MINISTRY OF EDUCATION AND SCIENCE OF UKRAINE

National Aerospace University “Kharkiv Aviation Institute”

DEVELOPMENT OF HELICOPTER PILOT PROJECT

Study Guide

Kharkiv “KhAI” 2014

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Розглянуто загальну характеристику проектування вертольотів. Показано доцільність реалізації одногвинтової схеми вертольота (92…95 %), якій в аванпроекті приділено основну увагу.

Запропоновано методику визначення параметрів вертольота за критерієм мінімуму злітної маси та двигуна силової установки за максимумом потрібної потужності найбільш навантаженого режиму польоту.

Наведено визначення параметрів чотирьох вагових категорій вертольотів: R-22, PZL SW-4, Mi-2 і Mi-8. Одержані результати визначають достатню вірогідність проектування за заданою похибкою обчислень.

Подано значний обсяг статистичних матеріалів, що використовують при проектуванні.

Для студентів, які розробляють аванпроект вертольотів за спеціальністю «Літаки та вертольоти».

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The basics of helicopter designing are given. The pilot project reveals the feasibility of helicopter single-rotor configuration (92…95 %) implementation.

Procedure of helicopter parameters determination according to a minimum criterion of takeoff mass and to a maximum of required power of power plant during the most load flight condition is proposed.

Parameters determination of four helicopter mass groups: R-22, PZL Sw-4, Mi-2 and Mi-8 are presented. The obtained results confirm sufficient reliability of designing according to the specified computational error.

The sufficient amount of statistical data used in designing is presented.

For students majoring in "Airplanes and Helicopters", who are taking part in development of the helicopter pilot project.

Figs 161. Tabl. 40. Bibliography: 32 publ.

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You will never become an engineer just by reading textbooks and analysing somebody's constructions. Future engineer is impossible without independent attempts to design something on his own. The greater the number of independent engineering decisions with the advice of the theory, the better engineering skills.

N. I. Kamov

INTRODUCTION

As the aircraft, helicopters are characterized by:

1) more complex flight aerodynamics and dynamics of the main rotor, compared to airplane wings;
2) inability to register all aeroelastic effects on the blades of the main and anti-torque rotors negatively affecting their mass characteristics;
3) multi-mode motion (actually, four-mode motion, longitudinal, transverse, directional, and vertical control);
4) capacity of the engine unit fixed for heavy loaded conditions;
5) multi-part glider;
6) wide nomenclature of materials used;
7) complexity of airframe structural members;
8) complexity of assembly and adjustment and testing works (up to 45% of total complexity of helicopter fabrication);
9) high requirements to the quality of the helicopter and its individual structural members.

The need and duty to fulfil these requirements increases development and production costs of helicopters.

First four features come from the use of blade. Assembly complexity comes from a large variety of permanent link constructions joined by riveting, welding, soldering, pressing, gluing, expansion, etc.

Increasing demand for helicopter reliability has led to the development of special high-resource joints and methods of strengthening helicopters' structural members under complex load conditions.

Today helicopter manufacturing means extensive cooperation, frequent changes of production objects, and a lot of preproduction.

Technological and organizational preparations for serial production are conducted with series-parallel method, i.e. simultaneously. With some time shift, they perform technological refinement of drawings, design process operation, engineer, manufacture, and develop equipment. This helps to significantly reduce preparation time for batch manufacturing of a new helicopter.

Helicopter production depends on the complexity of the construction, its elements and components. Future engineers should remember that the main and anti-torque rotors, skid landing gear, transmission, and partial operating system (especially helicopter components such as fuselage, wheel landing gear, wing, if any, fins, power plant, etc.) are similar to airplane design but have certain helicopter specificity.

This study guide describes basic design tasks for engineering helicopter components and systems, its load conditions.

Helicopter is an aircraft that uses aerodynamic way of flight in which lifting and propulsion forces, control forces, and moments are created by the main and anti-torque rotors.

Helicopter design process is iterative. Its reliability is evaluated by given error of determining the parameters of the design object $\xi = \frac{\left\lfloor (m_0)_j - (m_i)_{j-1} \right\rfloor}{(m_0)_{j-1}}$, where ξ is given accuracy of calculating the take-off weight.

Development of helicopter pilot project is the final stage of a complex training of bachelors in aviation and astronautics. It includes several tasks: statistical formation of helicopter shape; calculations of aerodynamic and flight characteristics; calculations to ensure the static strength of regular zone components; development of prefabricated units (spars, panels, etc.), load-bearing elements of mechanical linkage to control system, engine (engines) mount systems; technology of helicopter manufacturing; sheet-metal stamping and machining; calculations of economic efficiency; development of safety systems, special units, etc.

Students receive the task in their third year; identify the scheme, options, and general structure of the helicopter according to the assignment and then continue working on this topic in special departments.

Pilot project is carried out according to the procedure of preliminary helicopter design: This means that future engineers must integrate all the knowledge they have in many fields, namely general structure of helicopters, design of power plants and systems, design of assemblers and units, aerohydrodynamics, integrity, technology, materials science, economy, etc. Moreover, they have to introduce all the accumulated experience of foreign and domestic helicopter design into their work as well as study features of their devices, configurations, constructive and power schemes of the units, applied constructional materials, production and operation, identifying tendencies and development prospects.

What we consider most peculiar with helicopters is the diversity of their application (multiple uses) [2]:

Modifications of Helicopters

- – Passenger *;
- – Cargo-passenger;
- – Transport **;
- – Agricultural ***;
- – Aeromedical ****;
- – Search and rescue;
- – Training;
- – Aerobatic;
- – Research *****;
- – Patrolling ******;
- – Reconnaissance **;
- – Fire-fighting **;
- – Antiterrorist, etc.

* basic[1], economy class, vip-class, air taxi, sightseeing;
** civil, special or for special purposes;
*** chemical purposes, operational delivery of agricultural products and agricultural cargo, controlling herds of cattle;
**** sanitary evacuation, ambulance, mobile clinic;
***** flying laboratory;
****** visual instrument to monitor oil and gas pipelines, electricity transmissions, highways, forests, fields, lakes, rivers and seas, areas and facilities for the benefit of governmental structures and departments as well as private companies and enterprises.

Certainly, different applications usually need different helicopter equipment.

To determine a helicopter's statistical shape, students must:

- – collect and process statistical data;
- – develop tactical technical requirements;
- – select and justify the helicopter scheme;
- – determine take-off weight of the helicopter in the zero approximation;
- – calculate the construction mass of main units, power plant, fuel, equipment, and management;
- – choose the engine and its characteristics;
- – determine geometric measurements of basic units (main rotor, fuselage, fins, landing gear);
- – find the centre of mass;

[1] The base variant of the helicopter equipped with typical, standard set of flight navigation control and communication equipment for flight under visual flight rules. Under the request of the customer for a helicopter additional equipment for making specific tasks can be installed.

– calculate appearance;

– select, justify, develop, and coordinate load-bearing structures of helicopter units;

– draw the necessary conclusions.

After this, students must use KOMPAS-GRAPHIC, ADEM or AutoCAD to perform drawings, usually

– general view of the helicopter (size A1), and

– load-bearing structure of the helicopter (size A1).

1. CONCEPT DESCRIPTION OF HELICOPTER DESIGN

Helicopter design is a scientific and technological process to make it possess desired performance, operating, economic, production, and technological characteristics. Helicopter design aims at the correct selection of its configuration and parameters; development of structural and loading layout drawings; determination of mass-stiffness and geometry features of its components and units providing desired performance. Therefore, with the allowance made for physical and technical limits as well as for time constraints, helicopter design turns into an iterative process of meeting halfway between different requirements needed for optimal selection of parameters.

Helicopter design should meet the total set of requirements specifying purpose, dimension, and helicopter type; its flight data; structural, operation, and economic characteristics. Al of them are incorporated into requirements specifications for helicopter design and form together with *Airworthiness requirements*, *Aviation rules and regulations*, state and Industry standards main normative documents which specify a process of helicopter development.

When trained in helicopter design, students receive their design assignment in the form of initial data; process statistical data; and then develop performance requirements. A well-grounded development of performance requirements is the first step to providing successful design results.

Development of performance requirements is a really advanced matter because of contradictory measures required to meet different and rather numerous requirements qualified to copter and the need to overcome this contradiction in the most successful way.

The second serious design problem is a flight and technical data ratio of the helicopter under design. Flight data for helicopter under design are specified with an allowance made for these data progress dynamics within some years in the future.

These problems can be solved due to the statistical data analysis of helicopters similar to the one under design. Tabulation of different statistical data for helicopters is not such a simple problem as students can imagine. What is extremely important here is a reliability of the statistical materials used. Unreliable data results in many design errors.

Helicopter configuration is selected judging from the performance requirements and current level of aircraft engineering development. Optimal helicopter parameters and copter efficiency specified by efficiency estimation criterion plays a significant role in the configuration selection. At the initial design phase, this criterion allows helicopter designers to specify the most feasible combination of helicopter parameters and characteristics on the basis of mass balance of its weight components and units.

1.1. Helicopter as the Object of Design

Helicopter is an aircraft with the aerodynamic principle of flight applied. They consume the fuel-provided energy to create lift and propulsive forces, control forces and moments using the air. This type of flying objects has high potential for further development as it can meet constantly growing economical and other requirements. Aerodynamic principle of flight can be realized within the developed velocities and flight altitudes because it is too small.

As a design object, modern helicopters are advanced technical and engineering systems with the developed hierarchical structure, a great number of components and internal connections increasing approximately in proportion to components square.

Helicopters consist of a number of functional subsystems determining their fruitful properties. Every such subsystem incorporates a set of simple and complex systems as well as their components. Even though we divide helicopters into subsystems for different study and analysis purposes, this doesn’t mean that they are fully independent. Helicopter systems are interrelated and interdependent.

On the other hand, helicopter is a technical structure incapable to act beyond production. It is capable of solving specified problems as a part of a more complex system involving helicopters and flight crews, technical means and pre-flight maintenance staff and flight personnel. So, a helicopter is a subsystem of a complex system at the higher hierarchical level—an integrated system to be understood as an organic combination of labour and material resources and operations resulting in development of certain effect. As an integrated system, aviation, in its turn, is a part of the transport system. Approximate diagram of the transport system organization with helicopters used to ship large dimensional cargo is illustrated in Fig. 1.1 [3].

Specific character of a helicopter as a transport means dictates the necessity of wide application of the computer-aided design for its development based on the system design principle. Under system approach to the design, helicopter and its units are shaped on the basis of complex experiment-calculated method with the allowances made for many factors including

– purpose, versions, and conditions of helicopter application and operation;

– unit location in the integrated system;

– limitations and restrains related to the design and engineering conditions, structural materials;

– aeroelasticity, strength, operating and service life, mass requirements.

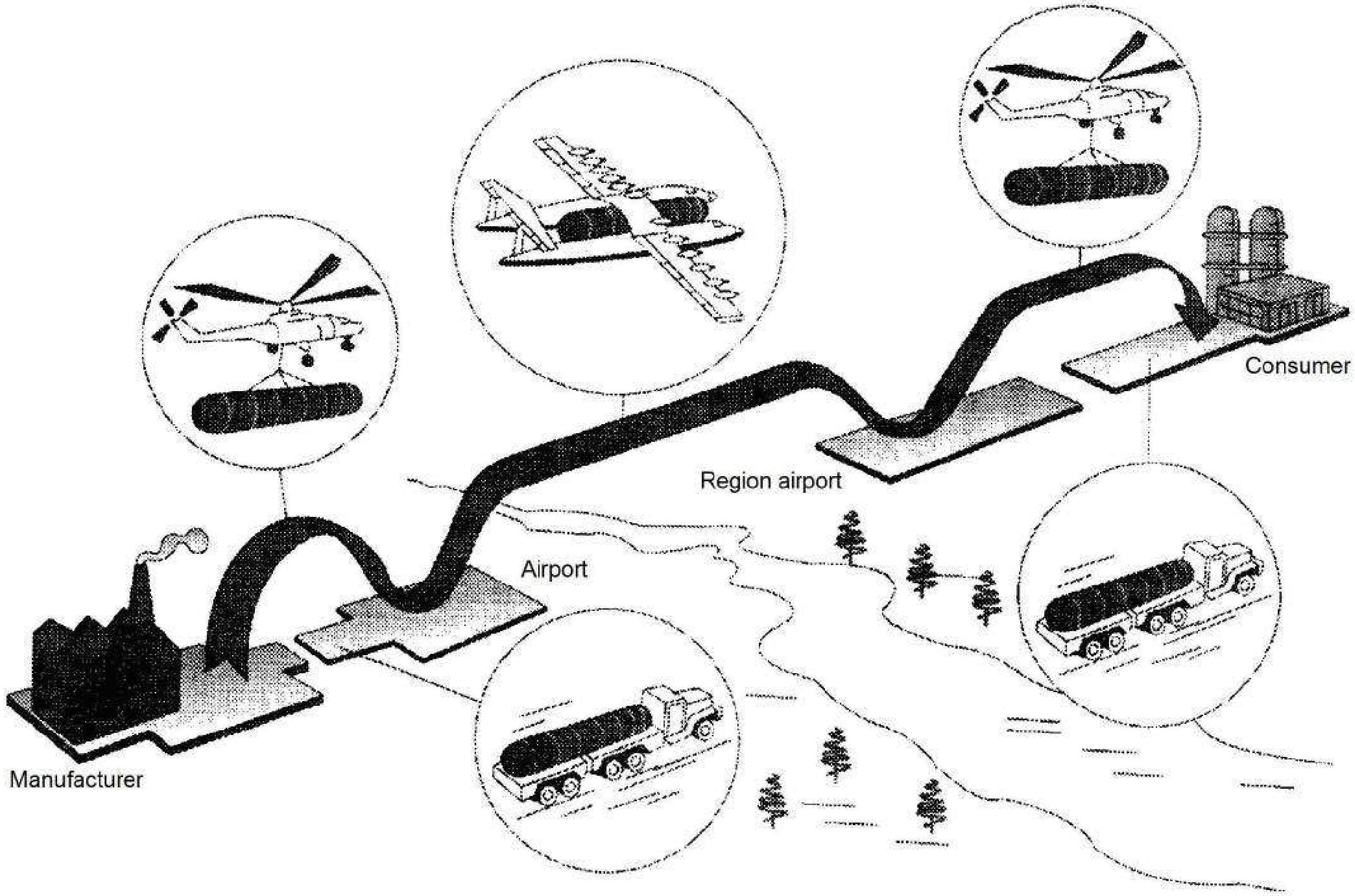


Fig. 1.1. Transport System for Shipping Large Dimensional Cargoes

At the initial stage of helicopter design, its designer develops the conception; determines what problems they will encounter and assumes which procedures and methods will help in solving them (s) (including standard flight profile); ascertains operating conditions $\vec{Z}$; find out efficiency evaluation criteria and parameter optimization $\vec{\Phi}(\vec{S},\vec{Z},\vec{p})$ where $\vec{p}$ is the vector of helicopter parameters and its components (technical aspect) [4]. The models of the unit appearance p and its performance $\vec{S},\vec{Z}$ are developed within the accepted conception connecting spaces $\vec{S},\vec{Z},\vec{p}$ and their representations in the space of the partial criteria (Fig. 1.2) [4].

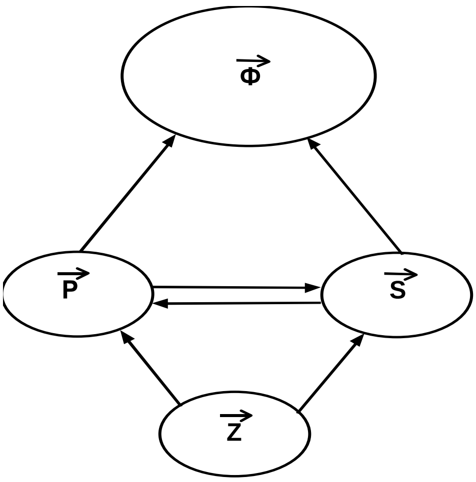


Fig. 1.2. Interconnection of parameters $\vec{S},\vec{Z},\vec{p}$ and their representations in the space of the partial criteria $\vec{\Phi}$

Target functions $W(\vec{\Phi})$, e.g. fuel efficiency (for transport helicopters), reduced capacity (for flying cranes), etc. can be used as criteria when estimating helicopter efficiency and parameters optimization.

The model of the helicopter technical nature enables the designers to solve the problem of the helicopter appearance. The problem can be defined as a search for the scope of its parameters $\vec{p}$ (within the range of specified limitations), providing attainment of the values $W(\vec{\Phi})$ of the selected target function close to optimal under specified operating conditions $\vec{Z}$ at the selected totality of typical problems $\vec{S}$.

Helicopter operation model makes it probable to optimize the totality of typical problems (flight profiles and application fields) as well as specified operating conditions $\vec{Z}$ for the views of helicopter under analyses.

So, helicopter design is a non-linear multi parameter problem. It is based on the iteration procedure requiring fruitful interaction between different institutions. An iteration procedure of structure W is favourable and feasible to be presented in such a way that the totality of helicopter technical advances (its systems and units) could be clearly defined as well as the estimation criteria of helicopter technical aspects corresponding to possible problems and their solutions.

When used, mathematical modelling helps the designers to find a minimum set k of independent efficiency partial criteria $\vec{\Phi}(\vec{\Phi}_1,\vec{\Phi}_2,\ldots,\vec{\Phi}_k)$. In case that the mathematical problem has adequate definition, the totality of these partial criteria completely determines the target function $\boldsymbol{W(\vec{\Phi})}$ and the domain of problem existence in space $\vec{\boldsymbol{\Phi}}$ (Fig. 1.3).

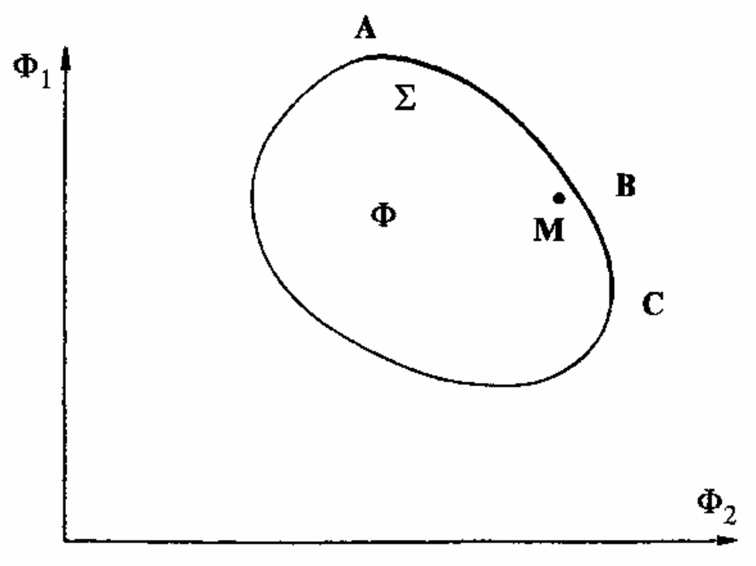


Fig. 1.3. Domain of Problem Existence in Space $\vec{\boldsymbol{\Phi}}$

Generally, the development of helicopter complexes (HC) covers three main stages of research, namely conception, appearance, and design.

Conceptual stage provides time and space to specify the main idea of a new helicopter, determine its performance and prove the compliance with existing norms.

Appearance stage involves developing the helicopter technical view within the conception adopted; selecting the configuration; finding the most feasible combination of the helicopter main flight data and its systems so as to meet technical and operation requirements as well as secure application efficiency. During the second stage, designers often work on several versions of the helicopter appearance which parameters best meet different technical and operation requirements. At last, they choose the version with the best value of the accepted criterion. Minimal mass of the helicopter or its cost [5] are the most frequent criteria.

Design stage includes clarification of separate technical and operating characteristics, development and adoption of technical conceptions and their efficiency estimation.

In this tutorial, we conventionally understand a helicopter complex as a cycle of researches undertaken in order to determine the conception of the design object, to figure out basic tasks to be fulfilled by this complex, to provide optimal arrangement of the systems, subsystems, and helicopter components to **optimize** the performance and technical view with allowances made for technical, production, and economical probabilities determining prospects of helicopter engineering development.

Despite significant differences between helicopters and other types of aircraft, their design and production meet general outlines of aeronautical engineering development and include

– development of technical and operation requirements;
– development of technical offers (pre-designing pilot project);
– draft designing and prototyping;
– detailed designing, prototype manufacturing, and its flight tests.

1.2. Brief Outlook of Helicopter Design Procedures

In the history of engineering, few examples can be found to expose so much time taken between the technical idea made its appearance and its practical realization as it took place in the development of the rotorcraft. Helicopter manufacturing history is characterized by many ups and downs of interest to helicopters as rotorcraft dictated by raising problems, contradictions, successes or lack of success.

Aircraft and rotorcraft engineering development includes the development of its design procedures.

Still in Middle Ages, Chinese people knew that rotating rotors can create lift. In 1475 Leonardo da Vinci schematically presented an aircraft capable of “screwing” into the air by the use of the Archimedes propeller. The very first “aerodynamic” model of helicopter was made by Mikhail Lomonosov in 1754. It had two coaxial propellers rotated by the clock ring. Balanced by counterweight through the unit, this model was able to lift due to rotating propellers.

Developing, helicopter design procedures become more and more complicated because more complex problems arose which could not be easily solved. From trial-and-error method and practical construction, helicopter design has arrived at multistage multiple-iteration process of the scientific and research involving wide application of mathematical techniques, complicated technical means, and high level of generalization.

