layer plate structure, that is, the sandwich plate is simulated using an equivalent plate. If the panel is a composite laminated panel, the honeycomb core can be treated as a special single layer, and then the sandwich plate can be simulated using an equivalent composite laminated panel [31]. The honeycomb core equivalent calculation in this project is based on the theory of sandwich panels [32], which can be equivalently represented as a homogeneous orthogonal anisotropic single-layer plate. Figure 41 shows a schematic of a hexagonal honeycomb. The equivalent elastic parameters are expressed as:

$$E_x = E_y = \frac{4}{\sqrt{3}}\left(\frac{t}{l}\right)^3 E \tag{1.36}$$

$$E_z = \frac{2\gamma}{\sqrt{3}}\frac{t}{l}E$$

$$G_{xy} = \frac{\sqrt{3}\gamma}{2}\left(\frac{t}{l}\right)^3 E$$

$$G_{xz} = \frac{\sqrt{3}\gamma}{2}\frac{t}{l}G$$

$$G_{yz} = \frac{\gamma}{\sqrt{3}}\frac{t}{l}G$$

$$\mu_{xy} = \frac{1}{3}$$

Where:

E: The elastic modulus of the sandwich material;

G: The shear modulus of the sandwich material;

l: The length of the side of the honeycomb cell wall;

t: The thickness of the honeycomb cell wall;

γ: The correction factor, which depends on the process, usually taken between 0.40 to 0.60. In this project, it is taken as 0.50.

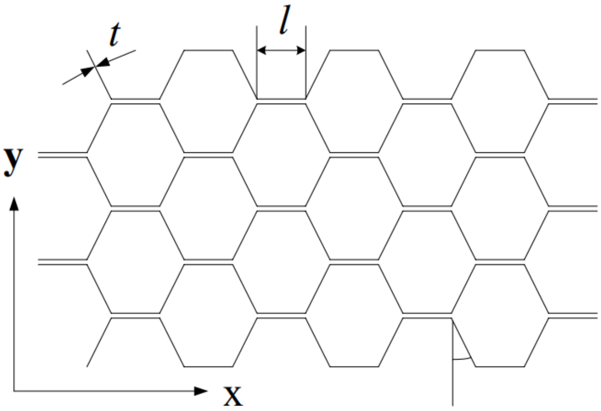


Figure 1.41 Honeycomb sandwich schematic and cell diagram

(In the figure 1.41, l is the edge length of the honeycomb core, t is the thickness of the honeycomb core wall, and θ is the internal angle value of the honeycomb core cell. For a regular hexagonal honeycomb, θ=30°)

The honeycomb core selected in this project is Nomex aramid honeycomb, model NRH-3-48. The elastic modulus of the material is 2GPa, the shear modulus is 0.623GPa, the thickness of the honeycomb core material is 0.1mm, and the honeycomb edge length is 3mm. The equivalent elastic parameters of the honeycomb core were calculated using the equivalent formula, and the results are shown in

Table 1.7. This result will be used for the subsequent structural design.

Table 1.7 – Equivalent Parameters Calculation Results for Nomex Honeycomb Core

Parameter	E_x/MPa	E_y/MPa	E_z/MPa	G_{xy}/MPa	G_{xz}/MPa	G_{yz}/MPa	μ_{xy}
Vaule	0.171	0.171	38.491	0.032	8.991	5.994	0.33

1.5.3 Overall design scheme for composite wing structure

Initial design conditions and wing geometry parameters

The main performance parameters and conditions of the entire aircraft related to the wing design given by the overall design are:

The total weight of the aircraft is 326.48Kg, the wing load factor is 4.4, the safety factor is 2, the NACA 2412 airfoil is selected, the wing is designed as a high wing, there is no fuel tank inside the wing, and there is no landing gear installed on the wing.

The following are the wing external design parameters determined by the aircraft aerodynamic requirements:

- Wing area is 14.28m^2;
- Wing span is 10.6m, and the body width is 1.46m;
- Aspect ratio is 7.87;
- Root chord length is 1.71m;
- Tip chord length is 0.98m;
- Wing installation angle is 0°.

Material selection and design requirements for aircraft wing structure

Principles for selecting materials in aircraft composite design [16]:

1. Mechanical performance: The material must have sufficient strength and stiffness to withstand the load and ensure safety.
2. Durability: The material must be able to withstand long-term use and environmental changes, such as temperature and humidity.
3. Weight: The material must be lightweight to reduce the weight of the aircraft and fuel consumption.
4. Production cost: The cost of the material should be reasonable to ensure that the production cost is not too high.
5. Processability: The material must be easy to process to ensure production efficiency and quality.
6. Reliability: The material must have sufficient reliability to ensure normal operation under different conditions.
7. Environmental sustainability: The material must meet environmental requirements to ensure environmental sustainability.