Development of helicopter design procedures is impossible without strong interrelation between helicopter design and science. Helicopter manufacturing is one of the most scientifically profound branches of engineering. Available experience in helicopter design shows that as soon as their specific parameters grow, the scope of estimation and experimentation increases drastically.

With the increase in load ratio to be higher than 40%, the preliminary estimating researches and developments become the decisive factor. Should new design conceptions be used, a scope of researches intended to provide high operability of the new structure increases as well.

The following are main directions in research specific for the design of a new helicopter [7]:

– selection of optimal helicopter parameters (development of a great number of variants followed by varying the most significant parameters of helicopter and its main units);

– calculations associated with the provision of strength properties and structure operability as well as safety from different types of auto oscillations;

– search for new ways to reduce loads and stresses as well as to increase dynamic tension and endurance limits of parts and components made of traditional and new materials;

– optimization of parameters (calculation and estimation results frequently interfere with the determination of helicopter parameters which, in their turn, influence certain characteristics of helicopter units).

If we analyse helicopter design procedures, we shall see that two groups of methods are basically used, direct and indirect methods with the latter involving statistical as well as trial-and-error methods.

The range of direct methods is certainly wide and covers automated and system design procedures based on analytical design.

Analytical methods lean upon simultaneous solution of equations system with an account made for the most significant relations among helicopter parameters and characteristics as well as different limitations. Application of analytical methods has opened wide horizons for parametric research aimed to find out what happens if the design parameters are changed as well as limitations of helicopter's technical and economical characteristics. This was the beginning of optimal helicopter design. Analytical methods made it possible to positively influence the estimation design criteria accounting for multipurpose helicopter application and to improve dynamics of helicopter operation.

Helicopter design benefited significantly from the development of the general design theory for large systems as it has provided definitely scientific approach to the prediction of the future helicopter parameters and characteristics making helicopter design logically complete. Intentionally employed, system design allowed for dividing design process into stages as well as helicopter into subsystems and units. Modern system design reveals itself as strictly ordered complex approach which pays great attention to complicated associations and interrelation of the system components (and involves optimal design method as a constituent part). This approach allows the designers to go from optimizing separate facilities and characteristics to considering helicopter a complex system and finding the system criteria of optimality with the help of mathematical models and wide application of computers.

Automated design systems (ADSs or CADSs) are designed and developed in the scientific research institutes and design offices. Their application provides higher quality of the design, cuts the design cycle, and increases labour intensity thus accelerating scientific and technical progress.

Analytical design methods and procedures are based on relations among helicopter parameters and characteristics presented in the form of analytical formulae.

Finite algebraic and transcendental equations as well as usual differential equations and equations with partial derivatives are usually used for the mathematical description of a helicopter. Algebraic and transcendental equation describe geometry mass, aerodynamic and partially power relations, and ratios between helicopter parameters and characteristics. General differential equations describe trajectory parameters and dynamic characteristics. Differential equations with partial derivatives describe dynamic characteristics and some strength properties.

After mathematical modelling, the task of system design can be reduced to the problem of finding the global function extrema of many parameters in the form of functional equality and inequality (with certain limitations for these parameters).

Modern computer-aided design is the advanced direct iteration method, analytical method of optimal design.

1.3. General and Specific Requirements to Helicopter

What is the most important about helicopter design, it must requirements specifying its purpose, dimensions, type, flight data, structural, operating, economical characteristics, etc. All these requirements specified in technical assignment for the helicopter design. Together with *USSR civil helicopter airworthiness standards* (НЛГВ), *Aviation regulations* (АП), state (ГОСТ) and industry standards (ОСТ) and the like, they specify the process of helicopter development.

All the requirements put forward to helicopters under design can be arbitrarily divided into two groups:

– general requirements specifying the helicopter’s level of technical complexity;

– specific requirements enabling the helicopter design to fulfill its tasks.

General requirements are detailed in airworthiness standards [8] and aviation regulations [9].

Helicopter airworthiness standards are an accumulation of state requirements to civil aircraft airworthiness focused on flight safety. Airworthiness is determined by the aircraft capability to conduct a safe flight within the total range under specified operation conditions (with certain allowances made for normally operating components of the aviation transport system, ATS).

Requirements for helicopter as well as airplane airworthiness are based on the probability of hazardous, emergency, and fatal situations under failure of different units and airborne systems.

Assumption is made in airworthiness standards for civil helicopters that probability of emergency (e. s.) and fatal (f. s.) situations shall not exceed, respectively:

– single system failure, $P_{e.s} \le 10^{-6}$, $P_{f.s} \le 10^{-8}$;

– probable combination of system failures, $P_{e.s} \le 10^{-5}$, $P_{f.s} \le 10^{-6}$.

On this assumption, the helicopter flight, take-off, and landing performance requirements are established for functional system failures under probable changes of performance during operation period.

The first edition of *Civil aviation helicopter airworthiness standards* for helicopters with a take-off weight no less than 10000 kg was published in 1971. It included a number of requirements for helicopters'

– flight safety;
– performance, stability, and controllability;
– structural strength;
– structural components, systems, and units;
– engine and transmission;
– power plant and fire protection systems;
– equipment (including engine equipment), etc.

Airworthiness requirements must be met when designing, producing, testing and certifying, giving permissions for operation and maintenance, exporting and importing civil aircraft and rotorcraft as well as when developing state and industry standards, specifications, and assignments. Aviation registers are obliged to control how airworthiness standards are held. Some deviation from airworthiness requirements is tolerable provided that they are compensated by other measures ensuring the same safety level.

There are international and national (state) airworthiness standards. International standards and recommendations are developed by ICAO and were first published as the Appendix to Chicago convention. It contained a wide range of airworthiness standards to be available for every country-member of ICAO in order to be used as a basis for the development of national airworthiness standards.

Countries-members of ICAO have their own national airworthiness standards, or they use airworthiness standards provided by the countries with more advanced civil aviation engineering. Federal Aviation Regular (FAR) is a foreign airworthiness standard the most widely used by ICAO members.

The second edition of *USSR civil helicopter airworthiness standards* came into force in 1987. The comparative analysis of these standards and Part 29 of *USA Federal Aviation Regulations* (FAR-29) shows that safety levels established are the same although there are certain differences in structure and contents as far as requirements for standardized characteristics are concerned.

Transport Category Rotorcraft Airworthiness Standards [49] form an integral part of *Aviation Regulations* (AR 29). Per their structure and content, these regulations conform with the corresponding corrections to FAR-29 (in the form of paragraphs, items, supplementary sections).

AR 29 includes the following sections:

A – General: applicability; special requirements having reverse force.

B – Flight: performance, flight data, controllability characteristics on ground and water, different flight requirements.

C – Strength requirements: flight loads, control surface and control system loads, ground loads, water loads, requirements to main structural components, emergency landing conditions, fatigue strength estimation.

D – Structure and manufacturing; rotors, control systems, landing gears, raft floats and bodies, cargo and passenger arrangement, fire protection, cargo tie-down fittings, etc.

E – Power plant, rotor actuator systems, fuel system, fuel system units and components, oil system, cooling system, air bleed system, exhaust system, power plant units and controls, power plant fire protection system.

F – Equipment, instruments, and arrangement; electrical systems and equipment; lighting; safety equipment; different purpose equipment.

G – Additional requirements to equipment airworthiness.

I – Operating limitations and information: operating limitations, marking, and placards; rotorcraft flight manual.

K – Special aviation requirement: rotorcraft limited operation according to instrument flight rules.

Specific requirements for helicopters under design specify their target purpose, performance, operating, technical, and economical characteristics, i.e. tactical-technical requirements. Tactical-technical requirements (TTR) specifies all important characteristics of a future helicopter such as its purpose, lift capacity, dimensions of cargo compartment, number and type of engines, range or time in flight, endurance, crew size as well as performance data (static and service ceilings, cruising and maximum speed, rate of climb, time to climb, range). Operation requirements are also specified including estimated life of main components (general and life between overhauls), maintenance man-hours, and intervals between scheduled maintenance checks. Moreover, TTR supplements specified requirements to helicopter operating conditions, specific flight conditions, equipment components, etc. depending on the actual helicopter's purpose and class. So, a helicopter under design is mainly (excluding design configuration) and fully determined by the tactical-technical requirements which must be included in its technical assignment (T3) as in the main design document.

Successful design is probable if the designers clearly understand what the helicopter is intended for and what parameters and characteristics it should have. Customer`s tactical-technical requirements usually give the answer to these questions. In training future designers, TTR development is based on the processing and analysis of design and statistical data.

Precise requirements depend on the purpose of helicopter and its operating conditions and may have different nature but they always aim at the highest level of technical and economical efficiency as compared to the existing copters of the same class and category. At the same time, helicopter designers should remember that excessively high requirements result in the deterioration of both mass and weight properties and other characteristics and, as a rule, reduce helicopter efficiency. For example, an increase in helicopter load-to-takeoff weight ratio is the most significant means to increase hovering ceiling and rated capacity of a transport helicopter. But to put up the hovering ceiling to 3000 m under standard conditions, you need to increase the engine design power in 1.5 times as compares with the ceiling equal to zero. In this case, high throttling back of the engine at the cruising speed results in 10%-growth of specific fuel consumption (under flight altitude of 500 m) and requires more powerful engines which (when installed) would result in greater weight of the main mechanical units and, thus, in the smaller range [10].

When developing TTR, helicopter designers should proceed from the tendency to use a helicopter for special purposes and make the allowances for the requirements specified in state and industry standards.

Tactical-technical requirements should state and prove [10]:

– purpose, main and auxiliary problems to be solved by the helicopter;

– flight performance, technical parameters and characteristics;

– operation and application conditions;

– design load (list if cargo, weight, dimension), cargo and passenger cabin, doors and hatches dimension;

– crew size, etc.

In order to bring TTR developed by future designers as close to real ones as possible, they must include the section given below.

Purpose. The main purpose of a helicopter is specified in the design documents. The designer should prove the helicopter feasibility; develop other probable variants of its application; specify its operating conditions.

Flight performance is specified by these parameters:

1. Range (time in flight) is determined by the purpose and application of the helicopter under design. Unreasonably high range requirements may result in helicopter overweight and deterioration of its manoeuvrability. To calculate a required volume of the full tanks under extended flight, it is necessary to specify ferry or maximum range under correspondingly reduced load. In some cases, time in flight in certain mode can be specified instead of the range.

2. Helicopter hovering and service ceiling altitudes are specified based on geographic characteristics of the regions where the helicopter will operate. Do not accept unreasonably high altitude values! If the helicopter is expected to operate in high ground terrain, a hovering ceiling altitude can be specified with an allowance made for the ground proximity effect.

3. Maximum flying speed is specified with an allowance made for the statistical data on helicopters of the assumed configuration. Future designers should remember that a maximum speed of modern helicopters is in most cases dictated not by the rated power plant capacity but by aerodynamic limitations of the lifting rotors. For this reason, you should use special lifting rotors or other configuration concepts (helicopters with wings, helicopters with extendable landing gear, combined helicopters, etc.) to get high flying speeds.

4. Rated flight altitudes.

Stability, controllability, and manoeuvrability. The purpose of helicopter dictates the requirements for stability, controllability, and manoeuvrability and may considerably influence its configuration and equipment selection as well as control system and lifting rotor design.

Flight safety. Safety of flight is the main and undeniable requirement to any aircraft. It is provided by equipping the helicopter with two or more engines and specifying its altitude range. Altitudes are determined with an allowance made for the relief of ground surface (terrain) and must be higher than altitudes of probable landing fields in basic operation regions. TTR should also prove:

– minimal tolerable speed of vertical takeoff;

– maximal tolerable speed of vertical landing for the helicopter descending under autorotation condition.

Equipment. Besides the essential set of power plant, flight navigation instruments, and transmission monitors, helicopters are provided with a great variety of avionics and special equipment. It is desirable that TTR lists all this equipment with its weight, dimension, and installation place specified.

Structural and operating maintainability requirements specify:

– structural-load bearing diagrams and materials used for main structural components;

– view from the pilot compartment as well as arrangement of cargo compartment and crew life-support and survival system;

– dimensions of certain helicopter components ensuring their transportation by ground vehicles or other types of transport.

Helicopter designers should pay due attention to operational maintainability requirements. TTR should specify:

– necessary level of structure adaptation in order to provide maintenance and overhaul under operating conditions: easy access to the units and components, easy removal and interchangeability, computer monitoring, etc.;

– requirements for cargo compartment or use of external tie-down load;

– requirements for passenger compartments dictated by the corresponding sanitary norms [9, 10].

Strength and service life. Requirements to static and dynamic structural strength must meet strength standards for helicopters. When putting forward the service life requirements, helicopter designers should proceed from the achieved level with an allowance made for structural and process peculiar features meeting specific requirements.

Crew size. Requirements to the crew size must be specified based on proper analysis of the problems to be solved and taking into account application and operation of similar purpose helicopters.

1.4. Helicopters Basic Configurations and Their Efficiency Proof

Helicopters are mainly classified by their counteract torque reaction (Fig. 1.4) to include such configurations as single rotor with tail rotor or two rotor (coaxial, longitudinal, and lateral) (Fig. 1.5).

Helicopters of a single rotor configuration (Fig. 1.6) are the ones most widely (around 92 %) used and, thus, most frequently manufactured. Such configuration monotony can be mainly explained by the fact that at the modern stage of aviation, single rotor configuration has more advantages and less disadvantages as compared to other helicopter configurations. Coaxial and longitudinal configurations are less used (Figs 1.7, 1.8).

1.4.1. Helicopter Main Configuration

Helicopter configuration is dictated by the purpose of the rotorcraft, its application efficiency, certain tactical-technical requirements and their contradictions as well as continuous growth of probability to meet these requirements. Helicopter configuration and its parameters must conform to a certain criterion of efficiency.

If we compare statistical data on light and medium helicopters of single rotor and two rotor coaxial configurations, we shall find out that they don’t have significant difference, and the best configuration cannot be easily chosen [5]. But this is not true for helicopters of longitudinal configuration because they provide greater takeoff mass and require more powerful power plants.

Under these conditions and well aware of the fact that there are no heavy helicopters with coaxial configuration, we can assume that single-rotor configuration can be feasible even for helicopters of great dimension. It is proved by the serial Mi-26 helicopter with the worldwide highest lifting and load-carrying capacity.

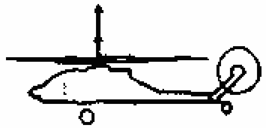
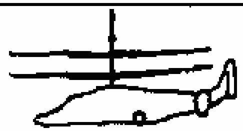
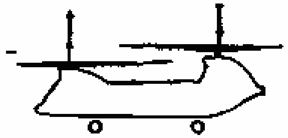
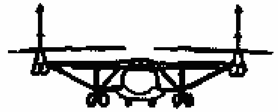
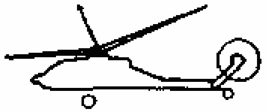
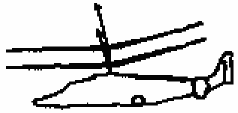
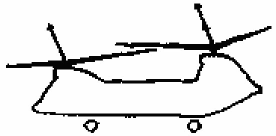
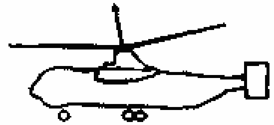
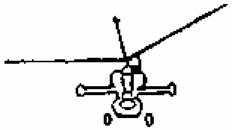
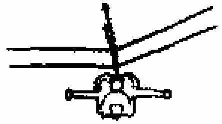
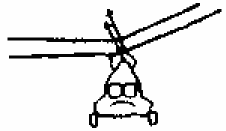
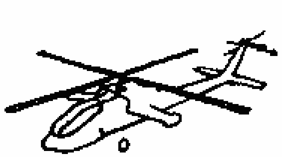
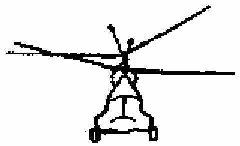
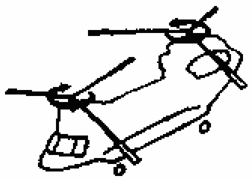
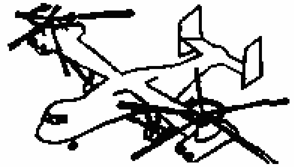
Control and balancing type	Helicopter Configuration			
	Single rotor with tail rotor	Two rotor coaxial	Two rotor longitudinal	Two rotor lateral
Altitude control				
Longitudinal control				
Lateral control				
Directional control				
Counteract torque reaction principle				

Fig. 1.4. Helicopter Control and Balancing Principle

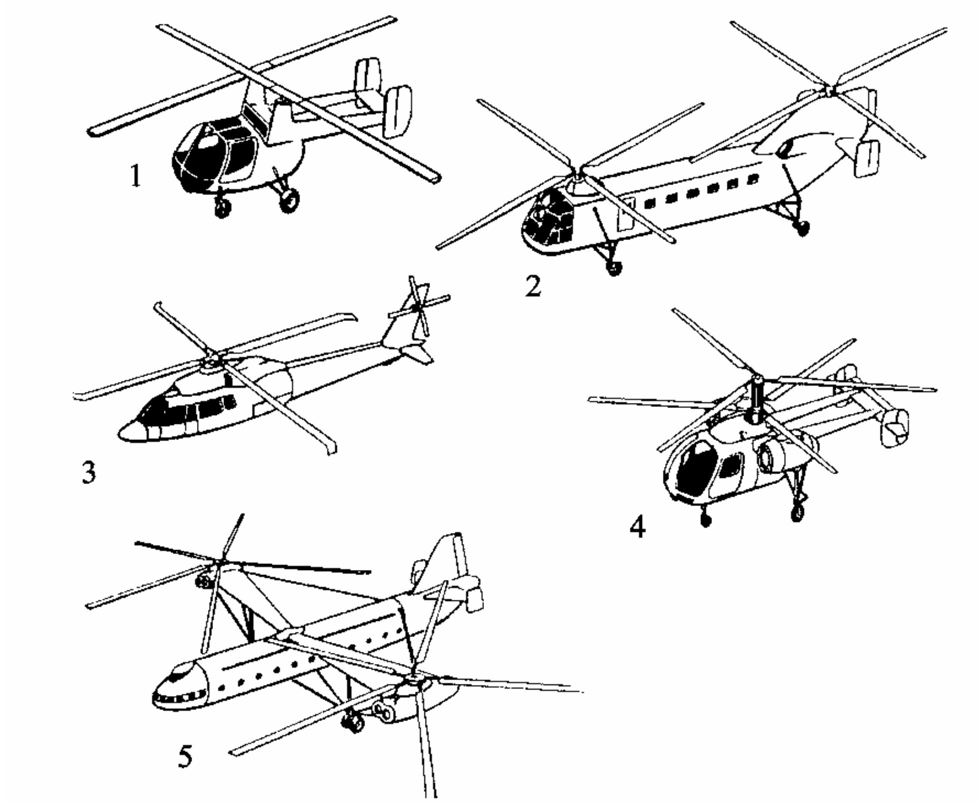


Fig. 1.5. Main Configurations of Helicopter:
1 – with crossing axis of rotors; 2 – longitudinal; 3 – single rotor; 4 – coaxial; 5 – lateral

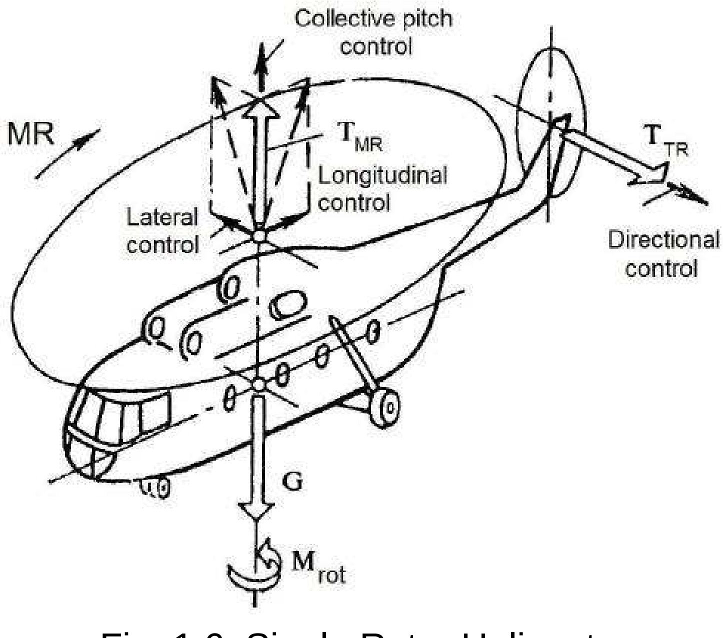


Fig. 1.6. Single Rotor Helicopter:
$T_{m.r.}$ – rotor thrust; $T_{t.r.}$ – tail rotor thrust; M_{torq} – main rotor torque;
G – helicopter weight

Fig. 1.7 shows coaxial helicopter configuration and acting forces. Longitudinal helicopter configuration and acting forces are given in Fig. 1.8.

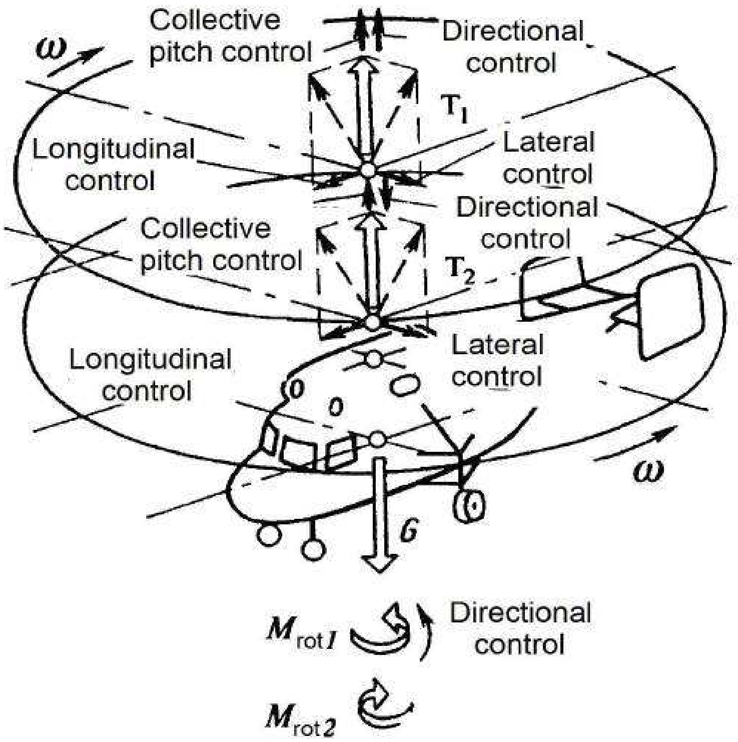


Fig. 1.7. Two Rotor Coaxial Configuration Helicopter:

$\mathbf{T}_{\mathbf{1}}$ – upper rotor thrust; $\mathbf{T}_{\mathbf{2}}$ – lower rotor thrust; $M_{torq.1}$ – upper rotor torque; $M_{torq.2}$ – lower rotor torque; **G** – helicopter weight

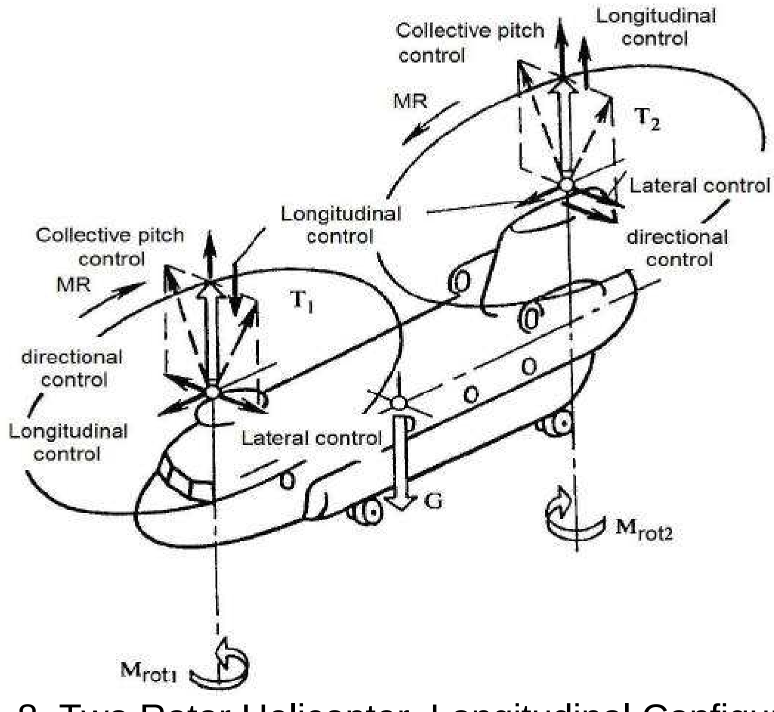


Fig. 1.8. Two Rotor Helicopter. Longitudinal Configuration:

$\mathbf{T}_{\mathbf{1}}$ — forward rotor thrust; $\mathbf{T}_{\mathbf{2}}$ — aft rotor thrust; **G** — weight; $M_{torq.1}$ — forward rotor torque; $M_{torq.2}$ — aft rotor torque

Basic advantages and disadvantages of most widely used helicopter configurations are given in Table 1.1.

Table 1.1

Advantages and Disadvantages of Helicopter Different Configurations

Advantages	Disadvantages
Single rotor-configuration with a tail rotor	
Simple design Simple control system Relatively low cost	Power loss on tail rotor Narrow range of probable centre-of-gravity Long tail boom increasing helicopter dimensions and hazard of contact under operation
Two rotor coaxial configuration	
Relatively low drag due to short fuselage of nice shape Small dimensions	Complicated control and transmission system Harmful interaction of main rotors Hazard of two rotors contact
Two rotor longitudinal configuration	
Perfect longitudinal stability Wide range of probable centre-of-gravity Perfect load-to-takeoff weight ratio during flights at relatively low distances High fuselage useful capacity	Complicated transmission and control system High inductive losses when flying at horizontal speeds Call for synchronized rotation of main rotors Complicated glide landing on self-rotating rotors

So, the problem of selecting feasible helicopter configuration should be solved by comparing different rotorcraft configurations and reasoning from its main purpose, specified TTR, and minimal takeoff weight with an allowance made for accepted efficiency criteria. Another important factor is capabilities provided and traditions followed by precise helicopter design offices.

Generally, comparative analysis of different rotorcraft projects should be conducted with an account made for such characteristics [11] as:

1) correspondence of the flight data obtained to the specified ones;
2) structural weight and value of the cargo transported;
3) desired strength and reliability of the structure;
4) rotorcraft stability and controllability;
5) flight safety;
6) human hours need to conduct project design;
7) complexity of problems to be solved;

8) risk to be unsuccessful in assignment completion because of potential contingencies;

9) extent of sophistication and manufacturing costs;

10) operational data;

11) rotorcraft operating economy.

1.4.2. Configuration Efficiency Proof

There are many ways in which helicopter designers solve the problems occurring during their search for optimal design. Efficiency usually becomes a measure of optimal design used to assess whether desired results were obtain in the best possible way. The reason for this is that helicopters have so many parameters and characteristics and meet so many different requirements that it is nearly impossible to bring them all to one and the only estimation criterion. Incorrect criterion selection results in unnecessary waste of design efforts and resources.

Traditional elaboration approach to the rotorcraft efficiency criterion lies in comparing fruitful effects of the copter employment and costs needed for their implementation.

In a general way, estimation criteria must meet these basic requirements:

a) be measurable (computable) in a familiar way;

b) provide allowances for main purposes and goals of the object development as well as operating conditions and limitations;

c) involve those parameters and characteristics of the object which must be estimated or optimized;

d) be non-contradictory at any stage of design.

It is desirable that there is only one estimation criterion at every stage of design though this is often a pure wish as it is difficult to give preference to a single criterion and there is a need to know solutions as far as several probable criteria are concerned.

Modern helicopter design usually employs three fundamental factors to estimate helicopter efficiency [12], namely:

– rotorcraft perfection characterized by the mass ratio of its components;

– flight capacity defined by the scope of useful work performed by a rotorcraft during a time unit;

– rotorcraft fuel efficiency.

Helicopter perfection estimated through a load-to-takeoff weight ratio is one of the most significant specific parameters able to reveal helicopter's structural properties. E.g., an increase in load-to-takeoff weight ratio is the most essential means to increase hovering ceiling and reduce transport capacity.

But the available experience shows that an increase in load-to-takeoff weight ratio is rather delicate and risky thing. By simply reducing the weight of a unit or component, we may reduce strength and thus provoke significant modification of the final object, fail to secure the required flight performance, and, as a rule, go far beyond set development terms. For reasonable reduction of the helicopter structural weight it is necessary to know how parameters and loads produce effect on the weight of its components.

There are many essential ways to increase load-to-takeoff weight ratio:

– select reasonable arrangement, layout, helicopter load-carrying diagram and structural components;
– be careful then selecting the most feasible helicopter configuration and components;
– reduce acting variable stresses in helicopter components;
– employ unrevealed excess strength and stability;
– provide mass reserve;
– use new materials;
– enhance endurance limit (fatigue range) of helicopter parts and units;
– upgrade design quality;
– apply efficient promotion systems [12].

The following load-to-takeoff weight ratios [5, 12] are recognized as per:

1) full load

$$\bar{m}_{fl} = m_{fl} / m_0 = \frac{m_0 - m_{empty}}{m_0} = 1 - \bar{m}_{empty}; \quad (1.1)$$

2) payload

$$\bar{m}_{pl} = m_{pl} / m_0; \quad (1.2)$$

3) useful load

$$\bar{m}_{ul} = m_{ul} / m_0 = \frac{m_{pl} + m_{fuel}}{m_0} = \bar{m}_{pl} + \bar{m}_{fuel}; \quad (1.3)$$

4) fuel

$$\bar{m}_{fuel} = m_{fuel} / m_0 . \quad (1.4)$$

All types of load-to-takeoff weight ratio are linked by the relationship

$$\bar{m}_{fl} = \bar{m}_{op} + \bar{m}_{ul} = \bar{m}_{op} + \bar{m}_{pl} + \bar{m}_{fuel}, \quad (1.5)$$

where $\bar{m}_{op}$ is a relative weight of helicopter operational load (items),

$$\bar{m}_{op} = m_{op} / m_0 . \quad (1.6)$$

Tyshchenko et al. [12] put it down that load-to-takeoff weight ratio with reference to a payload is rather a demonstrative characteristic which makes no allowances for different in lift-to-drag ratio and specific fuel consumption of the compared copter engines.

By employing load-to-takeoff weight ratio as per full load in the form of (1.6), we can secure minimal takeoff weight of the helicopter under design. The minimal takeoff mass corresponds to a weight balance equation (1.1) and provides such a combination of the copter components' parameters and characteristics that strictly meets tactical-technical requirements. This is based on the assumption that weight characteristics of the copter components are equivalent to the corresponding parameters, copter and component properties collection.

Flight capacity. General payload capability of a helicopter as a transport means is characterized by the dependence between range $m_{pl} = f(L)$ and payload weight which specifies impartial factors of transport efficiency:

– work performed in flight at specified range

$$A = m_{pl} \cdot g \cdot L; \tag{1.7}$$

– flight capacity

$$\Pi = \frac{A}{t_{flt}} = m_{pl} \cdot g \cdot V_{aver}; \tag{1.8}$$

– specific capacity

$$\overline{\Pi} = \frac{\Pi}{m_0}, \tag{1.9}$$

where ***A, П,*** $\overline{\Pi}$ are expressed respectively in daN·km, daN·km/h, and (daN·km/h)/kg; t_{flt} is total flying time from take-off to landing; V_{av} is average flight speed.

Flight capacity

$$\Pi = m_{pl} \cdot g \cdot K_v \cdot V_{cruis} \tag{1.10}$$

is the most significant component of the total capacity depending on performance and rotorcraft weight values. It specifies its overall transport capabilities in flight hours but an allowance is not made in explicit form for the copter lift-to-drag ratio and fuel efficiency of its engines. In addition, capability indication in the form of (1.10) does not allow us to compare different helicopter versions designed based on one and the same technical assignment due to their close to identical capacity.

In (1.10), $K_v = V_{av} / V_{cruis}$ is the coefficient of time losses caused by manoeuvres necessary before take-off and landing as well as climbing and descend according to GosNII GA data $K_v = f(V_{cruis}, L)$ (Fig. 1.9).

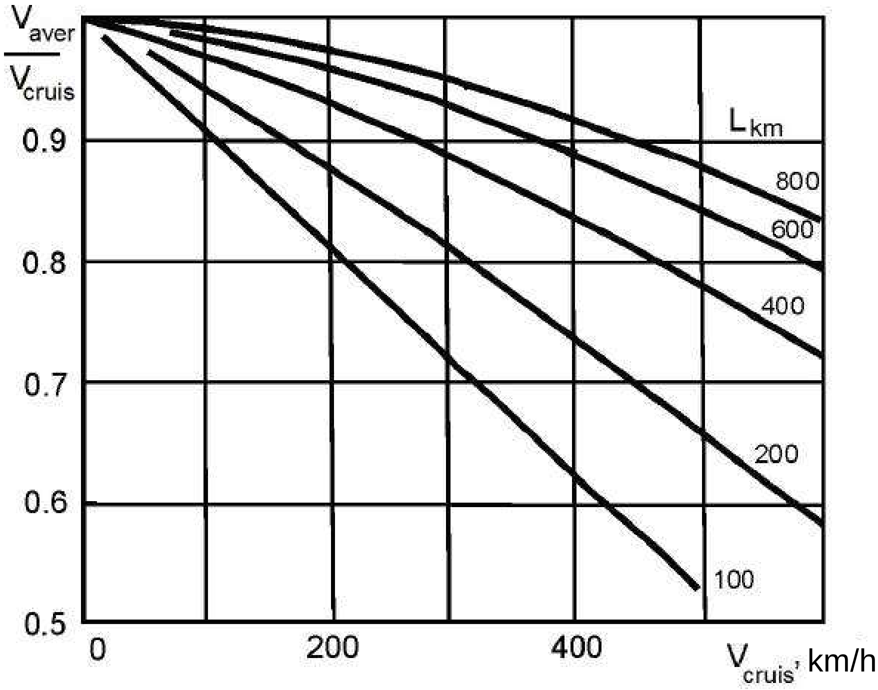


Fig. 1.9. Dependence $K_v = f(V_{cruis}, L)$

Fuel efficiency. Amount of fuel used for a unit of transport work, kg/daN·km is the most significant indicator of fuel efficiency:

$$C_L = \frac{m_{fuel}}{m_{pl} \cdot g \cdot L} = \frac{\bar{q}_k}{m_{pl} \cdot g} = \frac{m_{fuel}}{m_{pl} \cdot g \cdot m_0 \cdot L}, \quad (1.11)$$

where $\bar{q}_k$ is relative fuel consumption per one kilometre of flight at specific range, kg/kg·km:

$$\bar{q}_k = \frac{m_{fuel}}{m_0 \cdot L}. \quad (1.12)$$

$\bar{q}_k$ specifies lift-to-drag ratio, specific fuel consumption, and engine power losses but an allowance is not made for helicopter perfection as per its weight. Flight conditions allowing for minimal C_L are also left undefined.

Fuel efficiency does not account for the transport working hours. Two transport vehicles, e.g., a truck driving along bad roads and a prop plane have approximately identical values of C_L [5] though time required for transport operation is obviously different. So, it is rather attractive to find such an indicator of transport efficiency which will incorporate indicators and values mentioned above.

Helicopter reduced capacity.

Reduced capacity is used as a generalized criterion specifying efficiency of transport and passenger helicopters [5], $\frac{\text{daN} \cdot \text{km/h}}{\text{kg/km}}$:

$$\tilde{\Pi} = \frac{\overline{\Pi}}{\overline{q}_k} = \frac{K_V \cdot \overline{m}_{pl} \cdot g \cdot V_{cruis}}{\overline{m}_{fuel} / L}. \tag{1.13}$$

It combines transport capacity, lift-to-drag ratio, and helicopter load-to-takeoff weight ratio as per payload and fuel consumption efficiency of helicopter power plant.

(1.13) shows what a capacity would be of a conventional helicopter having the same (as the helicopter under design) cruising speed, weight advantages, specific fuel consumption, and lift-to-drag ratio but different takeoff mass. In this case, a takeoff weight of a conventional helicopter must meet 1 kg of consumed fuel per 1 km.

For $\tilde{\Pi}$, all versions compared demonstrate the same kilometric consumption under the influence of takeoff weight change.

To determine the optimal flight range in compliance with the reduced capacity, we should compare different versions of the helicopter under design and select the most feasible one.

Krivtsov et al. [5] provide us with the values of $\tilde{\Pi} \cdot 10^{-3}$ for helicopters under flight range of 300 km, in $\frac{\text{daN} \cdot \text{km/h}}{\text{kg/km}}$: 35 is for MI-4, 167 is for MI-26, and 157 is for MI-8. For helicopters recently developed, $\tilde{\Pi} \cdot 10^{-3}$ is equal to 300…360 $\frac{\text{daN} \cdot \text{km/h}}{\text{kg/km}}$.

Fig. 1.10 shows how reduced capacity influences flight range of MI-6 [22].

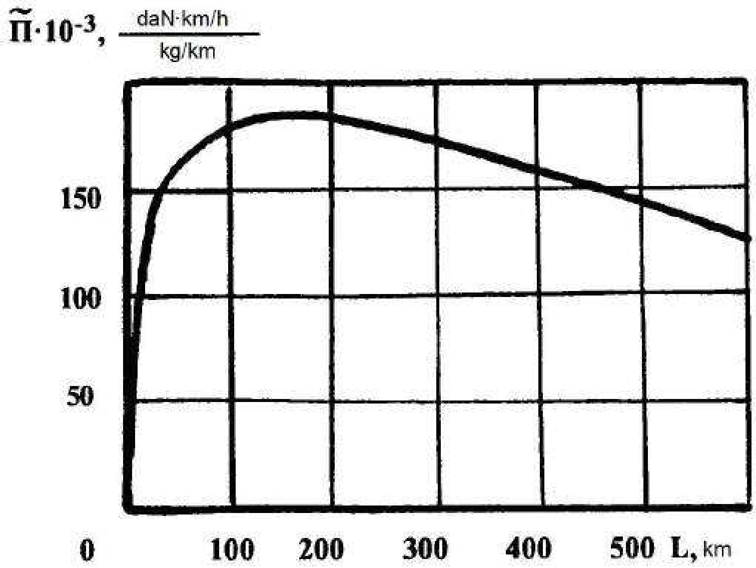


Fig. 1.10. Influence of Flight Range upon
Reduced Capacity $\tilde{\Pi} \cdot 10^{-3}$ for Helicopter MI-6

(1.13) may be represented in the form useful for parametric investigations to estimate the influence of individual parameters varied upon $\tilde{\Pi}$.

Hourly fuel consumption for a helicopter q_h can be expressed via engine specific fuel consumption $\boldsymbol{C_e}$ and engine power $\mathbf{N_e}$: $q_h = C_e \cdot N_e$.

In steady flight, $N_{av} = N_{req} = \eta \cdot \xi \cdot N_{ef}$, where N_{av}, N_{req}, N_{ef} are available, required, and effective power on the engine shaft respectively; ξ, η are power filling factor and efficiency factor of mover.

Then $q_h = \dfrac{C_{ef} \cdot N_{av}}{\xi \cdot \eta} = \dfrac{C_{ef} \cdot N_{req}}{\xi \cdot \eta}$.

Kilometric fuel consumption for a helicopter

$$q_k = \frac{q_h}{V} = \frac{C_{ef\ cruis} \cdot N_п}{K_v \cdot V_{cruis} \cdot \xi \cdot \eta} = \frac{C_{ef} \cdot G_0}{0.27 \cdot K_v \cdot \xi \cdot \eta \cdot K}, \tag{1.14}$$

where $C_{e\,cruis}$ is specific fuel consumption during cruising flight, $\dfrac{kg_{fuel}}{kW \cdot h}$; $\boldsymbol{G_0}$ is takeoff weight, daN·10^3; $\boldsymbol{K}$ is copter lift-to-drag ratio.

Starting from helicopter specific capacity and relative kilometric fuel consumption, the reduced capacity criterion for a parametric analysis can be written in the following way:

$$\tilde{\Pi} = 0.27 \cdot K_v^2 \cdot \frac{m_{pl} \cdot V_{cruis} \cdot K \cdot \xi \cdot \eta}{m_0 \cdot C_{ef\ cruis}}. \tag{1.15}$$

If a helicopter is used as a flying crane to transport long-size cargoes, its application efficiency for such operations is estimated by (1.13) or (1.15).

Helicopter application for erection works or crane operations is mainly associated with hovering, so that allowances are made providing desired hovering duration and safety operation in case of one engine failure. Reasoning from safety operation of the flying crane helicopter, we can identify three kinds of relationships between the helicopter takeoff weight and payload:

– erection works must be performed during the period allocated for continuous operation at takeoff power under performing operations with the use of the engines takeoff power;

– under engine rating the cargo weight would be less than in the first case mentioned above;

– In case of one engine failure, the weight of the lifted cargo is determined on the assumption of the second engine operating under maximum possible power. Payload weight here may be close to null (zero).

So, the criteria used to estimate operation efficiency of a flying crane helicopter should be oriented to such types of operations to be mainly performed under hovering.

Specific capacity $\tilde{\Pi}_{cr}$ can be also used as a criterion for parameters optimization. For the flying crane helicopter, it is expressed by the ratio of relative capacity $\overline{\Pi}_{cr} = K_T \cdot \dfrac{m_{pl}}{m_0} = K_T \cdot \overline{m}_{pl}$ to the relative fuel flow per hour $\overline{q}_h = q_h / m_0$.

Reduced capacity of a flying crane helicopter is

$$\tilde{\Pi}_{cr} = \frac{\overline{\Pi}_{cr}}{q_h} = K_T \frac{m_{pl}}{q_h} . \tag{1.16}$$

From the dependence between a helicopter power and its weight under hovering (Velner expression), we have

$$\tilde{\Pi}_{cr} = 37.5 \cdot \eta_0 \cdot \xi \cdot K_T \frac{m_{pl}\sqrt{\Delta}}{C_e\sqrt{p}} , \tag{1.17}$$

where $K_T = T_{erect} / T_{flight}$ makes an allowance for the difference between T_{erect} being the time spent for erection and T_{flight} as time in flight (including time spent on manoeuvring prior to takeoff and after landing as well as time needed to get to the point of operation and fly back).

When specific operating conditions of a flying crane are unknown, we take $\mathbf{K_T}$ = 0.5 reasoning from the statistical data available.