Based on the above principles and the ideas presented in previous sections, the aircraft wing designed in this project uses a combination of composite laminates and sandwich panels, which can effectively reduce the weight of the wing structure while ensuring its strength, stiffness, and stability. The layering material used in this project is carbon fiber/epoxy prepreg, which is currently a mature material. The thickness of a single layer is 0.15mm, and its performance data is shown in Table 1.8. The honeycomb core used is Nomex honeycomb, and its mechanical properties are shown in Table 1.9. The theoretical equivalent anisotropic single-layer plate parameters of the sandwich panel are shown in Table 1.8.

Table 1.8 – Unidirectional carbon fiber/epoxy prepreg performance data

Density(kg/m^3)	E_{11}(GPa)	E_{22}(GPa)	G_{12}(GPa)	G_{23}(GPa)	G_{13}(GPa)
1450	135	10.3	6.6	2.6	6.6

Table 1.8 – (Continue)

v_{12}	X_t(MPa)	X_c(MPa)	Y_t(MPa)	Y_c(MPa)	S(MPa)
0.21	1548	1226	55.5	218	89.9

Table 1.9 – Mechanical properties of honeycomb core

Planar Compressed		Longitudinal Shear		Transverse Shear	
Strength(MPa)	Modulus	Strength(MPa)	Modulus	Strength(MPa)	Modulus
Average	Average	Average	Average	Average	Average
1.81	107.6	1.15	36.9	0.70	23.8

After determining the layout of the wing structure and the selected materials, specific design requirements need to be drafted for the wing structure, which mainly include the following aspects:

1. Design requirements for static strength of the structure.

Generally, the allowable stress for static strength design of metal materials is expressed as stress, known as allowable stress. Because metal materials are isotropic materials, and the stress-strain relationship is linear below the yield stress. For metal materials, allowable stress and allowable strain are consistent. However, fiber-reinforced composite materials are completely different from metal materials, and their single-layer mechanical properties are anisotropic. The longitudinal strength and modulus are nearly two orders of magnitude higher than the transverse strength and modulus, but the failure strain of the longitudinal and transverse directions are in the same order of magnitude and differ little. Moreover, the strain distribution of laminates along the thickness direction is linear, but the off-axis stiffness of a single layer is different, making the stress distribution along the thickness direction irregular. Laminates with the same thickness but different lay-up directions can have significantly different failure stresses, while the failure strains do not differ much. Therefore, the design allowable value for the static strength of laminates is selected as strain, known as the design allowable strain. In this project, the single-layer carbon fiber/epoxy prepreg composite material used has a generally allowable strain value of under the design load:

- The compressive design allowable strain≤3200με;
- The tensile design allowable strain≤3200με;
- The longitudinal and transverse shear design allowable shear strain≤5600με.

The allowable tensile and compressive strains for the Nomex honeycomb sandwich structure used in this project are set at 1000με.

2. Structural stiffness design requirements

The stiffness design requirements for aircraft structures should ensure that harmful deformations do not occur under operating loads. Maximum deformation should not impede normal aircraft operation, and deformations should not cause significant changes in external loads or internal force distributions. According to the wing deformation requirements for aircraft, it is assumed in this project that the deformation at the wingtip should not exceed 15% of the wing half-span. The designed wing half-span is 5000mm, so the deformation at the wingtip is controlled within 700mm.

3. Structural Stability Requirements

Although experimental results have shown that composite laminates still have bearing capacity after buckling, the effect of over-buckling on fatigue performance is still difficult to evaluate. Therefore, the stability design requirement for the wing structure in this project is that the overall instability of the wing structure is not allowed to occur under the design load, that is, the minimum characteristic value of overall instability should be greater than 1.

4. Design requirements for composite laminate:

When designing a laminated plate, the first step is to follow the general principles of laminate design to ensure that the strength and stiffness requirements are met under various design loads, and that buckling does not occur under specified loads. Based on this, favorable laminate designs are considered to maximize the load-bearing capacity of the laminated plate while meeting strength, stiffness, and buckling requirements.

1.5.4 Design of composite material wing structure

The following primary structural members take acting loads in wing: spars, stringers, skin, ribs, and false spars. Character of work of these members under load defines load-carrying structure.

The work nature and degree of usage of skin and longitudinal elements (spars and stringers) in bending are taken as the main criterion (distinctive feature) characterizing wing load-carrying structures. Wing load-carrying structures are subdivided onto spar-type, torsion-box and Monoblock load-carrying structures.

In spar-type wing load-carrying structures bending moment is taken by spars mainly. Skin in such structure is thin, may not work (means made of fabric), or it is partially working (meaning takes only flow of tangential forces caused by torsion torque) or it works and takes tangential forces and a share of bending moment together with stringers. Skin also takes air aerodynamic force and transmits it to stringers and ribs. Spar webs work in shear caused by cross-cut force and tangential forces in twisting. Stringers reinforce skin having increased its critical stresses under compression and shear, and together with skin take a share of bending moment. Spar webs together with skin form contour taking torsion torque. False spars close contour, and they are used as supports for hinge fittings of ailerons and flaps.

Spar-type wings are subdivided onto single-spar, double-spar and multi-spar types. Number of spars is defined by wing planform, its dimensions, acting loads, joints and others.

In single-spar as shown in Figure 1.42 and Figure 1.43 wing total bending moment and cross-cut force in wing are taken by spar arranged in the place of maximum airfoil construction height ($X_{sp.}$ = 0.2...0.3b for low-speed aircraft and $X_{sp.}$ = 0.4...0.5b for high-speed aircraft). Rear false spar is placed at distance of 0.65...0.7b from airfoil nose. Torsion torque $M_{trq.}$ is taken by two contours I and ll and distributed between them approximately in compliance with formula:

$$\frac{M_{trq.I}}{M_{trq.II}} = \frac{\omega_I^2}{\omega_{II}^2}, M_{trq.I} + M_{trq.II} = M_{trq.} \qquad (1.37)$$

Here ω_I and ω_{II} mean areas limited by spar web and skin of I II I and II contours.

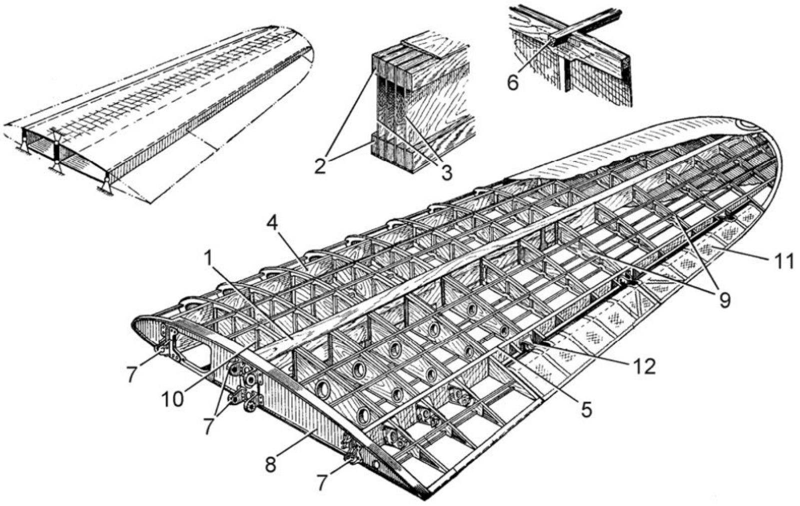


Figure 1.42 Design of Single-spar Wing

1 – spar, 2 – spar cap, 3 – spar web, 4 – front false spar, 5 – rear false spar, 6 – stringer, 7 – joints, 8 – end rib, 9 – ribs, 10 – intermediate rib, 11 – aileron,12 – aileron hinge fitting

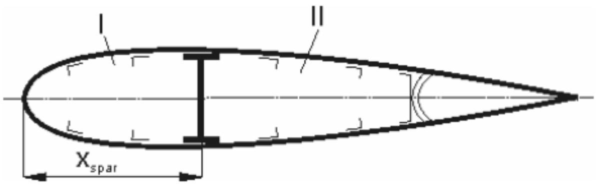


Figure 1.43 Sketch of Single-spar Wing Airfoil with Rear False Spar

According to the above wing layout and design requirements, two different layouts of the single-spar structure were chosen for this aircraft. Since the left and right wings are symmetrical, only one wing is studied in this project.

Scheme One: Front-Spar and Rear-Wall Wing. This layout has a front spar located 20% from the leading edge, which is the main longitudinal load-bearing member of the wing skeleton. The rear wall is located 70% from the leading edge, and together they form a closed chamber that resists torsion and transfers wing loads. There are five ribs arranged along the span, with no reinforcing ribs and no stringers on the skin. The overall wing frame model is shown in Figure 1.44.