With an allowance made for variations and limitations specified above, the equation becomes:

$$m_0 = \frac{m_{pl} + \mathrm{m}_{\mathrm{crew}} + \mathrm{m}_{\mathrm{eq}}}{1 - \overline{\mathrm{m}}_{fr}(\mathrm{p}) - \overline{\mathrm{m}}_{\mathrm{cpp}}(\mathrm{p}) - \overline{\mathrm{m}}_{fuel}(\mathrm{p})} , \tag{1.18}$$

where $\overline{m}_{fr}(p)$, $\overline{m}_{pp}(p)$, $\overline{m}_{fuel}(p)$ are formulae used to determine relative weight of the airframe, power plant and fuel under variations of $\mathbf{p}$ (from $\mathbf{p_{min}}$ to $\mathbf{p_{max}}$).

Algorithms which can be used to select main parameters of helicopters and their components with allowances made for weight characteristics will further below.

1.5. Classification of Helicopters

Any weight analysis starts from dividing rotorcraft into main weight groups and subgroups. Such presentation of weight is needed to provide analysis of the rotorcraft components, centre of gravity as well as moment of inertia calculation and weight monitoring throughout helicopter design and manufacturing.

Modern helicopter design has accumulated several variants of weight data presentation distinguished by completeness rate, group composition, terminology, etc. which do not provide a single and unified approach to data examination.

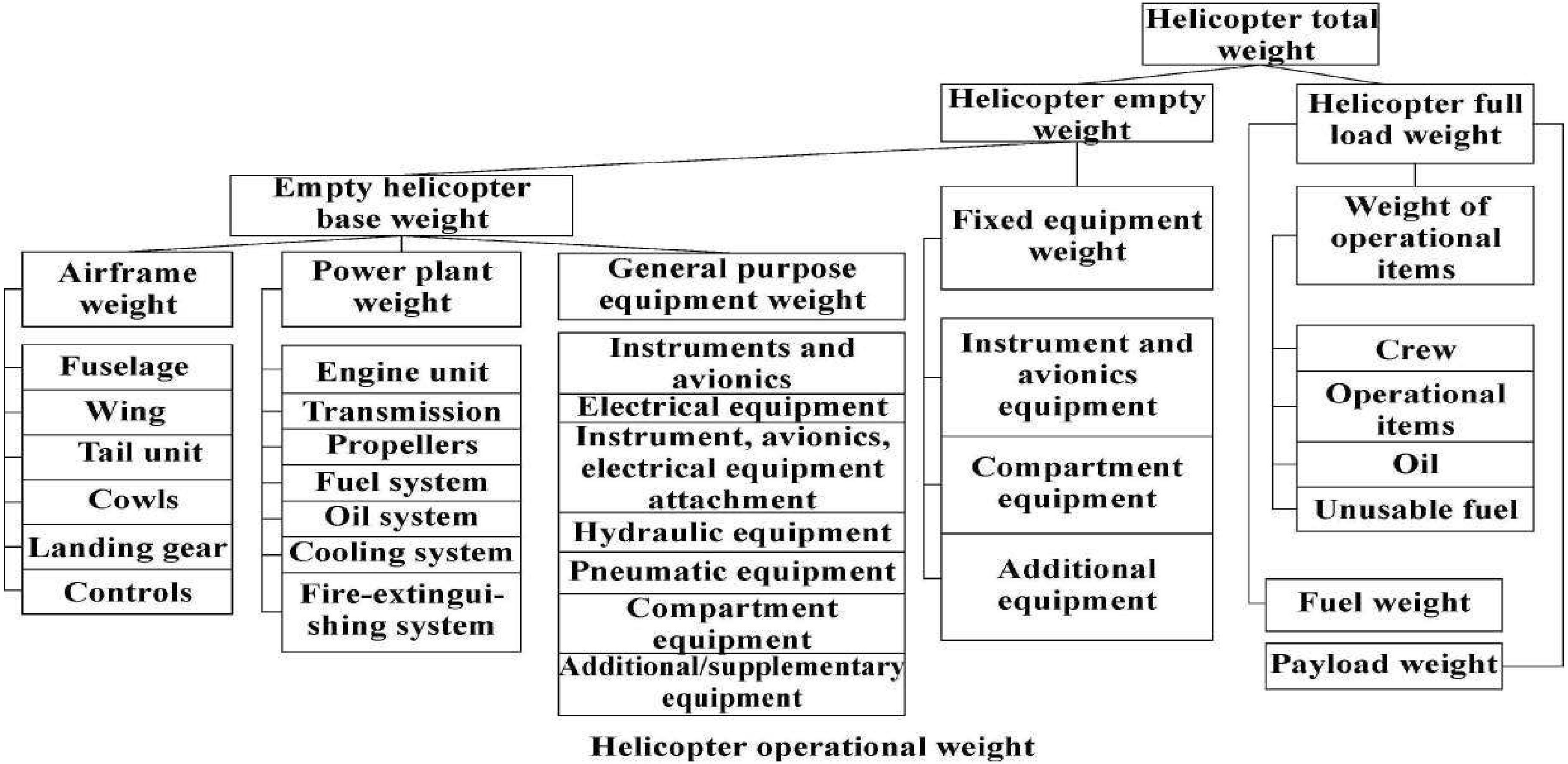


Fig. 1.11. An Example of Helicopter Classification by Its Weight

The following basic requirements can be subjected to the classification by weight:

1. Provision should be made in classification for a single valued division of helicopters weight into groups and subgroups on the basis of technical publications and unified terminology; and it must offer capability and convenient way for performing calculations to the required accuracy.

2. Classification should sufficiently enough correspond to different weight divisions used in international practice provided that it does not contradict fundamental principles of its formation.

3. Every helicopter weight group or subgroup should meet strictly specified functional purpose in its structure.

4. Classification should be persistent as per group and sub-group composition of rotorcraft weight.

5. Division of helicopter weight groups and sub-groups into smaller components should provide every design operation with a specified accuracy.

6. Data obtained in compliance with the classification take should serve as a reliable statistical material as they become accumulated as far as weight calculation methods and formulae continue their development for helicopter components.

Analysis of the available classifications demonstrates that classification shown in Fig. 1.11 [12] meets these requirements and is free of many disadvantages typical of other classifications known which makes it convenient to work with. From now on we shall use both this classification and terminology.

When selecting parameters and characteristics of any rotorcraft, designers use statistical data as a basis for their calculations. Therefore, they pay great attention to their collecting, processing, and analysing.

Statistical data should be organised in chronological sequence and cover helicopters of the same weight category as the one under design. Preference should be given to helicopters recently developed.

Analysed helicopter parameters and characteristics must be in harmony with each other. It is inadmissible to use data on helicopters with different takeoff weights. For example, under no condition should a variant with maximum payload be used while range is taken with an assumption for additional (at the expense of payload) fuel.

In order to provide compliance of statistical and design data, designers should keep to helicopter division by weight categories (Table 1.2).

Table 1.2

Approximate Classification of Helicopters by Weight Categories [14]

Helicopter Type	Light Liaison	Light Multi-purpose	Light Transport	Medium Transport	Heavy Transport
Lifting capacity $\bar{m}$	0.3	1.0…1.5	3.0…4.0	6…10	20
En-route weight $\bar{m}$	1.5…2	3.5	10…12	20…30	50

Statistical data are tabulated, processed, and then carefully analysed according to processing results. In this case, they help the designer to:

– learn about performance, weight and geometric parameters and characteristics of helicopters and their components, parameters of power plants, etc.;

– study the accumulated data on helicopters prototypes, their design, production, and operation;

– get idea about the modern level of helicopter development;

– find out current trends and prospects for the development of helicopter class under study;

– develop necessary TTR, etc.

In addition to digital material of the statistical table, designers should have three general arrangement drawings of every variant for at least five variants.

In order to select optimal rotorcraft parameters, designers calculate weight still at the initial stage of design. But such calculation is probable only after design documentation is developed and issued. Therefore, at the stage of preliminary design, the weight of units (groups) is calculated by simplified formulae (as a mass) which specify how the weight of helicopter or its parts depends on structural dimension, ultimate load, materials, etc.

These formulae represent peculiar features of unit design, specific loading conditions (e.g., priority of fatigue strength over static one), suppression of any kind of propeller self-excited vibrations and oscillations as well as other phenomena associated with the peculiarities of helicopter structure.

Variable cycle loads with a great number of cycles (10 … 100 mln and in some cases, up to 1 billion of loads) are assumed to be the main loads specifying weight of the most units (instead of static or repeated static loads). Weight of such units as blades, hubs, swash plate assembly, actuating portion of system, main drive shafts, gear box mount as well as many others is almost dully dictated by the fatigue strength. The weight of fuselage, gear box casing, and landing gear is largely determined by the fatigue or repeated-static loads.

Designers should use mass formulae in the following basis cases:

1) to estimate weight of a helicopter under design;
2) to select of structural parameters under specified strength;
3) to calculate optimal parameters for a helicopter.

There are three main groups of methods used at the stage of the preliminary design in order to calculate weight:

1) theoretical methods based on strength analysis;
2) empirical methods based on the use of statistical data only;
3) mixed (or semi-empirical) methods accounting for the results of general analysis with a wide application of statistical data.

Theoretical methods pay special attention to the load factors acting on the unit model which weight is to be determined. Geometric parameters providing desired strength and, thus, unit weight are found according to tolerable stresses and acting load factors.

Though theoretical methods are considered to be the most precise ones, they are not free from inaccuracies coming from many factors, e.g. production requirements. In reality, it is difficult to manufacture structures equal in strength or use only components which weight precisely equal to design considerations. That is why statistical correction is introduced in weight calculation formulae.

Empirical methods reveal dependence between the helicopter weight and its specific parameters: If you know helicopter parameters, you can determine its weight. Though empirical methods based on statistical data are less reliable than theoretical ones, they also find wide application.

Mixed (or semi-empirical) methods both apply peculiar theoretical dependencies between helicopter parameters and its weight and make allowances based on statistical data. Thus, they hold an intermediate position between theoretical and empirical methods.

Formulae which come from mixed methods provide less accurate results as compared to the detailed strength analysis but they are more illustrative and simple and secure sufficiently reliable determination of weight for helicopters which parameters are far beyond the usual parameter change range.

At the stage of preliminary design, when basic helicopter characteristics and parameters are being selected, application of empirical formulae is quite reasonable. Further in design process, when more accurate determination of helicopter parameters and performance are required, it becomes feasible to use semi-empirical formulae to calculate weight. At last, the most accurate theoretical formulae should be used when estimating structural weight based on the selected parameters as far as optimal parameters are clarified and refined analysis of the performance is in progress.

Semi-empiric formulae seem to be a good choice because they both provide a reasonable accuracy in the sufficiently wide range of the parameters change and demonstrate the effect produced on the weight by basic parameters.

At the same time, they should not be considered universal because employed therein statistical coefficients reflect only current level of helicopter manufacturing. For another thing, certain looseness of results may come from deviations produced by scale affect, whereby a small unit proves to be relatively heavier than the larger one optimal as per weight. On this basis, we would say that semi-empiric formulae can be fruitfully employed in helicopter design provided that designers use available experience as a starting point and correct statistical coefficients with allowances made for configuration, manufacturing process, available materials, etc. Moreover, they should keep in mind that even units with optimal design, as a rule, demonstrate certain dissimilarity as per their weight characteristics.

Weight estimation formulae based on theoretical prerequisites with the use of statistical data provide reliable platform for comparative estimation of different helicopter versions as well as for finding the most favourable combinations of helicopter parameters reasoning from imposed requirements.

Though manufacturing progress, improvements in helicopter design and production, application of new structural materials frequently outlive statistical coefficients used in mixed formulae, these coefficients can be updated due to the fact that the nature of associations between parameters and weight remains the same and used successfully after allowances are made for the achieved level of helicopter industry.

So, mass (weight) balance equations are used to estimate weight and establish dependencies between relative weight $\bar{m}_{fr}$, $\bar{m}_{pp}$, $\bar{m}_{fuel}$ and their specifying parameters.

If presented in a formulae form, quantitative relationship between unit parameters and helicopter systems help designers to:

– compare real and calculated weight and find the reasons for structure overweight and ways of its overcoming;

– study the influence produced by helicopter parameters on its performance (parametric investigation) with a high degree of certainty;

– conduct wide research in optimal parameters for helicopters under design and compare their configurations;

– establish grounded weight limits.

Weight limit is a quantitative indicator as per weight planning assigned to a designer developing helicopter structure, equipment, systems, and units. Conceptually, weight limit is a minimal theoretical value of an article weight (flying vehicle, engine, their components, units, assemblies) than can be proved by the calculated and statistical data and realized by a complex of scientific research and organizational measures.

Weight calculation accuracy must be comparable with the accuracy of the initial design data and prerequisites.

It is an undeniable law of aircraft development that helicopters become heavier on their way from technical offer to serial manufacturing. This overweight comes from continuously originated new requirements, demands, ideas, and contingencies which need more profound structure development, study, and test. Therefore, a margin for structure overweight equal to 1.1 [12] is entered into the helicopter weight breakdown at the stage of technical offer development. With allowances made for probable overweight of blades, rotor hubs must be calculated with an account made for the centrifugal focus of blades as their weight grows by 10 % [12].

1.6. Certification Legal Foundation

Certification is an activity which serves to confirm that certification products comply with the specified requirements for aerotechnics. In this tutorial, we understand aerotechnics (AT) as all aircraft and their components used for equipping air-supported vehicles.

Here is a short list of the things and entities subjected to compulsory certification:

– pilot projects;

– aircraft and aerotechnics manufacturers;

– aviation companies providing transportation, freighting and aviation operations, aircraft maintenance and overhaul;

– aerodromes and airports;

– educational establishments which train respective specialists;

– air-supported vehicles, aircraft engines, propellers, aviation grounds, and other types of equipment as well as legal person when activity is related to the provision of flight safety.

Aerotechnics certification is a part of the flight safety securing system aimed at proving out that aerotechnics operates in full compliance with the airworthiness requirements and environment protection regulations.

The fact that certification object fulfils requirements is certified by a document, airworthiness certificate issued by the authorized special agency responsible for organization and compulsory certification of aviation and civil purpose aerotechnics. Certification system includes the national helicopter airworthiness requirement (NHAR).

Airworthiness certificate is a document certifying that a civil aircraft meets the helicopter airworthiness requirements and answers aviation regulations within the range of the specified operating conditions (limitations). It is issued on the basis of drawings, instructions, calculation and analysis results, bench tests, etc. Airworthiness certificate confers entitlement of this type civil flying vehicle to operate.

Certification of air-supported vehicle is conducted by State Aviation Register together with IAC; aircraft development contractors, designers for engines and equipment; Industry and Civil Aviation Scientific and Research Institute, and Aviation Research Institute. Basic concepts of certification are defined in *Civil air-supported vehicle certification regulations* and APU-21.

NHAR and Regulations specify that any aircraft can be admitted to operation only if it complies with the effective NHAR and AR which is confirmed by airworthiness certificate and certificate proving that this aircraft type is ready to flight. NHAR and AR also determine the compulsory order and procedures employed to estimate the aircraft compliance with regulations. According to NHAR and Regulations, equipment and engines undergo certification “prior to installation at aircraft, air-supported vehicle certification”.

Certification accompanies the whole process of aircraft design. It covers a wide range of investigations and estimations at every stage of object development. That is why it should be well programed to provide for every type of work as well as all facilities needed [15].

Provisions for the aircraft’s correspondence to the requirements are ensured at the stages of aircraft design, modelling, prototyping, and manufacturing, and, especially, under flight tests.

At the stage of draft, they check whether the design of certification objects as well as estimation methods correspond to the effective norms and formulate the certification program.

At the further stages including helicopter design, modelling, and prototyping, allowances are made for most NHAR and AR requirements. Actually, prototypes can be used to evaluate cockpit, passenger compartments (including emergency exits, seats, and emergency escape equipment), baggage-luggage compartments, composition and arrangement of the airborne equipment, power plant layout, etc.

Functional systems such as control, power supply, flight navigation, and life support systems are developed at the stage of the rotorcraft manufacturing in the course of tests performed at full and semi full size test benches as well as during flight test (flying laboratories are essential here). These are necessary to study aftereffects of functional system failures as well as flight dynamics with the flying personnel involved. At the same time, they conduct detailed tests of rotorcraft structure and its systems to secure their correspondence to helicopter Airworthiness Regulations and Aviation Rules on aircraft strength requirements.

All the rotorcraft equipment including engines is subjected to certification they are installed at the aircraft. Every article installed must meet category-specific requirements. Certification of the equipment “prior to installation in aircraft” involves thorough estimation of its correspondence to equipment specifications based on the laboratory and bench tests. This means that their structure, operability, and characteristics are tested in order to find out how they behave when exposed to forces (vibrations, temperature, pressure, etc.). Tests conducted with the employment of benches, simulators, and flying laboratories make it possible to provide maximum rotorcraft readiness for flight tests. If properly implemented, certification program gives the chance to complete around 60 % of rotorcraft certification procedures prior to the beginning of the flight tests and to reduce its period significantly.

Aircraft flight tests are final and the most responsible stage of certification. They provide detailed testing of the rotorcraft and its total functional systems (including engines and equipment) under conditions most close to real operation. Flight tests account for the estimation of about 40% of helicopter Airworthiness Regulations and Aviation Rules requirements. These are stability and controllability requirements, requirements for strength, critical flight (operating) conditions, control systems, power plant, and flight navigation systems as well as requirements for flight safety in case of functional system failures and unfavourable weather conditions (icing, poor weather for landing, etc.). As flight tests are the most complicated stage in rotorcraft development and certification which influence significantly the total duration of the aircraft development cycle, it is really important to prepare a test program that would incorporate methods and means able to secure maximal test intensity (e.g. simultaneous flight tests for several rotorcraft with specified missions; automated processing of test results, etc.).

All-in-all, certification flight tests can be divided into state and factory ones. State certification flight tests serve to provide reference evaluation and to prove that the rotorcraft fulfils helicopter Airworthiness Regulations and Aviation Rules requirements. These tests usually take account of the scope and results of factory tests. If the results of both factory and state tests are evaluated positively, the airworthiness certificate is issued for the rotorcraft type granting permission to stat its operation.

Fig. 1.12 provides you with a simplified diagram of helicopter specification.

The overall scheme of airworthiness certification refers to rotorcraft certification as well. In addition, rotorcraft certification involves certification of development organization, rotorcraft type, operator, aerodrome, airport, flying personnel, etc. Without the airworthiness certificate, rotorcraft cannot be legally operated or sold.

Today, Ukrainian system of Aviation Regulations is brought close to those of the USA and European Union. Legal foundations of the AR are specified in Ukraine by the Air Code which determines air legislation.

In compliance with a joint agreement on civil aviation of ex-USSR states, International Aviation Committee (MAKrus – IACeng) is the executive authority responsible for all activities associated with rotorcraft certification.

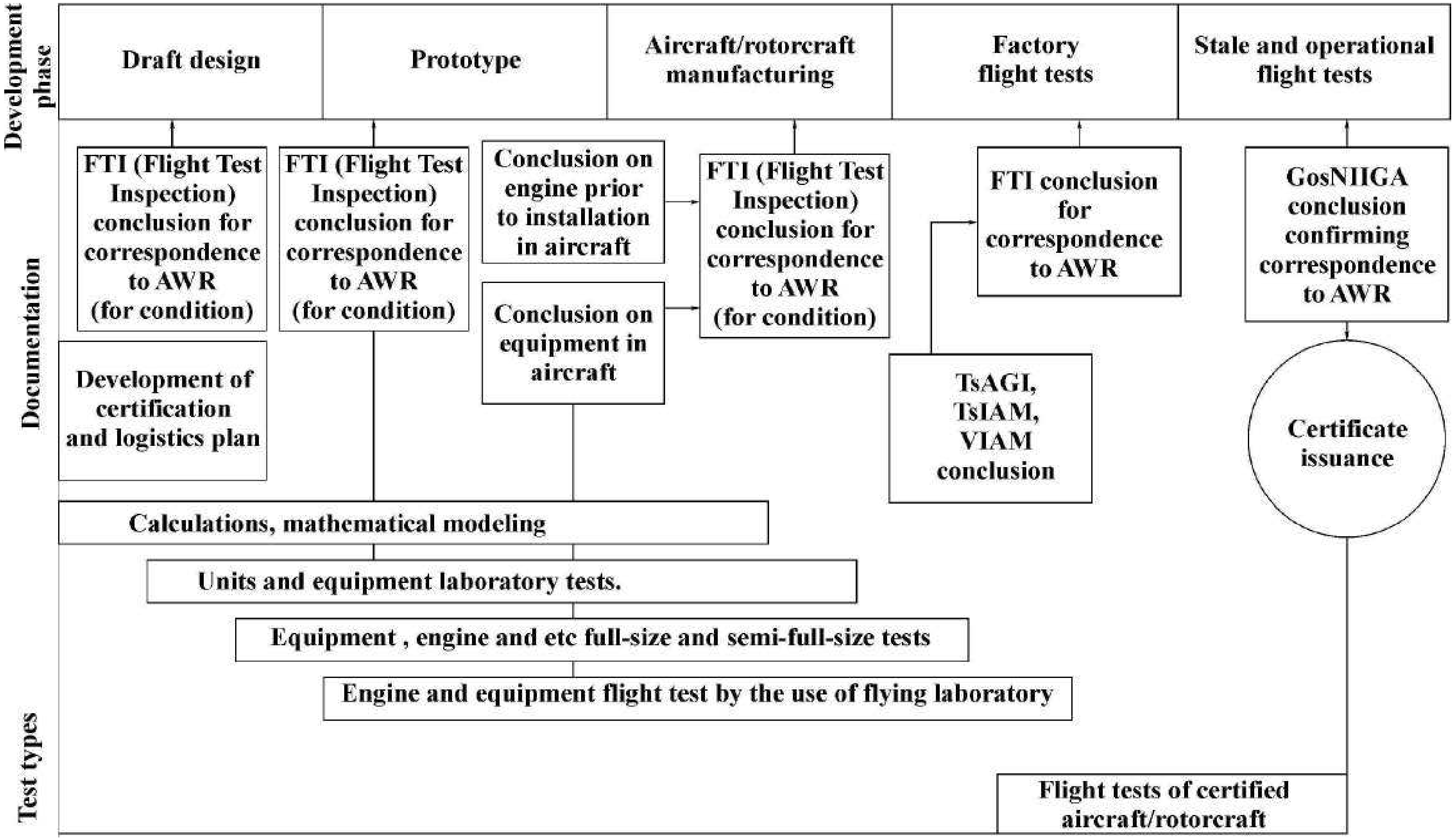


Fig. 1.12. Civil Aircraft/Rotorcraft Certification Block Diagram

Aviation regulations and requirements as well as their codes are given in Table 1.3.