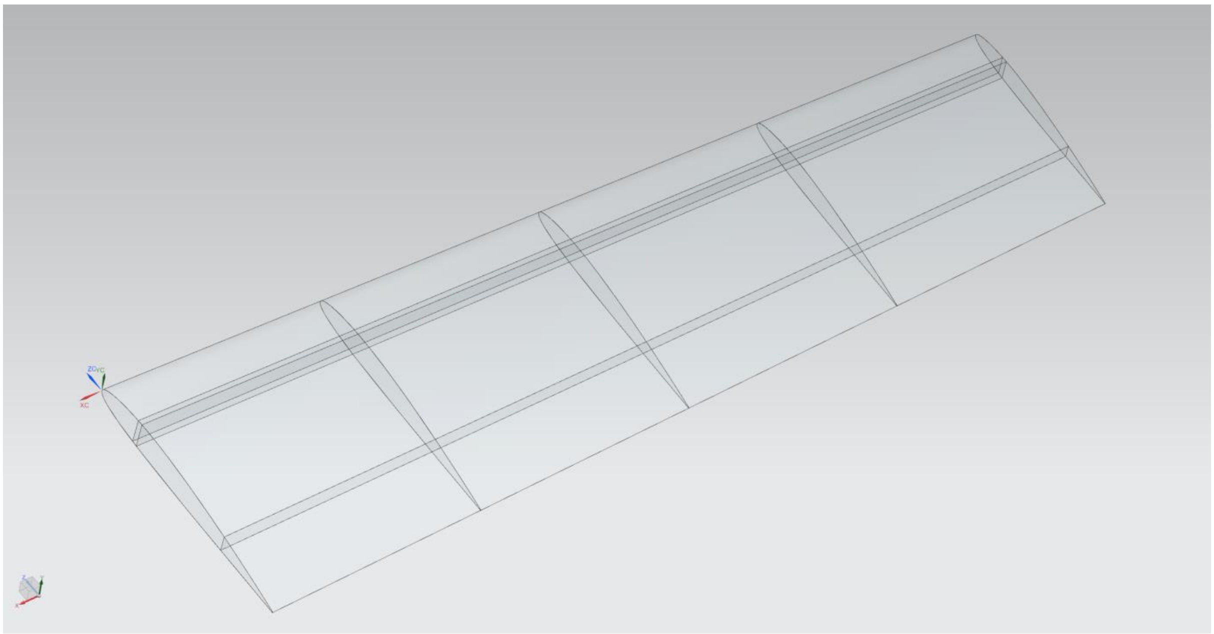


Figure 1.44 Sketch of Front-Spar and Rear-Wall Wing

Scheme 2: Single-Spar Double-Wall Wing. The spar is located at the maximum airfoil construction height of 29.9%, and the front wall is located 8% from the leading edge, while the rear wall is located 70% from the leading edge. They are the main longitudinal force-bearing members of the wing skeleton, and the spar and walls form two anti-twist closed chambers to transmit the wing load. Five wing ribs are also arranged along the span, without reinforcing ribs, and there are no stringers on the skin. The overall wing frame model is shown in Figure 1.45.

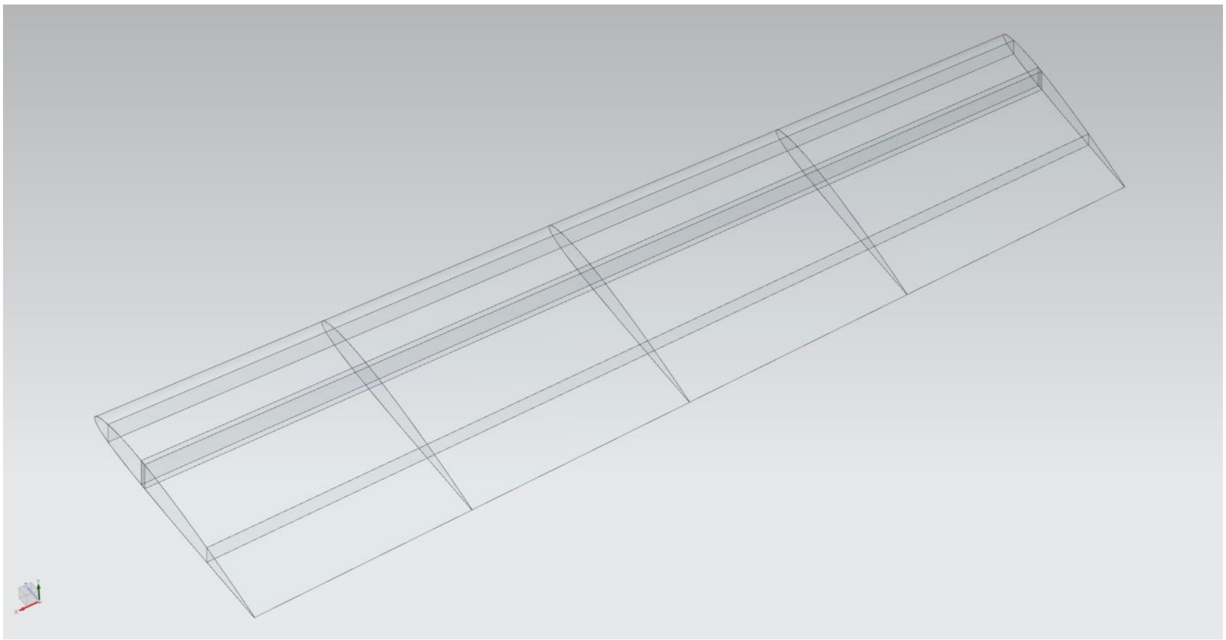


Figure 1.45 Sketch of Single-Spar Double-Wall Wing

1.5.5 Design of details for composite material wing structure

Due to the unknown initial dimensions of the wing structure, the initial conditions of the wing structure given in this project are relatively small, or in other words, they are only the basic laminate thickness. The laminate angles chosen are 0°, 90°, ±45°, and the laminate sequence is simply arranged according to these four angles. In order to reduce data and save computing time, the same initial laminate values are used for the same structures in both scheme 1 and scheme 2. The following are the detailed initial laminates for the spar, wall, rib, and skin.

- Wing Spar

Spars are the most important bearing elements of wing longitudinal set. They are mad in a form of double-cap thin-wall beam, which caps work in compression-tension under action of bending moment, webs (struts and braces of truss-type load-carrying structure) take cross-cutting force and together with skin form contour, in which flow of tangential forces appear due to action of torsion torque. Spar mass is about 25…50% of wing mass or about 4…5% of airplane take-off mass.

Relatively to structural design spars may be of beam-type (Figure 1.40, a), truss-type (Figure 1.46, b) and combined beam-truss type. Truss-type spars include caps, spar web upright members, braces, which are reasonable for large construction heights of airfoil and low intensity of loads.

Beam-type spars are the most widely used, which have two caps, web reinforced with upright members.

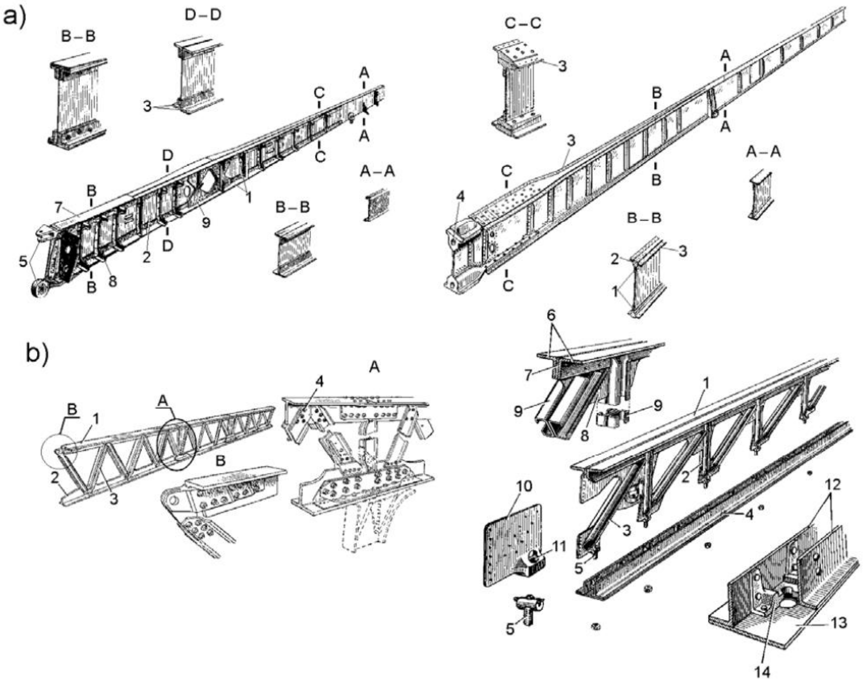


Figure 1.46 Examples of Spar Load-Carrying Structure

The wing spar adopts a "I" type integral carbon fiber composite structure, as shown in Figure 1.47. As the main longitudinal load-bearing component, the structural stiffness and strength requirements are high. The total length of the wing spar is 5000mm, in scheme 2, the height of the spar root is 205.3mm, and the height of the spar taper section is 117.7mm. The layup of the flange is: [±45/±45/0/90/0/90/±45]s, a total of 20 layers, with a thickness of 3mm; the layup of the web is: [0/90/±45/0/-45/90/-45/0/90]s, a total of 20 layers, with a thickness of 3mm.