Table 1.3

List of AR for Civil Rotorcraft Certification

Aviation regulations (AR)	Code
AR, part 21. Rotorcraft Certification Procedures	AR-21
AR, part 27. Airworthiness Standards/Requirements for Normal Category Prop Rotorcraft	AR-27
AR, part 29. Airworthiness Standards/Requirements for Transport Category Prop Rotorcraft	AR-29
AR, part 33. Airworthiness Standards/Requirements for Rotorcraft Engines	AR-33
AR, part VD. Airworthiness Standards/Requirements for Rotorcraft Auxiliary Engines	AR-VD
AR, part 34. Environmental Protection. Emission Standards for Rotorcraft Engines	AR-34
AR, part 35. Airworthiness Standards/Requirements for Propellers	AR-35
AR, part 36. Rotocraft Perceivable Noise Certification	AR-36
AR, part 39. Airworthiness Directives	AR-39
AR, part 145. Maintenance and Overhaul Companies	AR-145
AR, part 183. Aviation Register Representatives	AR-183

2. HELICOPTER DESIGN

Development of a helicopter pilot design is the final phase of training complex for bachelors majoring in *Aviation and aeronautics*. It incorporates several stages:

– design of the helicopter appearance based on statistical data;
– calculation of its performance and aerodynamic characteristics;
– calculations for static strength for regular zone components;
– design assembly units (spars, panels, etc.), primary structures of push-pull control linkage, engine attachment system;
– planning procedures for manufacturing helicopter parts including sheet metal stamping and machining;
– analysis of economic efficiency;
– development of environmental safety security system;
– special-purpose facilities.

A pilot design is performed according to the procedure of the helicopter preliminary designing. To be successful in it, future designers should integrate their knowledge in the design of airplanes and helicopters, power plants, helicopter systems, units, components, aerohydrodynamics, strength, aircraft manufacturing, material science, economy, and other fields. Furthermore, they should introduce all the accumulated experience of foreign and domestic helicopter design into their work as well as study peculiarities of their design and layout, load-carrying diagrams of their components, applicable structural materials, production and operation, and, to top this all, reveal development trends and prospects.

2.1. Goals and Organization of Design Process

Helicopter design aims at developing a new helicopter possessing unique characteristics in order to overcome challenges traditional for its type. In other words, the goal of helicopter design lies in creating rotorcraft which would provide the highest efficiency under specified limitations. These limitations are dictated by:

– physical laws of aircraft flight mechanics according to aerodynamic principle;
– current scientific and technical achievements in aircraft engineering and adjacent branches of industry;
– manufacturing methods and economical aspects.

Modern helicopter is a sophisticated technical complex with a highly developed hierarchy and a great number of units, components, and internal links and connections. For any helicopter, they provide:

– lift creation;
– stability and controllability at desired path;

– motive (driving) force;
– fulfilment of target function;
– life support;
– control and navigation under different flight conditions, etc.

All helicopter systems are interrelated and interconnected with most of them incorporating sets of less sophisticated systems and subsystems.

As an aviation complex as well as an important part of country's transport and defence systems (see Fig. 1.1), helicopters integrally combine manpower and material (physical) resources to perform desired useful functions. Helicopter design procedures are roughly the same that for all aircraft (refer to Fig. 2.1).

As helicopters are both complex systems and parts of systems yet more complex, their design requires system approach. Established scheme of research and engineering investigation associated with the development of a new helicopter is shown in Fig. 2.1 in the form of a multistage process of decisions making with the iteration cycles both between separate stages and within every stage.

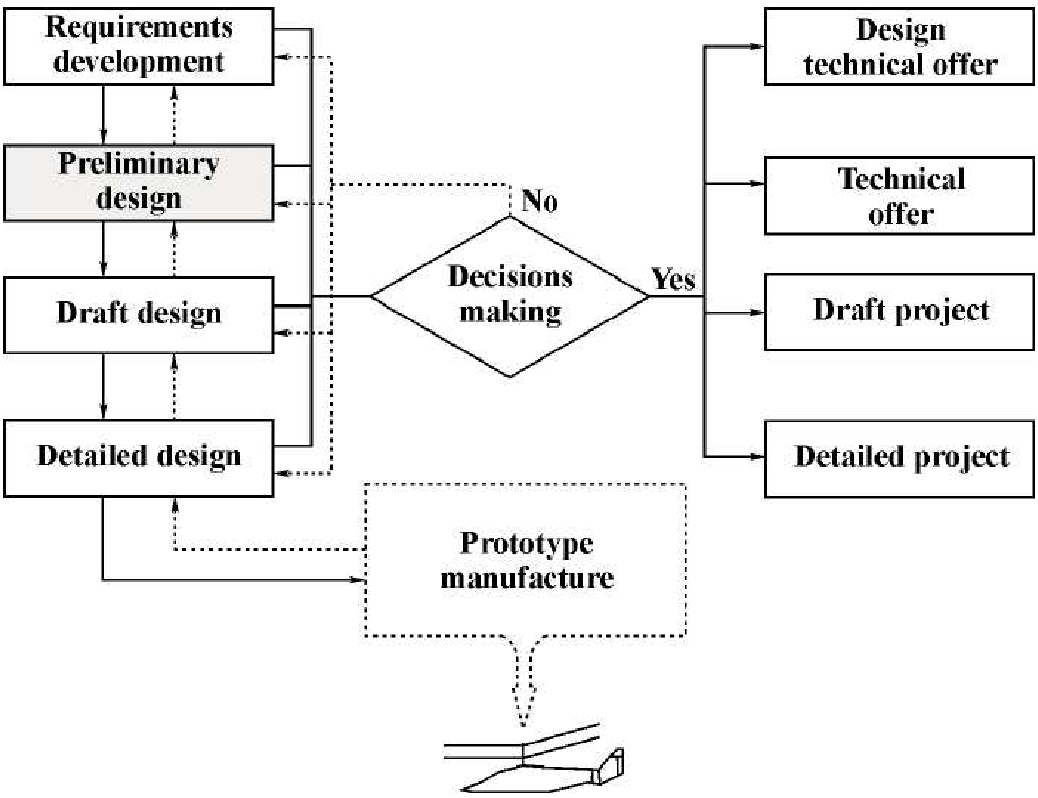


Fig. 2.1. Main Phases of Helicopter Project Development

This process can be largely divided into four stages which are interrelated but still differ in their goals and objects.

2.2. Stages of Helicopter Design

The process of helicopter design includes several stages.

The first stage of "outside design" involves investigation of sophisticated organizational-technical systems of which helicopters are their component.

At this stage, designers make decisions on the helicopter type, desired characteristics and their distribution by airlines, base aerodromes and their equipment, classification and parameters of the cargo to be transported, flying personnel, etc. They introduce the conception of future helicopter and define quantitative design targets as well as list functions the helicopter must realize under particular operating conditions.

These investigations result in the technical assignment that provides the whole development procedure and its parts.

The second design stage is the development of the technical offer (preliminary designing, pilot project). It consists in selecting the helicopter configuration and determining its basic parameters, systems and components which would fulfil specified functions.

At this stage, the helicopter conception stated before is realized on the basis of the thorough analysis of technical assignment, chief designer's ideas, design office experience, and SRI recommendations.

Designers determine basic dimensional, weight, and power characteristics of the helicopter under design at the first approximation; formulate control principles for different path segments and elaborate them for different flight profiles envisaged by the requirements. In addition, they select and adopt helicopter's geometric, weight, and aerodynamic characteristics, engine altitude performance and fuel consumption, airframe structure, equipment, flight-technical data, combat and transport efficiency, etc.

The stage expected outcome includes helicopter arrangement and layout drawings as well as necessary documentation describing its design performance, economical and operating characteristics. These documents and drawings are the basis for the authorized persona and organisations to decide whether the project is feasible or its technical assignment must be improved.

Preliminary design may be followed by the development of the technical offer (TO) if such provision is made in the technical assignment.

Technical offer is intended to reveal additional or clarified requirements to the article (specifications, quality index, etc.) which couldn't be specified in the technical assignment though can be performed on the basis of preliminary design and analysis of possible variants.

The third stage of draft design brings the preliminary design into accordance with many rather contradictive requirements including operating and production ones. Here, the helicopter's centre of gravity is clarified.

After the centre of gravity is calculated, designers address themselves to determining weight breakdown based on strength and weight analysis of the airframe, power plant, equipment, operational items, intended cargo, etc.

At this design stage, helicopter's units and systems are subjected to many theoretical investigations and experimentation: Models of the helicopter and separate helicopter components are tested in wind tunnel; aerodynamic design is clarified as well as stability, controllability, and aero-elasticity.

There researches provide the necessary data for improving helicopter arrangement and correcting weight calculations. Full size or computer-generated models are developed to study space interrelation of helicopter systems and units, equipment arrangement; to estimate crew and payload.

The stage ultimately results in draft or conceptual design which includes clarified characteristics of helicopter's units and functional components. After this, the authorized committee evaluates the project and makes a decision about its future implementation.

Second and third stages account for general design of the helicopter as an integral system.

The fourth stage involves the detailed design providing for technical documentation needed for production, assembly, and installation of helicopter units and systems as well as the helicopter as a whole. Moreover, it includes structure design and engineering aimed at helicopter's actual manufacturing, namely:

– research and experimentation with new materials and structures;
– static and dynamic tests for strength, vibration, and service life;
– bench tests of equipment as well as control and life support systems.

Information obtained at this stage enables the designers to specify and improve project data, revise weight and strength calculations.

Prototypes, ground and flight tests are the integral part of the helicopter design. They help helicopter developers to correct technical documentation and optimize manufacturing procedures; to determine real characteristics of the helicopter and find out whether they correspond to the technical assignment; to decide whether the design needs alterations; to male suggestions about the helicopter's series production.

Do not forget that technical assignment always comes along with design initial data (ID).

2.3. Contents of Technical Assignment

Technical assignment of a conventional helicopter typically includes the following sections:

Introduction

1. Purpose of helicopter
2. General requirements

3. Expected operating conditions
4. Performance requirements
5. Requirements for helicopter structure
6. Requirements for operational systems and equipment
    6.1. Passenger and buffet/galley equipment layout
    6.2. Control system
    6.3. Hydraulic system and landing gear
    6.4. Air-conditioning system
    6.5. Compartment (air) pressure control system
    6.6. Oxygen equipment
    6.7. Flight data recording system
    6.8. Power plant
    6.9. Fuel system
    6.10. Auxiliary power unit
    6.11. Power plant instruments
    6.12. Electro technical equipment
    6.13. Compartment lighting
    6.14. Exterior lights
    6.15. Flight navigation equipment
    6.16. Radio communication equipment
    6.17. Fire protection system
    6.18. Crew protection means
7. Requirements for environmental protection
8. Physiological-sanitary requirements
9. Requirements for operational maintainability, controllability, technical diagnostics means, maintenance and overhaul
    9.1. Maintenance and overhaul system. General requirements
    9.2. Maintenance conditions
    9.3. Overhaul conditions
    9.4. Testability
    9.5. Flight navigation and avionics equipment. Maintenance and repair requirements
10. Ground servicing
11. Requirements for manual operation
12. Requirements for evacuation means
13. Flight safety security means
14. Requirements for metrological support of helicopter development, production, test, and operation
15. Requirements for helicopter operation in foreign airports.

Supplement to TA.

2.4. Initial Data

Helicopter design data incorporate:

– technical assignment together with checklist data specifying design targets and limitations;

– statistical data showing world level of achievements in the field of helicopter design for the specific helicopter type;

– characteristics, fields of application, and lists of the new technology gains in the field of science and engineering which can be used to improve the helicopter under design.

Technical assignment is developed in cooperation with bodies interested in the development of a new helicopter (customer) and design offices involved in helicopter development (development contractor). Both the customer and the development contractor may take a lead in design procedures.

Depending on the helicopter type, technical assignment incorporates the following sections.

Purpose lays down the type of a new helicopter and its purpose.

General names standard material and procedure specifications which a new helicopter must meet. These are airworthiness standards for civil helicopters (Aviation Regulations AR-27, AR-29); noise and environment standards; general specifications and limitations regulated by industry standards (e.g., general technical requirements for passenger compartment layout, its furnishing and auxiliary equipment).

Performance lists specified output characteristics such as speed, range and flight altitude, lifting capacity and classes of transported cargo, takeoff/landing and manoeuvrable characteristics, etc.

Production helicopter technical level and economical characteristics section determines helicopter efficiency (transport, combat, fuel, etc.) thus providing for reasonable evaluation of its cost and compatibility. These data make it possible to establish optimization criteria for design project.

Prospects for further development of helicopter and its principal systems section contributes information on the development trends concerning the helicopter under design and probable variants of its modification.

Helicopter structure, systems, and equipment section specifies the service life (in flying hours and number of landings) and lifetime (in years, total and to first overhaul) of both airframe and strong components; put down service life requirements to vendor items and equipment.

Power plant and its systems section sets basic characteristics of the engine used, modified, or developed for the helicopter under design including its service life and noise parameters. Newly designed engines require special technical assignments.

Special equipment gives the account of fundamental problems which must be solved by this equipment including flight navigation, automatic flight control, communication between the helicopter and ground air traffic control systems, helicopter intercommunication, special equipment and airborne systems as well as flight data recording.

Requirements to reliability, maintenance and overhaul system section accommodates quantitative data on probable abnormal inflight circumstances, helicopter fleet reliability rate in the course of operation, probability of on-time departure as well as maintenance and overhaul program.

Level of standardization and unification ascertains the standards which secure necessary unification of helicopters, their engines, vendor items and equipment.

Technical assignment regulates and determines design process imposing significant limitations on the selection of design concepts or even directly defining them. For example, safety requirements under emergency or passengers comfort level dictate the layout of passenger compartment, number of engines, etc.

Students should remember that it is a rare case then designers have the final variant of technical assignment at their disposal prior to the beginning of the project development. Generally, technical assignment is developed, improved, and supplemented together with the new helicopter itself.

A new helicopter design project can be successful only if it incorporates advanced scientific and research conceptions accumulated at different branches of science and engineering and available at the moment when the design process starts. It highly relies on new structural materials, manufacturing procedures, attachment fittings, improved aerodynamic characteristics, new engines with enhanced specific parameters, advanced airborne equipment and systems, etc.

Experts say that about 50 to 150 new technical solutions should be realized to get a competitive helicopter. At least 2/3 of them should be developed and proved prior to the start of the project in order to reduce technical risk. Academies of Science, branch scientific and research institutions, design offices, and universities work hard and provide for research and experimentation necessary to develop a new helicopter.

2.5. Technical Specifications Supplementing Operation Requirements

Helicopter airworthiness standards, aviation regulations, FAR, ICAO standards form legal basis for elaboration of operation requirements.

Under no condition, a helicopter can be developed that would meet every requirement equally. No qualitative or quantitative relationships can overcome these contradictions because there is no single design recipe. But there are certain physical mechanisms which help helicopter designers to meet the requirements in the best way.

Helicopter performance mainly includes

1. Crew weight, m_{cr}, kg.
2. Payload weight, m_{pl}, kg.
3. Maximum velocity, $\mathbf{V_{max}}$, km/h.
4. Maximum range under standard takeoff mass, $\mathbf{L_{max}}$, km.
5. Hovering ceiling, H_{hov}, m.
6. Dynamic ceiling, H_{dyn}, m.
7. Helicopter weight category.
8. Range and value of payload.

These are the parameters for future helicopter designers to know at the stage *of preliminary design* of their projects.

1. Orienting to the specified payload weight m_{pl} with the provision made for correspondence to the statistical data, the helicopter takeoff weight is subjected to determination at zero approximation

$$m_0^0 \approx 4.48 \cdot m_{pl}^{0.92}; \qquad (2.1)$$

$$m_{ul} = K_{full\ l} \cdot m_{pl}, \qquad (2.2)$$

where $K_{full\ l}$ = 1.2…1.3 depending on the type of engine installed and fuel quantity (range); $K_{full\ l} \approx$ 1.25 is taken under zero approximation; m_{ul} is useful load weight.

Weight category of the helicopter under design is specified according to the obtained value of $\boldsymbol{m_0^0}$.

2. Unit load $\mathbf{p_i}$ is better determined as $p_3 = p_{aver} = \sum_{i=1}^{5} p_i / 5$ where 5 is a number of helicopters available in the statistical table $\Delta\mathbf{p}$ = 25 N/m^2; $p_1 = p_{aver} - 2 \cdot \Delta p$; $p_2 = p_{aver} - \Delta p$; $p_4 = p_{aver} + \Delta p$; $p_5 = p_{aver} + 2 \cdot \Delta p$ taken as a variation step of light helicopters unit load, and $\Delta\mathbf{p}$ = 50 N/m^2 for medium helicopters.

3. With an allowance made for the helicopter engineering experience according to the statistics, tip speed value ωR is selected as

$$\omega R = \begin{cases} 180 \ldots 200\ m/s\ under\ \ \boldsymbol{m_0} < 10\ t; \\ 200 \ldots 210\ m/s\ under\ \ 10\ t \le \boldsymbol{m_0} \le 25\ t; \\ 220 \ldots 230\ m/s\ under\ \ \boldsymbol{m_0} > 25\ t. \end{cases} \qquad (2.3)$$

4. Statistical table presents data on helicopters, MBB/Kawasaki VK-117; Augusta A129; Bell 222A; Westland Whirlwind sr.3.HAR.Mk.10; and Sikorsky S-55.

5. NACA 23012 airfoil is taken for propeller blade airfoil portion.

6. Tip speed is assumed to be equal to ωR, *m/s*.

7. According to the statistical data, the takeoff weight at zero approximation is taken as m_o^o, kg.

8. For light helicopters, relative fuel quantity under the specified range **L** at the altitude $\mathbf{H} = 500$ m and a cruising speed V_{cruis} under $\omega R = const$ can be determined as

$$\bar{m}_{fuel} = k_{fuel} \frac{C_{e\,cruis} \cdot L}{V_{cruis}} \cdot \left(0.765 \cdot \tilde{N}_{0\,max}\right) \cdot g \,, \qquad (2.4)$$

where $k_{fuel} \approx 1.19$ is the coefficient accounting for 5% litre fuel reserve and fuel consumption under transient conditions, etc.

9. Propeller blade airfoil portion, its aerodynamic and geometric characteristics.

Design data supplementary digits should be realized during the design progress.

All calculations are recommended to be tabulated.

Helicopter aerodynamic design is an integral part of the general design intended for testing its performance feasibility which requires certain design procedures.

3. SELECTION OF HELICOPTER PARAMETERS

3.1. Equation of Existence for Helicopters

As we know, helicopter is a complex system characterized by many parameters which are chosen so that they meet specific requirements for structure, strength, technology, economy, operation, etc.

Agreed compromise between these requirements is traditionally expressed in the form of mass balance equation (or existence equation) since minimal takeoff mass is one of the key assessment criteria for the efficiency of design object.

According to the mass classification of helicopters (see Fig. 1.9) [12], their takeoff mass can be given the following form of mass balance equation:

$$m_0 = m_{airfr} + m_{p-pl.} + m_{fuel} + m_{payload} + m_{crew} + m_{eq}, \tag{3.1}$$

where m_{airfr}, $m_{p.pl}$, m_{fuel}, $m_{payload}$, m_{crew}, and m_{eq} are masses of airframe, power plant, fuel, payload, crew members, and equipment.

If we assign $m_{payload}$ and m_{crew} and define m_{eq} proceeding from the helicopter's purpose and operating conditions, mass balance equation can be written in relative values:

$$1 = \overline{m}_{airfr} + \overline{m}_{p-pl.} + \overline{m}_{fuel} + \overline{m}_{payload} + \overline{m}_{crew} + \overline{m}_{eq}, \tag{3.2}$$

where $\overline{m}_{airfr}$, $\overline{m}_{p-pl.}$, $\overline{m}_{fuel}$ are relative masses of airframe, power plant, and fuel; $\overline{m}_{airfr} = m_{airfr} / m_0$, $\overline{m}_{p-pl.} = m_{p-pl.} / m_0$, $\overline{m}_{fuel} = m_{fuel} / m_0$.