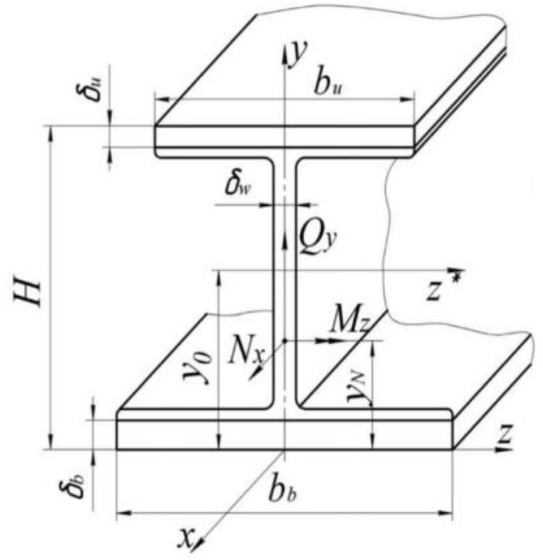


Figure 1.47 Cross section of the I-beam

- Wing Wall

The wall is a beam with weakened flanges and is also a longitudinal load-bearing member as shown in Figure 1.48. In this project, the flanges are ignored, and the wall is only considered as a honeycomb sandwich structure reinforced by webs. In scheme 2, the height of the wall 1 root is 147.7mm, and the height of the wall 1 taper is 82.8mm. The height of the wall 2 root is 125.4mm, and the height of the wall 2 taper is 72.7mm. The wall has a layup of [±45/±45/0/90/core]s with a total of 15 layers and a thickness of 8.1mm, where the height of the honeycomb core is 6mm.

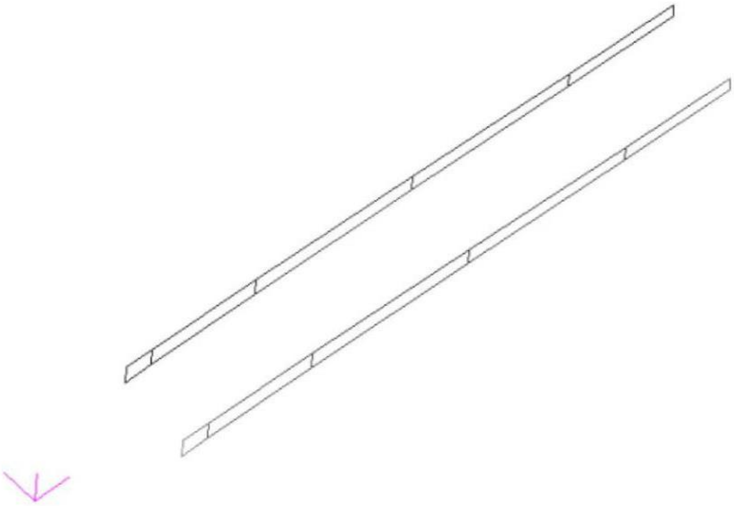


Figure 1.48 Wing Wall

- Wing Rib

Ribs are the transverse load-bearing elements, and this wing structure adopts 5 wing ribs distributed along the span, as shown in Figure 1.49. Due to the honeycomb sandwich structure, the ribs can also be regarded as web structures without considering the flanges. In order to reduce the calculation workload, the rib structure adopts the same form of stacking, with layers of [±45/±45/0/90/core]s, totaling 15 layers and a thickness of 8.1mm, with a honeycomb core height of 6mm.

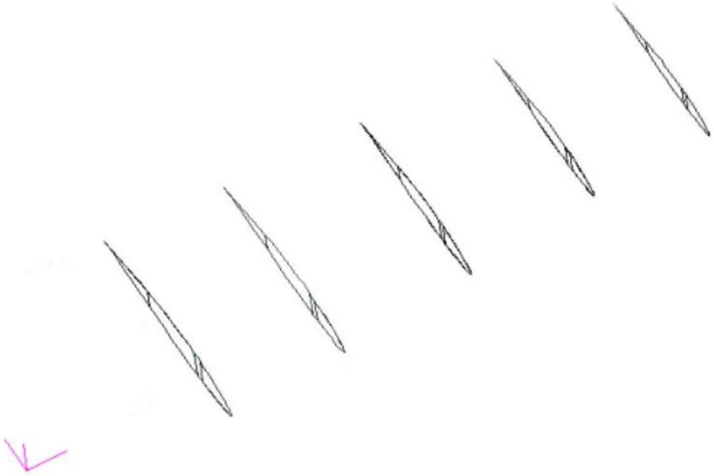


Figure 1.49 Wing Rib

- Wing Skin

Skin forms the surface of the wing and its function is to maintain the wing's shape as shown in Figure 1.50, directly withstand aerodynamic loads and transfer them to the longitudinal and transverse load-bearing components of the wing In this project, the skin adopts a honeycomb sandwich structure, with a layer structure of [±45/±45/0/90/core]s, a total of 15 layers, and a thickness of 8.1mm, where the honeycomb core has a height of 6mm.