According to Tyshchenko et al. [12], mass balance equation with its components expressed via parameters of aircraft and engine as well as via aerodynamic and performance characteristics is thus transformed into the equation representing interconnected and mutually compelled features of helicopter.

(3.2) connects helicopter parameters and characteristics. It is called the equation of existence because any helicopter to be developed can combine not any features but only features which meet this equation. Functions included in the equation vary depending on current level of science and engineering.

Equation of existence relies on the fact that aircraft mass balance may be also considered as the balance of all its features, as the condition for their quantitative compatibility, and, finally, as the condition under which it becomes possible for a helicopter designer to develop a helicopter with certain assigned complex of characteristics. In other words aircraft mass is the integral expression of its features. Enhancing certain features, you worsen others.

Masses of an aircraft and its groups are the equivalents of corresponding parameters and units.

The equation of existence leads us to the following conclusions [12]:

1. Designers can provide a helicopter with a certain complex of features only depending on its structural perfection with regard to its mass.
2. For the first time helicopter was developed at certain level of its structure perfection allowing keeping the mass balance under keeping required minimum of helicopter performance characteristics.
3. For a given m_0, helicopter may possess only certain limited features.
4. For a given m_0, every helicopter feature should not exceed its limit values.
5. All helicopter features are qualitatively related which means that for $m_0 = const$, some of them can be increased only at the expense of the others (which are reduced).
6. Helicopter mass must be re-arranged based on its purpose (speed, flight range, etc.).
7. If you increase the number of features specific for the helicopter (and change nothing else), you increase its takeoff mass.

Equation of existence does not answer the question how to reasonably chosen features and parameters. To find the answer, designers use a number of efficiency criteria which help them to optimise the helicopter under design.

3.2. Selecting Blade Airfoil

Aerodynamics of a helicopter’s carrying surface (rotor blade) depends on the shape of its sections.

Airfoil is an outline formed by the external (theoretical) blade surface crossing the plane perpendicular to its longitudinal axis. Airfoil is characterized by a set of parameters, namely chord b, airfoil thickness ratio ($\bar{c} = c/b$), airfoil centreline (camber) f, relative curvature (concavity) $\bar{f} = f/b$, pressure centre position $X_{press-cent}$, and aerodynamic centre $X_{aer-cent}$.

For a blade airfoil, helicopter designers must pay attention to a number of influential factors including structural-technological limits of spar arrangement, anti-flutter mass, anti-ice system, materials of which blade bearing members are made, limitations of blade production.

The airfoil itself (as an outline, a closed line) cannot develop neither lifting force nor drag. That is why we consider aerodynamic characteristics of a blade of infinite span.

Airfoil characteristics used in helicopter aerodynamics are the relations of lift coefficient C_y, drag coefficient C_x and longitudinal moment m_z of the blade airfoil with infinite span with regard to the angle of attack α and Mach number M of air flow.

Blade airfoil must meet the following requirements.

1. Blade airfoil must have high lift-to-drag ratio $K = C_y / C_x$.

2. Blade airfoil must have high critical Mach number M_{crit}.

3. Blade airfoil must have its centre of pressure (CP) immobile (or slightly shifted) chordwise at different angles of attack α (including blade pitch φ) because the shift of centre of pressure along the airfoil backwards develops the pitching down moment which will twist the blade to reduce the angle of attack and, therefore, to reducing blade lifting force of the main rotor (MR).

4. Blade airfoil must facilitate the transition to autorotation within a wide range of angles of attack so that the main rotor will transit to autorotation at any pitch in case of the engine failure. This requirement is fulfilled if the blade airfoil provides high lift-to-drag ratio which remains unchanged (or slightly changes) within a wide range of angles of attack.

5. Assigned airfoil shape and sizes must be reproduced scrupulously when blades are manufactured. Otherwise, its aerodynamic and other characteristics change, and the helicopter's performance characteristics worsen. Departure from the theoretical blade airfoil result in corrugations caused by manufacturing errors and unspecified loads.

6. Blade airfoil must provide minimal drag of the blade itself. Drag of the blade depends on the blade coating and finishing quality. Blade surface must be extremely smooth not only to reduce total blade drag but also to reduce main rotor power and increase critical angle at which stall occurs.

7. Chordwise, centre of gravity must have the extreme forward position in order to prevent blade "flutter".

Aerodynamic investigations are usually performed not into one airfoil but into the family of airfoils. *Airfoil family* is a group of airfoils which have the same nature of centreline and shape though their other geometrical characteristics (thickness ratio, airfoil camber, coordinates of maximum thickness point, etc.) vary depending on the series. *Airfoil series* is a group of airfoils which have fixed (the same) coordinates of maximum thickness point and camber though their thickness ratio or relative curvature can vary, alone or simultaneously. For example, airfoils NACA-23012 and NACA-3312 have different camber, and airfoils NACA-22012 and NACA-2412 have different coordinates of maximum thickness point. They belong to different series but to the same family.

For helicopter blades, convexo-convex asymmetrical airfoils are mainly used with slight camber and well-rounded nose. Airfoil thickness ratio $\bar{c} = 8...20\%$ reduces from shank to tip.

The most widely used airfoils are NACA-230, symmetrical NACA-000, high-speed TsAGI П-57-9, etc. Besides that, airfoils can be developed especially for specific helicopter MR blades.

Airfoil HH-02 (Fig. 3.1) was developed in order to optimize $C_{y\,max}$ at $M = 0.4$: HH-02 has the thickness ratio of 9.5 % and make it possible to reach high values of M without growing drag.

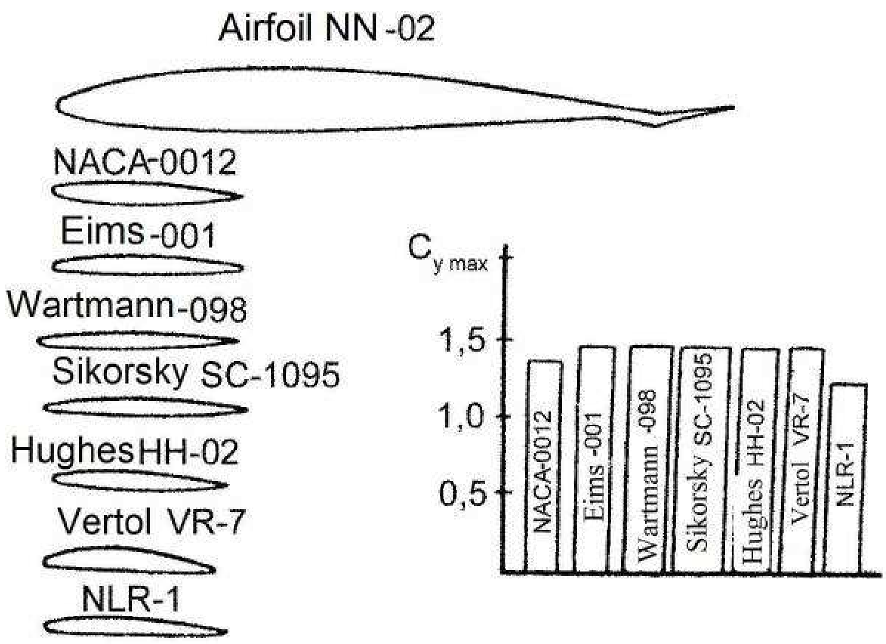


Fig. 3.1. Airfoil Comparative Characteristics for MR Blade

Specialists consider HH-02 airfoil to be rather competitive and possess better characteristics as compared to NACA-0012. This comes from the upward-bent end plate in the airfoil tail section which reduces pitching moment inherent for curved airfoil. At zero blade pitch, CP displaces along the airfoil from its ordinary position at 25% of the chord to 27%. This tolerates reduction of the anti-flutter mass in the blade nose. E.g., AN-64 flies with g-forces $n^{э}_{max} > 3.5$ at high speeds which proves high values of $C_{y max}$ of airfoil HH-02 (Tables. 3.1, 3.2). HH-02 favours the azimuths of the blade retreating as well.

Table 3.1

Basic Aerodynamic Characteristics of Airfoils

Airfoil	α_0	m_{z0}	$C_{x min}$	$C_{y max}$	α_{crit}	M_{crit}	Stall type
NRL-1	-1.0	-0.025	0.0071	1.29	12.3	0.86	Abrupt from TE
NACA –0012	-0.1	-0.007	0.0072	1.33	13.7	0.78	Abrupt from TE
Ames-001	-0.6	-0.005	0.0070	1.46	13.5	0.81	Abrupt from TE
FX-098	-1.3	-0.026	0.0066	1.44	13.0	0.81	Abrupt from TE
HH-02	-0.6	-0.002	0.0066	1.44	13.2	0.80	From TE
SC-1095	-0.9	-0.027	0.0073	1.46	13.5	0.80	Combined
VR-7	-1.6	-0.016	0.0071	1.51	12.5	0.75	From TE

Note: *TE* stands for *trailing edge*.

Table 3.2

Basic Geometry Parameters of Blade Airfoils
(in comparison with NACA-0012)

Airfoil	Thickness Ratio	Maximum Thickness, m	Nose Radius, m
NACA-0012	0.120	0	0.0158
Ames-001	0.103	0.014	0.012
Wartman-098	0.099	0.017	0.007
Sikorsky SC-1095	0.095	0.008	0.008
Hughes HH-02	0.096	0.020	0.008
Helicopter VR–7 (with plate at angle -3^0)	0.119	0.027	0.011
NLR-1	0.086	0.012	0.007
NLR-7371	0.165	0.017	0.055

Airfoil lift-to-drag ratio influences positioning the section along the blade as MR blade aerodynamic characteristics vary greatly along the MR radius (and in helicopter hovering).

In levelled flight, blade sections operate under oblique flow about the main rotor. In this case, flow velocity and the angle of attack over the blade sections vary not only in radius but also in azimuth ψ. The flow becomes unsteady. The blade boundary layer is exposed to centrifugal and coriolis forces. The oncoming flow velocity vector is not perpendicular to the blade leading edge (axis), and blade elements are streamlined at different slip angles.

In order to determine MR aerodynamic characteristics under these conditions, assumptions ("plane section" hypothesis, steadiness) should be made to simplify both the operation of blade section (airfoil) and its calculations.

Finally, airfoil selection is accomplished by optimising aerodynamic and aeroelastic characteristics of the rotor.

At a draft stage of helicopter design, airfoil is selected preliminarily so that to determine MR radial speed $(\omega \boldsymbol{R})$ accounting for limitations in compressibility for the advancing blade and in stall at the retreating blade as well as to assess how MR influence the helicopter mass and characteristics.

When selecting blade airfoil, remember that:

– at the retreating blade in vicinity to azimuth $\psi = 270^0$, airfoil must possess the highest possible value of $\boldsymbol{C}_{y\,max}$ at small values of $\boldsymbol{M} \le 0.4$;

– at the advancing blade in vicinity to azimuth $\psi = 90^0$, airfoil must possess the highest possible value of $\boldsymbol{M}_{crit}$ at small values of $\boldsymbol{C}_y \le 0.2$.

In addition, it is important to provide maximal lift-to-drag ratio of the blade section in vicinity to azimuths $\psi = 0$ and 180° where values of $\boldsymbol{C}_y$ and $\boldsymbol{M}$ are mean ($\boldsymbol{C}_y$ = 0.5...0.7).

Unfortunately, ideal airfoils which meet all the requirements at $\psi = 90°$, $270°$, $0°/180°$ will differ in geometry and thus cannot be manufactured. This means that a compromise solution must be found based on the most important operating conditions for this very helicopter. It is assumed that helicopter operation will be more efficient if its MR characteristics in hovering are improve which will allow increasing carrying capacity without changing power of the power plant. The task to increase MR carrying capacity in horizontal flight is secondary here.

Fig. 3.2 shows M_{crit}-to- C_y relation for NACA-23012 and high-speed airfoil.

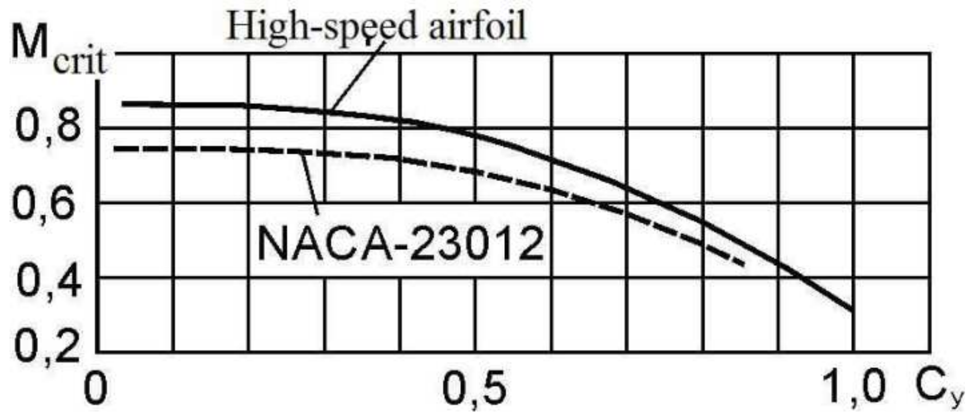


Fig. 3.2. Influence of C_y upon M_{crit}

When selecting ωR, pay attention to $M_{90} \le M_{permis}$, where $M_{permis} \approx M_{crit} + 0.1$. As the compressibility limitation $M_{90} \le M_{permis}$ defines the limit of the sum $(\omega R + V_{max})$, then either radial speed ωR or flight speed V_{max} must be limited.

To reduce adverse influence of compressibility when flying at V_{max}, symmetrical airfoils of small thickness ratio ($\bar{c} = 6...8\%$) as well as special tips are used in blade tip sections, their shapes shown in Fig. 3.3.

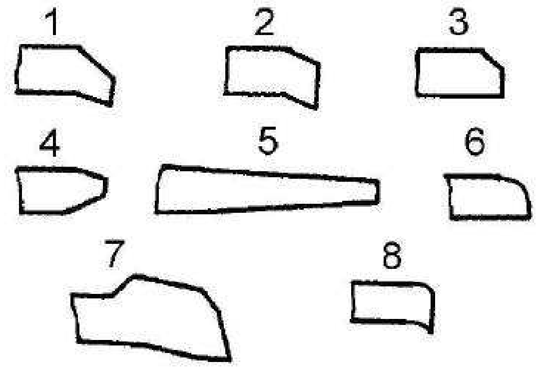


Fig. 3.3. MR High-Speed Blade Tip Shapes
(1 – trapezoid sweep tip; 2 – swept tip; 3 – swept leading edge tip; 4 – short trapezoid tip; 5 – long trapezoid tip; 6 – hyperbolic tip; 7 – BERP swept tip; 8 – sweepforward parabolic tip)

3.3. Determining MR Radial Speed

The value of main rotor (MR) radial speed $\omega \boldsymbol{R}$ has a great influence on helicopter performance. As with airplanes, $\boldsymbol{V}_{max}$ is limited in horizontal flight by the available thrust of the power plant. Additionally, it is limited by air compressibility at the advancing blade and stall at the retreating blade.

In modern helicopters, MR blade tip speed is selected with due attention paid to the facts that in hovering,

– relative efficiency $\eta_o(\eta_o = 0.72...0.75)$ of the main rotor is rather high,

– there is no stall at the retreating blade (coming parallel to the flow),

– there is no compressibility at the advancing blade (coming towards the flow).

180…220 m/s is the acceptable radial speed $\omega \boldsymbol{R}$ for MR blade tip of modern helicopters. According to statistical data for light helicopters, this value goes down to $\omega \boldsymbol{R}$ = 160…180 m/s.

We know that:

1. Helicopter power plants do not provide required power when number of revolutions changes greatly.
2. Modern transmissions do not provide variable degree of reduction.
3. It is reasonable to increase MR thrust by increasing MR pitch φ_0 under reducing its number of revolutions or its minimal permissible value.

Minimal permissible number of revolutions is limited by:

– the margin in stall for high-speed flights;

– the margin of directional control during takeoff/landing and longitudinal-lateral control when flying over route;

– the strength of the main reducer in MR torque;

– the kinetic energy of MR rotation which allows for transition to autorotation;

– AC generators and the whole electrical system of the helicopter.

Maximum permissible number of revolutions is limited by:

– the strength of main reducer, MR hub, swash plate assembly and blades with regard to centrifugal forces;

– the need to prevent shock stall at the tips of the advancing blade when flying at high altitudes and with high speeds;

– the need to provide sufficient MR blade margin in flutter;

– the need to prevent actuation of the engine turbine protection from spin-up that is spontaneous engine stop in flight;

– abrupt rise of vibration level and variable stresses at MR blades due to unsteady nature of shock stall development;

– the need to intensify the helicopter jolting;

– abrupt rise of power required for MR rotation.

The influence of flow stall upon MR is measured by the ratio of MR thrust coefficient to solidity ratio C_T / σ, which defines the averaged blade lifting force coefficient C_{y0}. For a rectangular blade,

$$C_{y0} \approx 3.2 C_T / \sigma . \tag{3.3}$$

When determined for the forward flight by stall, limiting value of C_T / σ at azimuth $\psi = 270^o$ depends on MR operation characteristic μ (Fig. 3.4):

$$\mu = V_{max} \cdot \cos \alpha_{MR} / \omega R , \tag{3.4}$$

where α_{MR} is the angle of attack of MR disk plane (α_{MR} is positive when the disk is tilted forward). In horizontal flight when α_{MR} is small, it is accepted that:

$$\mu = V_{max} / \omega R = \overline{V} . \tag{3.5}$$

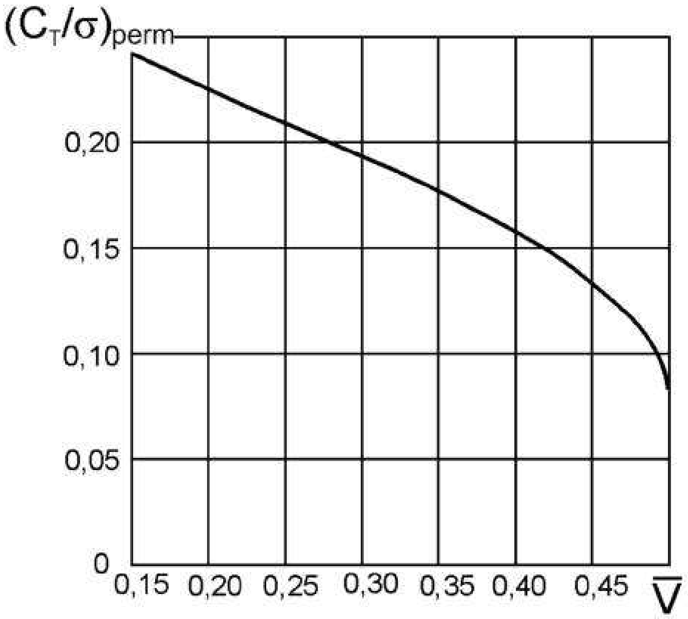


Fig. 3.4. Influence of μ (or $\overline{V}$) upon $(C_T/\sigma)_{perm}$

The growth of μ intensifies the irregularity of angles of attack over MR disk, causes stall at the retreating blade, and reduces C_T / σ. Stall is accompanied with growing vibrations and loads acting upon the rotor and control system due to high variable hinge moments of blades, periodically exposed to stall. That is why stall at the retreating blade often becomes the main factor to limit μ.

The influence of compressibility on the MR characteristics is measured by Mach number M_{90} at the advancing blade tip at azimuth $\psi = 90^o$:

$$M_{90} = \frac{\omega R + V_{max}}{a} , \tag{3.6}$$

where a is the acoustic speed in air.

Quantitatively, the influence of compressibility upon the helicopter characteristics may be determined by the data provided by Alieksieyev [25]. For high-speed airfoil, $M_{90} = M + 0.1$ increment of power required is about 15...18 %; at $M_{90} = M_{crit} + 0.15$, it is about 30 %. M_{crit} is a critical value at which local flow velocity becomes equal to the acoustic speed at least in one airfoil point. When $M_{90} = M_{crit}$, compressibility has practically no influence.

Helicopter designers should remember that number of MR revolutions is limited by air compressibility at azimuth $\psi = 90^o$ and by stall at azimuth $\psi = 270^o$.

When we solve (3.4) and (3.6) in relation to ωR and V_{max}, we get

$$\omega R = a \cdot M_{90} \frac{1}{1+\mu}; \tag{3.7}$$

$$V_{max} = a \cdot M_{90} \frac{\mu}{1+\mu}. \tag{3.8}$$

With the selected airfoil and given V_{max}, we can use (3.7) and (3.8) to calculate ωR and μ, plot $\omega R = f(V_{max}, M_{кр}, \mu)$, and determine ωR and μ by (Fig. 3.5).