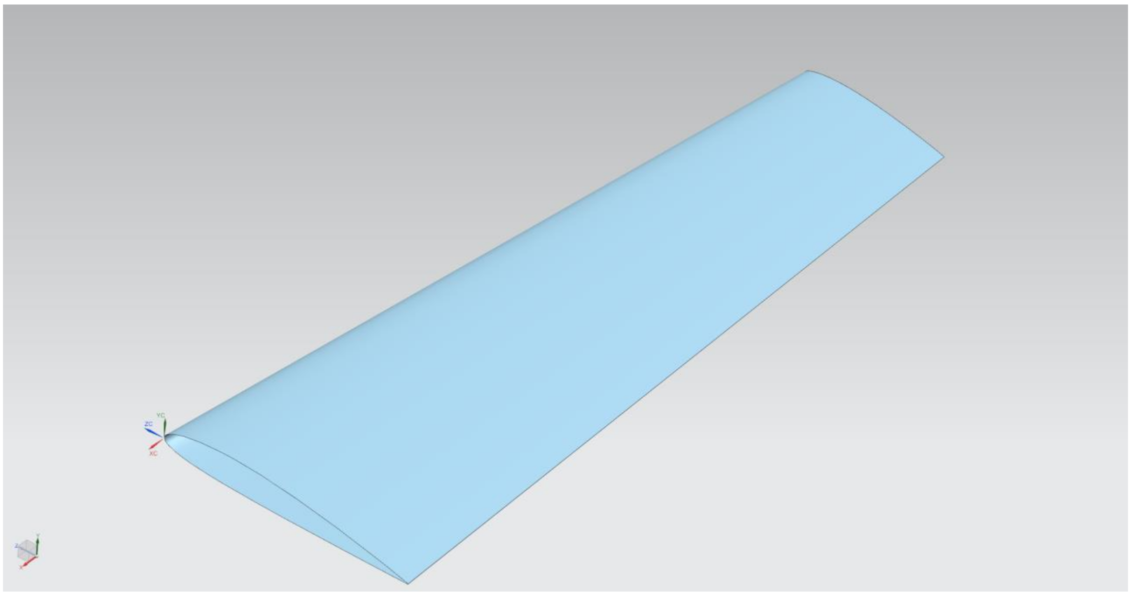


Figure 1.50 Wing skin

1.6 Conclusion of design section

Aircraft structural design must meet the requirements of strength and stiffness. Meeting strength requirements means that the aircraft structure can meet safety regulations under the external loads in various flight conditions. Meeting stiffness requirements means that the stiffness of the aircraft structure can ensure that there is no aeroelastic instability within the range of use, and the deformation meets the requirements. The structure of light aircraft is simple, but the requirements for safety are not simple. This project presents the digital prototype and structural model of a light aircraft and designs the composite material layup for the wing.

Conclusions:

1. The structural design of the light aircraft is simple, practical, and low cost. The vast majority of the aircraft structure is made of composite materials, with only the center wing and some connecting parts made of metal to ensure that the aircraft has sufficient strength and rigidity. The fuselage adopts a skin-type frame structure, which is simple in form and easy to manufacture, and conforms to the current trend of development of light aircraft.

2. The sandwich structure, as a special form of structure, is widely used in aerospace structures due to its high specific strength and stiffness. This project adopts the theory of sandwich structure and transforms the honeycomb core of the honeycomb sandwich structure into a homogeneous orthotropic single-layer plate of composite laminate, so as to reduce the complexity of calculation.
3. According to the structural design requirements of the composite wing of the light aircraft, two new structural schemes are proposed based on the existing aerodynamic shape and wing layout: the front spar and rear spar wall wing structure and the single spar double wall wing structure.
4. For the two proposed wing structure schemes, a more specific structural design was carried out. The laminated panel material and the honeycomb sandwich structure material were selected, the honeycomb sandwich core material specifications were determined, and the equivalent elastic constants of the honeycomb sandwich structure were calculated. The wing structure was designed in detail, and the initial laying thickness and sequence of the wing's load-bearing components and skin were determined.

Further work outlook:

1. The large commercial finite element software MSC.NASTRAN can be used to perform static strength analysis of each structural component, and the stress and strain values of each element and each layer of the composite material can be obtained. A program written in MATLAB can be used to check the calculation results according to the Tsai-Hill strength theory, in that case, the strength of the composite material structure can be considered under wet and hot and impact environments.
2. The main wing, and fuselage structure of the aircraft can be discretized into finite element models for analysis. The analysis results can determine whether the stress distribution of the main wing is reasonable or not, whether there is stress at the root of the main wing, and whether the calculated values of strength and stiffness are within the allowable range. After considering the effects of humid and impact environments, the critical locations of each component can be identified based on the calculated strength ratio. According to the calculation results, it can be concluded that the strength and stiffness of the main wing, and fuselage structures are sufficient, and the structural deformation meets the requirements. The fuselage structure ensures a large residual strength.

2 Economic Section

2.1 Initial data

- No. of working days in a month (W_{days}) = 22 days/month.
- Worker’s salary (W) = 44000 UAH/month.
- Total no. of days engineer works in this project (F) = 44 days.

2.1.1 Calculations of expenses spent on research of the project

Expenses during the course of this project is shown in the table below:

Table 2.1 – Total expenses on the project

NO.	Name	Expense (UAH)	Quantity	Total expense (UAH)
1	Computers	20000	3	60000
2	Working place	10000	1	10000
3	Stationeries	500	1	500
4	Software (full package)	41000	1	41000
	Total			111500

2.1.2 Regular wage for engineer

Regular wage ($W_{total.wage}$) is the total amount paid to an engineer to complete the project, UAH:

$$W_{total.wage} = F \cdot L_{days} \tag{2.1}$$

Where:

F – Total number of days engineer works in this project;

L_{days} – daily wages of engineer.

$$L_{days} = \frac{W}{N_{days}} = \frac{44000}{22} = 2000,$$

So, we obtain:

$$W_{total.wage} = 44 \cdot 2000 = 88000 UAH.$$

2.1.3 Extra (Bonus) wages for engineer

Bonus wage (W_{Bonus}) is an amount of money bonuses paid for certain qualitative or qualitative achievements, which is 20% of regular wage, UAH:

$$W_{bonus} = \frac{W_{total.wage} \cdot bonus\%}{100\%} = \frac{88000 \cdot 20}{100} = 17600 UAH \qquad (2.2)$$

Where:

Bonus percentage = 20%.

2.2 Social Measures

Social wage ($S_{measures}$) is an amount of money spent on other social activities, UAH;

$$S_{measures} = \left(W_{total.wage} + W_{Bonus}\right) \cdot \frac{S_N}{100} = (88000 + 17600) \cdot \frac{22}{100} \qquad (2.3)$$

$$= 23232\ UAH$$

Where:

$$S_N = Normative\ percentage = 22\%.$$

2.3 Management costs

Administrative expenses (E_d) is the amount of money we spent to buy and repair equipment we used in our projects, which is 15% of regular wage, UAH. It also includes the cost of utilities i.e. electricity, internet, heating etc.;

$$E_{ad} = W_{total.wage} \cdot \left(\frac{\alpha d}{100}\right) = 88000 \cdot \left(\frac{15}{100}\right) = 13200, UAH \qquad (2.4)$$

Where:

αd = Administrative percentage = 15%.

2.4 Depreciation of equipment

When calculating the expenses, it’s necessary to know the values of the annual depreciation from the values of fixed assets of the organization.

$$D = D_b \cdot R_d \qquad (2.5)$$

$$D_{computers} = 60000 \cdot (1/5) = 12000.$$

$$D_{working\ place} = 9000 \cdot (1/5) = 1800.$$

$$D_{software} = 40000 * (1/5) = 8000.$$

$$D = 12000 + 1800 + 8000 = 21800\ UAH$$

Where:

Depreciation rate $R_d = 1/useful\ life$;

Depreciation Base $D_b = W_{total\ cost\ of\ n} - V_{res}$;

The Residual value $V_{res} = 1000$;

$W_{total\ cost\ of\ n}$= Total cost of each fixed asset from the table 2.1.

Since the duration of the project research is 2 months, the value for depreciation will be approximately 3634 UAH.

2.5 Total cost of research

Total cost price of research (C_p) is the total amount of money spent on our research including all wages, UAH;

$$C_p = 500 + 88000 + 17600 + 23232 + 13200 + 3634 = 146166\ UAH \quad (2.6)$$

2.6 Cost profit

Cost profit (P) is a financial gain for engineer, especially the difference between the amount earned and the amount spent in producing.

$$P = C_p \cdot \left(\frac{profit\%}{100}\right) = 146166 \cdot \left(\frac{10}{100}\right) = 14616.6\ UAH \quad (2.7)$$

Where:

Percentage of profit = 10%.

2.7 Wholesale price

Wholesale piece (W_p) is the sum of total cost price and cost profit, UAH;

$$W_p = C_p + P = 146166 + 14616.6 = 160782.6\ UAH \quad (2.8)$$

2.8 Results of calculations

Table 2.2 – Calculation items used to calculate production cost

No	Calculation items	Expenses, UAH	Commentary
1	Regular wage of engineer	88000	$W_{total.wage} = F \cdot L_{days}$
2	Extra wage of engineer	17600	20% of regular wage
3	Social measures	23232	22% of $(W_{total.wage} + W_{Bonus})$
4	Management costs	13200	15% of regular wage
5	Depreciation	3634	Annual depreciation
6	Total cost price	146166	Sum of all above expenses
7	Cost profit	14616.6	10% of total cost price
8	Wholesale price	160782.6	$W_p = C_p + P$
9	VAT	32156.52	20% from wholesale price
	Market price	192939.12	Sum of $(W_p + VAT)$

2.9 Conclusion of economic section

In this project on research and development of the cost required in the design stage of the light aircraft, excluding the manufacturing stage, after a series of calculations, the required cost is 192,939.12 UAH.

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