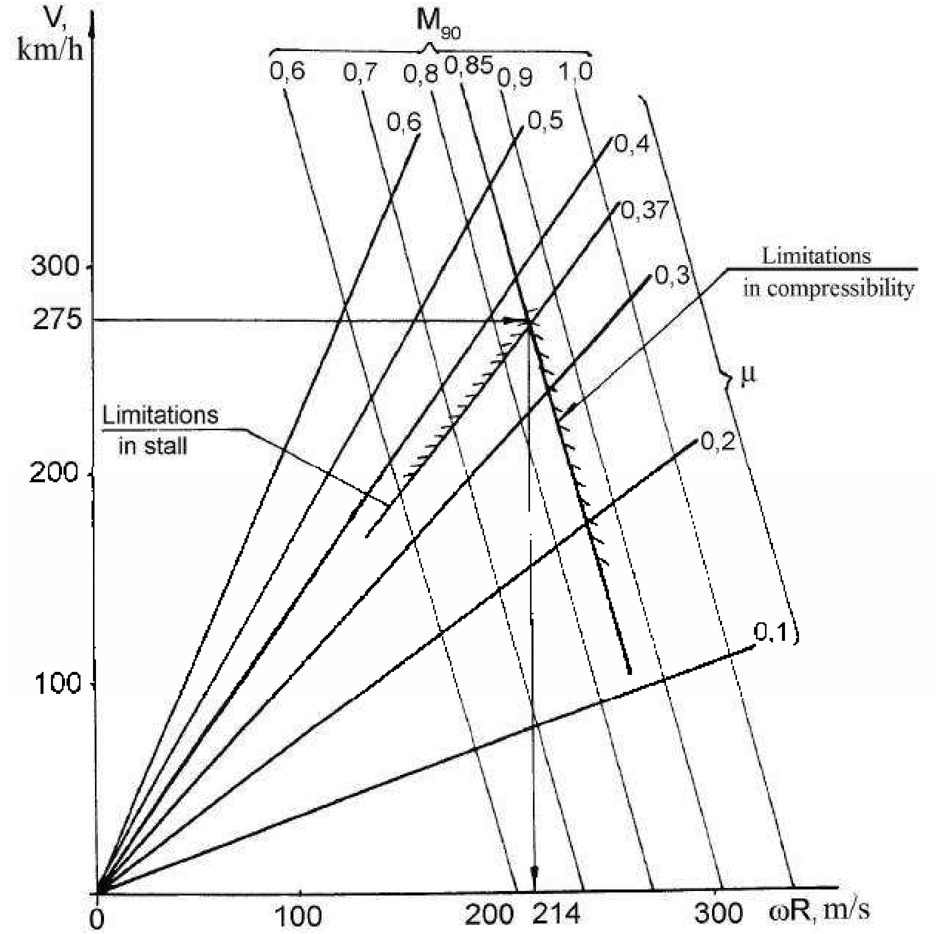


Fig. 3.5. Influence of ωR upon V at $M_{90} = const$ and $\mu = const$

Given recommendations and statistics data allow future helicopter designers to determine MR blade tip speed. For example, for $M_{90}^{perm} = 0.85$ and $V_{max} = 250$ km/h, $\omega R \approx 214$ m/s and $\mu = 0.32$.

3.4. Determining Helicopter Takeoff Mass in Zero Approximation

For a helicopter under design, takeoff mass is in the zero approximation determined as

$$m_0 = \frac{m_{payload} + m_{crew}}{\kappa_{full-load} - \overline{m}_{fuel}}, \tag{3.9}$$

where $m_{payload}$ is the specified payload mass; m_{crew} is a mass of crew members, kg (m_{crew} (1 person) ≈ 80…100 kg); $\kappa_{full-load}$ is the helicopter weight efficiency factor for full load, $\kappa_{full-load} = 1 - \overline{m}_{empty}$; $\overline{m}_{empty} = m_{empty} / m_0$ is a relative mass of thw empty helicopter; $\overline{m}_{fuel} = m_{fuel} / m_0$ is a fuel relative mass. Values of $\overline{m}_{empty}$ and $\overline{m}_{fuel}$ are determined by analysis of the statistical data. The sampling minimum should include five helicopters-prototypes closest in takeoff mass, performance characteristics, and purpose.

3.5. Selecting Varying Specific Loading

Permissible specific loading values of $\mathbf{p_i}$ is determined by specifying values of p_{aver} smaller than (p_1, p_2) and greater than (p_4, p_5): $p_{aver} = p_3 = \sum_{i=1}^{5} p_i / 5$, where 5 is the number of helicopters in statistical sampling (Table 3.3). For light helicopters, pitch of varying p_i can be equal to $\Delta p = 50$ N/m^2, where $p_1 = p_{aver} - \Delta p$, $p_2 = p_{aver} - 2\Delta p$, $p_4 = p_{aver} + \Delta p$, $p_5 = p_{aver} + 2\Delta p$.

Table 3.3

Table for Calculating Helicopter Parameters

<table>
<tr><td rowspan="3">No.</td><td rowspan="3">Formulae for parameters at ωR = m/s</td><td rowspan="3">Unit of measurement</td><td rowspan="3">Number of approximation</td><td colspan="5">Specific Loading p_i, N/m^2</td></tr>
<tr><td>p_1</td><td>p_2</td><td>p_3</td><td>p_4</td><td>p_5</td></tr>
<tr><td colspan="5"></td></tr>
</table>

Beginning from main rotor radius (paragraph 3.7, we recommend you to make calculations using the table includes varying p_i and approximations of $\mathbf{m_0}$ (m_0', m_0'', m_0''', etc.).

3.6. Determining Main Rotor Solidity Ratio

Solidity ratio is a ratio of the total area of all blades to the disk area:

$$\sigma = \frac{F_{blade}}{F} = \frac{Zb_{0.7} \cdot R}{\pi R^2} = \frac{Zb_{0.7}}{\pi R} = \frac{Z}{\pi \lambda_{aver}}, \tag{3.10}$$

where **Z** is a number of blades; $b_{0.7}$ is the blade chord at specific radius $\bar{r} = 0.7$; $\lambda_{aver} = \frac{R}{b_{0.7}}$ is a blade averaged aspect ratio; $\lambda_{aver} = 18.182$.

Main rotor solidity ratio is selected so that to prevent stall at MR blades when flying with maximal speed near ground or at the dynamic ceiling [17]. To meet this requirement, it should be provided that under given flight conditions, the ratio C_T / σ does not exceed the values after which variable loads start growing intensively in the longitudinal flight.

At the dynamic ceiling, flight is executed with economy speed V_{dynam}^{econ}.

When helicopter flies with $\mathbf{V_{max}}$ or at H_{dynam}, its lifting force reduces at azimuth $\psi = 270^o$ due to relatively small difference $(\omega \mathbf{R} - \mathbf{V_{max}})$ in the first case or due to reduction of atmosphere density (rarefaction) in the second case. This is compensated with the increasing blade pitch that may result in stall.

Thus,

$$\sigma_{V\,max} = \frac{C_{T_o V\,max}}{(C_T / \sigma)_{V_{max}}^{add}}; \quad C_{T_o V\,max} = \frac{1.63p}{(\omega R)^2}; \tag{3.11}$$

$$(C_T / \sigma)_{V_{max}}^{add} = 0.297 - 0.36\bar{V}; \quad \bar{V}_{max} = \frac{V_{max}}{3.6\omega R}; \tag{3.12}$$

$$\sigma_{H_{dynam}} = \frac{C_{T\,Hdynam}}{(C_T / \sigma)_{H_{dynam}}^{add}}; \quad C_{T\,Hdynam} = \frac{1.63p}{(\omega R)^2 \cdot \Delta_{H_{dynam}}}; \tag{3.13}$$

$$(C_T / \sigma)_{H_{dynam}}^{add} = 0.297 - 0.36\bar{V}_{H_{dynam}}^{econ}.$$

$$\bar{V}_{H_{dynam}}^{econ} = 0.20 \ldots 0.25,$$

where $\Delta_{H_{dynam}}$ is a relative air density at the flight altitude equal to H_{dynam}.

As a rule, main rotor solidity ratio is chosen by maximum between $\sigma_{V_{max}}$ and σ_{dynam}.

$$\sigma_{min}^{add} = \max\{\sigma_{V_{max}}, \ \sigma_{H_{dynam}}\}. \tag{3.14}$$

To calculate the reliability margin, take $\sigma_{red} = 1.03\sigma_{min}^{add}$.

3.7. Determining Main Rotor Radius

Using specific loading as the varying parameter allows helicopter designers to condition values of MR radius $\mathbf{R_i}$ for the takeoff mass of the helicopter under design $\mathbf{m_o^o}$ corresponding to the specific loading $\mathbf{p_i}$.

$$\mathbf{R_i^{I(II,\,III,\,...)}} = \sqrt{\frac{\mathbf{m_{o\,min}^{I\,(II,\,III,\,...)}g}}{\mathbf{p_i}}}; \tag{3.15}$$

$$\mathbf{D_i^{I(II,\,III,\,...)}} = \mathbf{2R_i^{I(II,\,III,\,...)}}. \tag{3.16}$$

3.8. Determining MR Blade Number

MR blade number is calculated as

$$Z = \pi \cdot \lambda_{aver} \cdot \sigma; \quad \lambda_{aver} \approx 18.182;$$

$$Z^{0,(I,\,II,\,III)} \approx 57.1 \cdot \sigma. \tag{3.17}$$

3.9. Determining Airframe Structure Relative Mass

Airframe structure relative mass is calculate by the formula

$$\bar{m}_{airfr} = \bar{m}_{fus} + \bar{m}_{wing} + \bar{m}_{t-unit} + \bar{m}_{LG} + \bar{m}_{ctl}, \tag{3.18}$$

where $\bar{m}_{fus}$, $\bar{m}_{wing}$, $\bar{m}_{t\text{-}unit}$, $\bar{m}_{LG}$, $\bar{m}_{ctl}$ are relative masses of the fuselage (together with cowlings and tail boom), wing, tail unit, landing gear, and control system correspondingly.

3.9.1. Determining Fuselage Relative Mass

Fuselage relative mass is determined as

$$\bar{m}_{fus} = K_{fus} \frac{S_{\phi}^{0.88}}{\left(m_0^{I,II,III}\right)^{0.75}}, \tag{3.19}$$

where K_{fus} is a fuselage mass coefficient; $\mathbf{m_0^{I,II,III}}$ is a takeoff mass in the first, second, ets. approximations; S_{fus} is the outer surface area of the fuselage.

S_{fus} is accepted on the basis of statistical data (Table 3.4) or calculated by one of the approximate formulae:

$$S_{fus} = \left(S_{f-pl} + S_{f-side}\right)\left(2 - 0.4\frac{S_{f-pl}}{S_{f-side}}\right); \tag{3.20}$$

$$S_{fus} = 2.85 \cdot L_{fus} \cdot \sqrt{S_{mid}}, \tag{3.21}$$

where S_{f-pl}, S_{f-side} are areas of the fuselage plan projection on side view; S_{mid} is a fuselage midsection area; L_{fus} is a fuselage length. The first formula gives more precise results.

Table 3.4

Helicopter Statistical Data

Helicopter	Fuselage mass m_{fus}, kg	Normal takeoff mass $\mathbf{m_o}$, kg	Fuselage relative mass $\overline{m}_{fus}$	Fuselage external area S_{fus}, m^2	Fuselage relative mass coefficient K_{fus}, kg$^{0.75}$/m$^{1.76}$
Mi-1	341	2470	0.138	32	2.29
Mi-2	445	3550	0.125	40	2.23
Mi-34		1300			
R-22	67	621	0.1079	9.6	1.84
Brantly B-2B	82	757	0.108	15	1.44
Brantly m-305	142	1315	0.108	24.9	1.39
B$_o$-105A	227	2100	0.108	24.2	1.78
MBB B$_o^{//}$-105LS	259	2400	0.108	25.4	2.149
Hiller FH-1100, RH-1100	135	1247	0.108	24.2	1.373
Hiller RH-1100	140	1292	0.108	24.2	1.44
Hughes 369	118	1090	0.108	21.7	1.36
OH-6A Hughes 500	125	1157	0.108	21.7	1.42
Hughes 500 D/E	147	1360	0.108	21.7	1.61
Hughes 500 MD	147	1360	0.108	21.7	1.61
Hughes 530 F	152	1409	0.108	22.4	1.61
Anstrom F-28A	105	975	0.108	20.5	1.32
Anstrom F-28F/280 Shark	115	1066	0.108	20.5	1.41
SA-3160, Alhuett III	205	1900	0.108	29	1.6
SA-316B	238	2200	0.108	29	1.79
SA-316C	243	2250	0.108	29	1.82
SA-330C, Puma	691	6400	0.108	69.3	1.85
SA-341G, Gazelj	194	1800	0.108	21.4	2.01
SA-342L	216	2000	0.108	21.4	2.18
AS-350B, Akurey	233	2150	0.108	33.4	1.56

In order to get precise values of S_{fus}, it is advisable to divide the fuselage without superstructures into several subsections, namely nose section, central section, tail section, and tail boom. In this case, nose and tail sections are replaced with semiellipsoids, cones or interposed figures adjacent to the central section. Central section gets the form of a cylinder with the equivalent diameter which is determined based on the equality of midsection and equivalent section perimeters. Tail boom becomes a truncated cone. Such attention to S_{fus} is explained by the essential importance of $\overline{m}_{fus}$.

3.9.2. Determining Tail Unit Relative Mass

In general, tail unit mass is approximately calculated as

$$m_o = q_{t-unit} \cdot S_{t-unit} ,$$

where q_{t-unit} is the specific mass (averaged mass of 1 м^2 of tail unit, kg/m^2); S_{t-unit} is the tail unit area, m^2.

Tail unit specific mass is about 5.6…12.4 kg/m^2. Smaller values of q_{t-unit} are specific for light helicopters.

For single-rotor configuration stabilizers,

$$m_{stab} = C_{stab} \cdot m_o , \tag{3.22}$$

where C_{stab} is the coefficient equal to 0.00136; $\mathbf{m_o}$ is a normal takeoff mass, kg.

Relative mass of horizontal tail is determined by

$$\overline{m}_{h-tail}(p_i) \approx K_{h-tail} \cdot \overline{S}_{h-tail} / p_i , \tag{3.23}$$

where K_{h-tail} is the coefficient of horizontal tail relative mass $K_{h-tail} \approx 131.4$ kg/m·s^2; $\overline{S}_{h-tail} = S_{h-tail} / \pi R^2$.

At the stage of preliminary design, $\overline{S}_{h=tail} \approx 0.004$ (compare: for a Mi-34, $q_{h-tail} \approx 4$ kg/m^2).

3.9.3. Determining Landing Gear Relative Mass

As a rule, landing gear relative mass is estimated in percentage of helicopter takeoff mass

$$\overline{m}_{LG} = K_{LG} , \tag{3.24}$$

where K_{LG} is the coefficient of LG relative mass. For skid-type LG, K_{LG} = 0.01; for non-retractable LG, K_{LG} = 0.02; for retractable LG, K_{LG} = 0.03.

Statistics on foreign helicopters gives different values of K_{LG} (Fig. 3.6) [18].

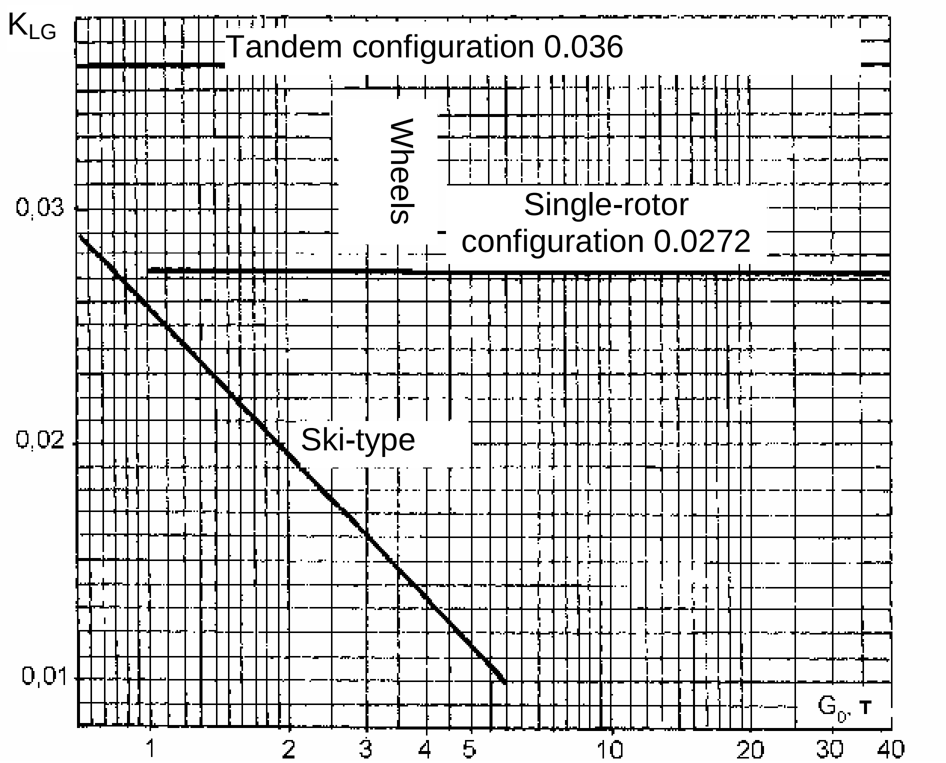


Fig. 3.6. Influence of Coefficient K_{LG}
upon Helicopter Takeoff Mass m_o (Weight G_0)

3.9.4. Determining Control System Relative Mass

When estimating the relative mass of control system, control system is divided into two parts: manual part (control linkage from command sticks to boosters) and booster part (swash plate, boosters with attachment system, control linkage from boosters to swash plate, primary hydraulic system).

Relative mass of the control system is determined by the formula [19]:

$$\overline{m}_{ctl-sys} = \overline{m}_{man-sys} + \overline{m}_{boost-sys}\ , \tag{3.25}$$

where $\overline{m}_{man-sys}$, $\overline{m}_{boost-sys}$ are relative masses of manual and booster systems.

$$\overline{m}_{man-sys} = K_{man-sys}\frac{R}{m_o^o}, \tag{3.26}$$

where $K_{man-sys}$ is the coefficient of manual control mass: $K_{man-sys}$ = 7…10.5 kg/m for transport helicopters without additional control system; $K_{man-sys}$ = 18…25 kg/m for helicopters with additional control for opening cargo doors, stairs, cowls, LG extension, etc.

$$\overline{m}_{boosy-sys}(p) = a_{boost-sys}K_{boost-sys}\frac{\sigma^2}{z\cdot p}, \tag{3.27}$$

where $a_{boost-sys}$ is the coefficient of booster relative control mass, $a_{boost-sys} \approx 30.8 \cdot R,\ m^2 / s^2$; $K_{boost-sys}$ is the coefficient of booster control mass, $K_{boost-sys}$ = 13.2 kg/m^3.

Very often light helicopters are not equipped with hydraulic control system. In this case, mass of all control system elements will depend on all blade hinge moments, sizes, and configurations; number of control sticks; type of control linkage and swash plate. Thus, calculation of control system mass is reduced to calculation of manual control system mass which can be compared with statistical data,

$$K_{man-sys} \approx (7.3...8.2) \text{ kg/m.} \tag{3.28}$$

With light single-rotor helicopters, the mass of control system elements depends on the helicopter's length varying in proportion to blade radius. Since all control linkages are short (except for linkage to tail rotor), the question can be limited to mechanical control linkage only.

3.10. Determining Helicopter Thrust-to-Weight Ratio Required

For modern helicopters, specific flight conditions include [12]:

– hovering at the static ceiling H_{stat} = 1000...1500 m;

– flight at the dynamic ceiling H_{dynam};

– flight with maximal speed $\mathbf{V_{max}}$ = 250…300 km/h at altitude **H** = 500 m;

– continued takeoff with one engine inoperative, when the other operates at contingency rating.

Each design case is characterized by the thrust required for driving MR and TR and meeting power loss. Thrust-to-weight ratio required is determined by the maximum engine power at the specified flight condition.

For all flight modes, power plant thrust is calculated based on specific thrust $\tilde{N} = N_{p-pl} / m_o \cdot g$ required to drive the main rotor under the fligt conditions. Specific thrust is reduced to the thrust at **H** = 0 ($\tilde{\mathbf{N}}_{\mathbf{H}}$) and **V** = 0 ($\tilde{\mathbf{N}}_{\mathbf{V}}$) accounting for the engine throttling degree ($\tilde{N}_{throt}$) in relation to the flight mode and rate of power use ξ:

$$N_{throt\,0} = \frac{\tilde{N}_{p-pl} \cdot m_o \cdot g}{\overline{N}_H \cdot \overline{N}_V \cdot \overline{N}_{throt} \cdot \xi}, \tag{3.29}$$

where 0 is the reduction index; $\overline{N}_H$, $\overline{N}_V$, $\overline{N}_{throt}$ are coefficients for variation of the available engine thrust in relation to altitude, speed, and engine throttling degree:

$$\overline{N}_H = 1 - 0.0695 \cdot H \quad (\mathbf{H}, \text{ km}); \tag{3.30}$$

$$\overline{N}_V = 1 + 5.5 \cdot 10^{-7} \cdot V^2 \quad (\mathbf{V}, \text{ km/hr}). \tag{3.31}$$

When calculating $\bar{N}_H$ and $\bar{N}_V$, helicopter designers use values of **H** and **V** which correspond to the mode under consideration (Table 3.5).

Table 3.5

Helicopter Flight Modes

Flight Parameters	Hovering at Static Ceiling	Flight at Dynamic Ceiling	Level Flight with Maximum Speed	Continued Takeoff with One Engine Inoperative
H, km **V**, km/hr	H_{econ} 0	H_{dynam} V_{dynam}^{econ}	500 V_{max}	0 V_0^{econ}

Rate of power use is a function of speed $\xi = f(\bar{V})$ (Fig. 3.7).

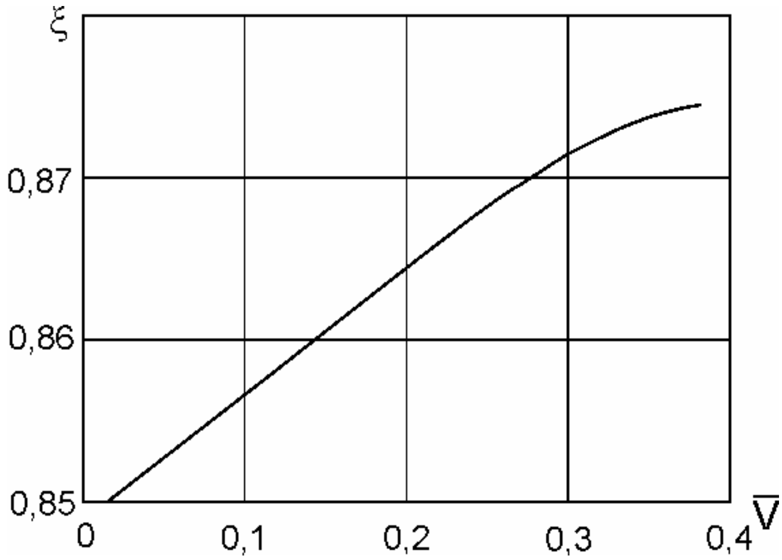


Fig. 3.7. Function $\xi = f(\bar{V})$

In hovering, ξ may be accepted as [19]:

$$\xi = 0.85 \quad \text{at} \quad m_o < 1000 \text{ kg}.$$

At economy flight speed, ξ_{Vecon} = 0.865; at maximum speed, $\xi_{V\,max} = 0.875$; at cruising speed, $\xi_{V\,cruis}$ = 0.872.

Table 3.6 gives the values of constant and variable effective power losses of power plant when it is transferred to main rotor [20]: $\xi = 1 - \sum \bar{\xi}$.

Table 3.6

Power Losses of Power Plant

Loss Characteristics	Kinds of Losses when Transferring Power to Main Rotor	Power Loss Factor $\bar{\xi}$
Constant Power Losses	Main rotor drive:	
	– in hovering;	0.08...0.1
	– in level flight	0.02...0.04
	Drive of engine and helicopter units	0.01
	Friction in transmission	0.03
	Drive of cooling fan	0.015
	Hydraulic resistance of input units	0.025
Temporary power losses	Switching on air heating anti-ice system	0.04
	Dust protection device (DPD):	
	– DPD OFF	0.025
	– DPD ON	0.06

Table 3.7 gives the coefficients of the engine throttling degree at different operation modes.

Table 3.7

Engine Throttling Degree

	Engine Mode of Operation		
	Contingency Rating	Nominal Rating	Cruising Regime
$\bar{N}_{throt}$	$\bar{N}_{conting} \approx 1.07\ldots1.1$	$\bar{N}_{nomim} = 0.9$	$\bar{N}_{cruis} = 0.76\ldots0.81$

3.10.1. Determining Specific Reduced Power Required for Hovering at Static Ceiling

Specific reduced power required for hovering at the static ceiling is calculated by the formula

$$\tilde{N}_{Hstat0} = \frac{\tilde{\bar{N}}_{Hstat}}{\bar{N}_{Hstat} \cdot \xi} = \frac{0.6385 \cdot \bar{T}^{3/2} \cdot \sqrt{p}}{\tilde{\bar{N}}_{Hstat} \cdot \xi_o \cdot \eta_o \cdot \sqrt{\Delta_{Hstat}}}, \text{ W/N,} \qquad (3.32)$$

where η_o is a MR relative efficiency factor in hovering, $\eta_o \approx 0.7$; $\Delta_{H_{stat}}$ is the air relative density at the static ceiling; $\bar{T}$ is a relative value of MR total thrust compensating helicopter weight G_0, fuselage drag $\Delta\bar{T}_{fus}$, horizontal tail drag (stabilizer) $\Delta\bar{T}_{HS}$, and wing drag $\Delta\bar{T}_{wing}$ in the MR flow:

$$\bar{T} = T / m_o \cdot g = 1 + \Delta\bar{T}_{fus} + \Delta\bar{T}_{HS} + \Delta\bar{T}_{wing}, \qquad (3.33)$$

$$\Delta\overline{T}_{fus} \approx 0.238 \cdot \overline{S}_{pl}; \ \Delta\overline{T}_{HS} \approx 1.38 \cdot \overline{S}_{HS}; \ \Delta\overline{T}_{wing} = 0.375 \cdot \overline{S}_{wing} \cdot \overline{\ell}_{wing};$$

where $\overline{S}_{fus-pl} = S_{fus-pl} / \pi \cdot R^2$, $\overline{S}_{HS} = S_{HS} / \pi \cdot R^2$, $\overline{S}_{wing} = S_{wing} / \pi \cdot R^2$; $\overline{S}_{fus-pl}, \overline{S}_{HS}, \overline{S}_{wing}$ are relative areas of projections of the fuselage, horizontal tai, and wing on the horizontal plane accordingly (they are taken from statistical data); $\overline{S}_{fus-pl}, \overline{S}_{HS}, \overline{S}_{wing}$ are areas of projections of the fuselage, horizontal tail, and wing; $\overline{\ell}_{wing} = \ell_{wing} / R$ is a relative wing span. For single-rotor helicopters, we assume $\overline{\mathbf{T}} \approx 1.04$ [19].

3.10.2. Determining Specific Reduced Power Required for Flight at Dynamic Ceiling

According to Vildgube [19], specific reduced power required for flight at the dynamic ceiling can be calculated as

$$\tilde{N}_{Hdynam\,0} = \frac{\tilde{N}_{Hdynam}}{\overline{N}_{no\,min} \cdot \overline{N}_{Hdynam} \cdot \overline{N}_{Vee} \cdot \xi_{Vee}} = \frac{1}{\overline{N}_{no\,min} \cdot \overline{N}_{Hdynam} \cdot \overline{N}_{Vee} \cdot \xi_{Vee}}\, x$$
$$x\left\{16.4 \cdot 10^{-3} \cdot \omega R\left[1 + 7.08 \cdot 10^{-8}\left(V_{dynam}^{econ}\right)^3\right] + 1.82\,\frac{p}{V_{dynam}^{econ} \cdot \Delta_{dynam}} + \right. \tag{3.34}$$
$$\left. + 13.2 \cdot 10^{-3} \cdot \overline{S}_e \left(V_{dynam}^{econ}\right)^3 \cdot \Delta_{dynam}\right\}, \ W / N,$$

where $V_{dynam}^{econ} = 164 \sqrt[4]{\frac{1.09 \cdot p}{\left(\omega R + 11.6 \cdot 10^6 \cdot \overline{S}_e \cdot \Delta_{dynam}\right) \cdot \Delta_{dynam}}}$ is an economy speed at the dynamic ceiling; Δ_{dynam} is the air relative density at the dynamic ceiling; $\overline{S}_e = \frac{\sum C_x}{m_o g}$ is the normalized area of the deleterious plate equivalent to fuselage drag.

For approximate design calculations, drag is calculated by empirical formulae based on statistical analysis (Table 3.8) [5] where m_o is in kilos; and $\sum C_x S$ is in squared meters.

Table 3.8

Helicopter Drag Coefficient

Helicopter type	Helicopter drag
Helicopters with non-retractable LG	$\sum C_x S = 0.018 \cdot m_0^{0.5646}$
Helicopters with retractable LG	$\sum C_x S = 0.0174 \cdot m_0^{0.5364}$
Future helicopters	$\sum C_x S = 0.0102 \cdot m_0^{0.5364}$
Helicopters-airplanes	$\sum C_x S = 0.00601 \cdot m_0^{0.5364}$

Besides, helicopter drag can be found approximately from Fig. 3.8 [21].

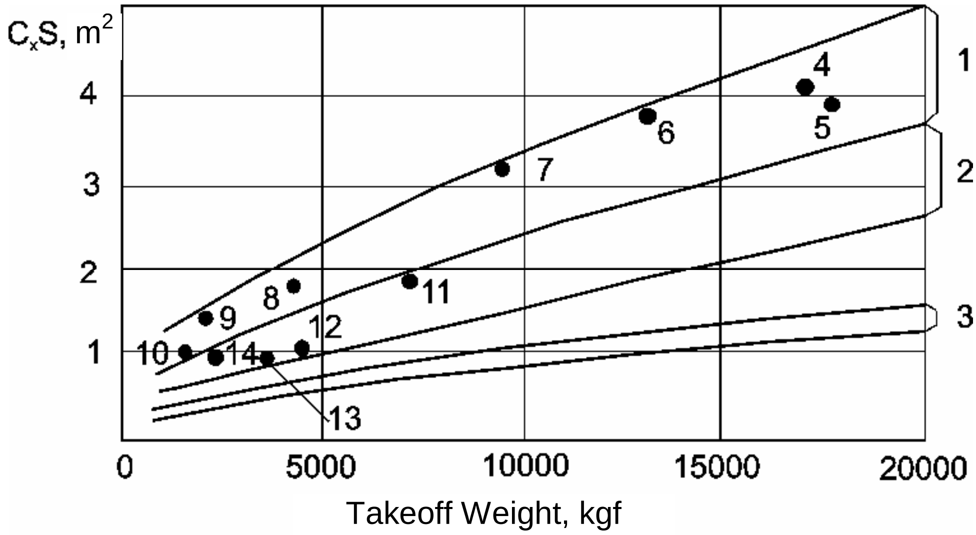


Fig. 3.8. Variation of Plate Area Equivalent to Helicopter Drag in Relation to Takeoff Weight:

1 – helicopters in operation; 2 – new generation helicopters and future helicopters; 3 – airplanes; 4 – Sikorsky SN-53A; 5 – Boeing-Vertol SN-47; 6 – Aerospatiale SA.321; 7 – Sikorsky S-61N; 8 – Bell UH-1B; 9 – Aerospatiale ″Aluett III″; 10 – Aerospatiale SA.341; 11 – Aerospatiale SA.330J; 12 – Sikorsky S-76; 13 – Белл 222; 14 – Augusta A.109

Continued takeoff and flight at the dynamic ceiling are executed with the economy speed V_{ec}. Difference in speeds is insignificant and can be explained by the changed ratio between the induced power and the sum of profile power and motion power.

Vildgube [19] provides us with the induction coefficients I_{induc} in relation to the speed for the horizontal flight of a single-rotor helicopter:

$$I_{induc} = \begin{cases} 1.02 + 0.0004 \cdot V_{max} \ (V_{max} \le 275 \, km/h); \\ 0.58 + 0.002 \cdot V_{max} \ (V_{max} > 275 \, km/h), \end{cases} \tag{3.35}$$

where I_{induc} is the induction coefficient.

Table 3.9 gives the induction coefficient I_{induc} in relation to flight speeds.

Table 3.9

Induction Coefficient

V, km/h	150	200	250	300	350	400
I_{induc}	1.09	1.1	1.12	1.18	1.28	1.38

3.10.3. Determining Specific Reduced Power Required for Horizontal Flight with Maximum Speed

According to Vildgube [19],

$$\tilde{N}_{V\,max\,0} = \frac{\tilde{N}_{V\,max}}{\overline{N}_{H\,V\,max} \cdot \overline{N}_{V\,max} \cdot \xi_{V\,max}} = \frac{1}{\overline{N}_{H\,V\,max} \cdot \overline{N}_{V\,max} \cdot \xi_{V\,max}} \times$$
$$\times \left[16.4 \cdot 10^{-3} \cdot \omega R \left(1 + 7.08 \cdot 10^{-8} \cdot V_{max}^{3}\right) + 1.67 \frac{pI_{induc}}{V_{max}} + \right. \tag{3.36}$$
$$\left. + 13.2 \cdot 10^{-3} \cdot \overline{S}_{eq} V_{max}^{3} \right], \; W/N.$$

3.10.4. Determining Specific Reduced Power Required for Continued Takeoff with One Engine Failed

For a power plant comprising two gas turbine engines [19], specific reduced power required for continued takeoff with one engine inoperative is determined by the formula

$$\tilde{N}_{cont\_takeoff\,0} = \frac{\tilde{N}_{cont-takeoff}}{0.865 \cdot \overline{N}_{Vee}} \cdot \frac{n_{eng}}{n_{eng} - 1} = \frac{1.156}{\overline{N}_{Vee}} \cdot \frac{n_{eng}}{n_{eng} - 1} \times$$
$$\times \left[16.4 \cdot 10^{-3} \cdot \omega R \left(1 + 7.08 \cdot 10^{-8} \left(V_0^{econ}\right)^3\right) + 1.82 \cdot \frac{p}{V_0^{econ}} + \right. \tag{3.37}$$
$$\left. + 13.2 \cdot 10^{-3} \cdot \overline{S}_e \left(V_0^{econ}\right)^3 \right], \; W/N,$$

where $V_0^{econ} = 164 \sqrt[4]{\dfrac{1.09 \cdot p}{\omega R + 11.6 \cdot 10^6 \cdot \overline{S}_{eq}}}$ is an economy flight speed near ground; n_{eng} is a number of engines.

(3.32), (3.33), and (3.34) allow us to study how loading effects $\tilde{N}_{Hst0}$, $\tilde{N}_{Hdynam0}$, $\tilde{N}_{V\max 0}$, and $\tilde{N}_{cont-takeoff\,0}$ within the range of accepted variations of p. Calculations results of $\tilde{N}_{Hst0}$, $\tilde{N}_{Hdynam0}$, $\tilde{N}_{V\max 0}$ and $\tilde{N}_{cont-takeoff\,0}$ are tabulated.

3.10.5. Determining Thrust-to-Weight Ratio Required

Analysis of calculation results has shown that due to helicopter purpose, performance requirements, and other characteristics, diagrams of $\tilde{N}_{Hst0}$, $\tilde{N}_{Hdynam0}$, $\tilde{N}_{V\max 0}$ and $\tilde{N}_{cont-takeoff\,0}$ can be different variants (Fig. 3.9).

In Fig. 3.9, borders of hatched areas correspond to the reliable operation under specific flight conditions,

$$\tilde{N}_{0\,max\,0}(p) = max\left[\tilde{N}_{Hstat\,0}(p),\ \tilde{N}_{Hdynam\,0}(p),\ \tilde{N}_{V\,max\,0}(p),\ \tilde{N}_{cont-takeoff\,0}(p)\right].$$

When selecting helicopter parameters and characteristics, designers use the function $\tilde{\mathbf{N}}_{0\,\max 0}(\mathbf{p})$ which is the design value of $\tilde{\mathbf{N}}$.

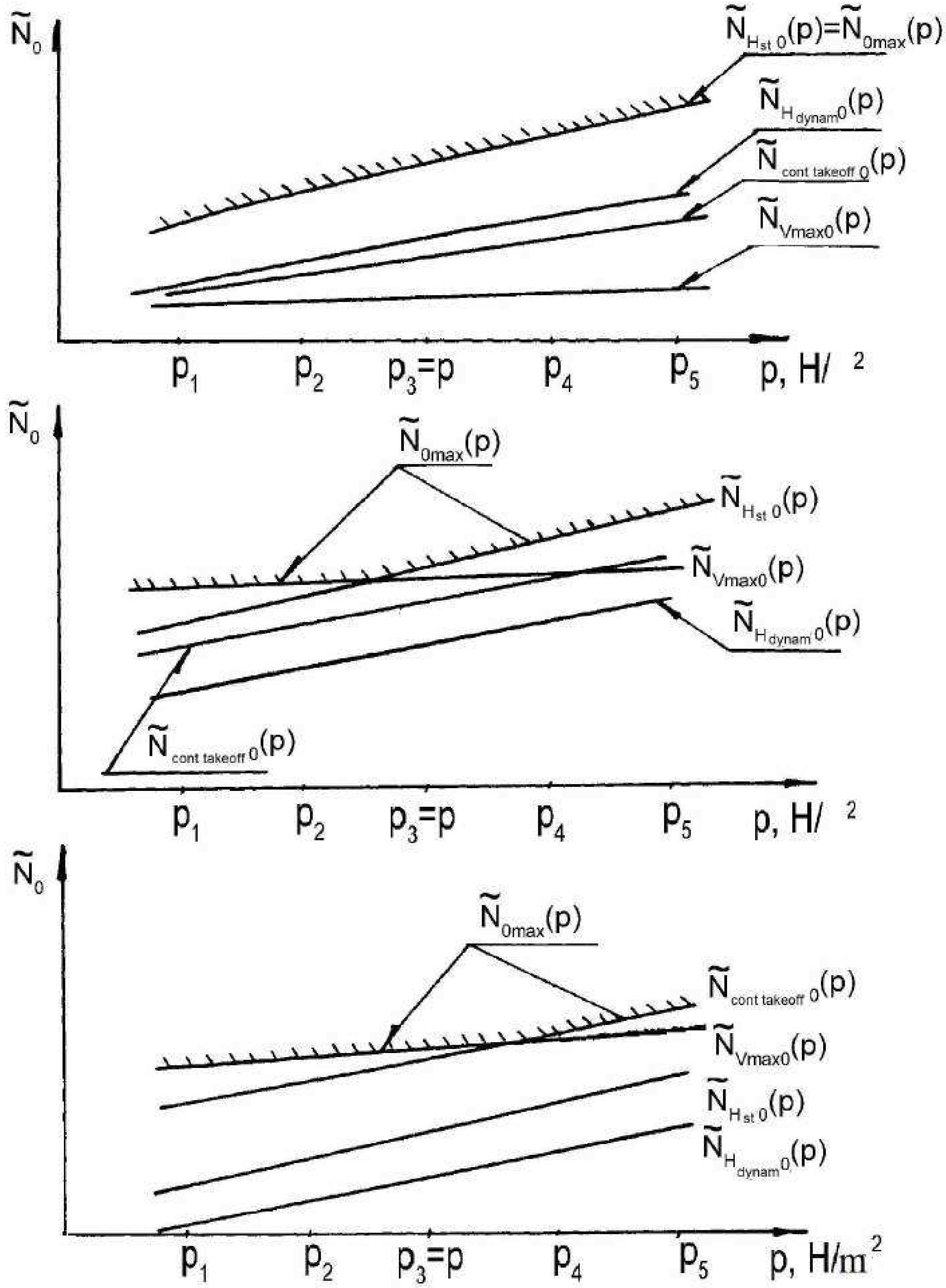


Fig. 3.9. Nature of Functions
$\tilde{N}_{Hst\,0}$, $\tilde{N}_{Hdynam\,0}$, $\tilde{N}_{V\,max\,0}$ and $\tilde{N}_{cont-takeoff\,0}$

3.11. Determining Fuel Relative Mass

When calculating fuel relative mass $\overline{m}_{fuel}$ for a given flight range **L**, we suppose that the helicopter flies at the altitude **H** = 500 m with the cruising speed V_{cruis} at $\omega \mathbf{R} = \mathbf{const}$:

$$\overline{m}_{fuel} = K_{fuel} \frac{C_{e_{cruis}} \cdot L}{V_{cruis}} \left(0.765 \cdot \tilde{N}_{omax}\right) \cdot g \,, \tag{3.38}$$

where K_{fuel} is a coefficient accounting for 5% navigation reserve for transient modes as well as for compensating probable calculation mistakes, $K_{fuel} \approx 1.19$; $C_{e\,cruis}$ is a specific fuel consumption during the cruising stage, kg/kW·h; 0.765 is the coefficient characterizing engine throttling degree for the cruising flight (~ 0.76…0.81).

For low-power piston engines, specific fuel consumption is

$$C_e \approx 1.14 \cdot N_{takeoff}^{-0.3236} \,. \tag{3.39}$$

For gas turbine engines (mounted on light helicopters), specific consumption fuel is

$$C_e \approx 1.94 \cdot N_{takeoff}^{-0.27064} \,. \tag{3.40}$$

For $V_{cruis} \approx 0.8 \cdot V_{max}$, (3.38) gets its final form

$$\overline{m}_{fuel}(p) \approx 1.14 \frac{C_{e\,cruis} L}{V_{max}} \tilde{N}_{0\,max}(p) \cdot g \,, \tag{3.41}$$

$$C_{e\,cruis} = C_{e\,takeoff} \cdot \overline{C}_{e\,H} \cdot \overline{C}_{eV} \cdot \overline{C}_{e\,t} \cdot \overline{C}_{e\,N} \,, \tag{3.42}$$

where $C_{e\,cruis}$, $C_{e\,takeoff}$ is the specific fuel consumption during the cruising and takeoff stages; $\overline{\mathbf{C}}_{\mathbf{e\,H}}$, $\overline{\mathbf{C}}_{\mathbf{e\,V}}$, $\overline{\mathbf{C}}_{\mathbf{e\,t}}$, $\overline{\mathbf{C}}_{\mathbf{e\,N}}$ are coefficients characterizing how specific fuel consumption varies with flight altitude, speed, ambient temperature, and engine throttling degree correspondingly:

$$\overline{C}_{eV} = 1 - 3 \cdot 10^{-7} \cdot V_{cruis}^2; \;\; \overline{C}_{e\,N} = 1.075; \;\; \overline{C}_{e\,H} = 0.995; \;\; \overline{C}_{e\,t=15^0 C} = 1.0 \,.$$

3.12. Determining Relative Mass of Power Plant

According to Bratukhin [13], relative mass of a helicopter power plant has the following form:

$$\overline{m}_{p-pl} = \overline{m}_{eng-sys} + \overline{m}_{rotor} + \overline{m}_{trans} \,, \tag{3.43}$$

where $\overline{m}_{eng-sys}$, $\overline{m}_{rotor}$, $\overline{m}_{trans}$ are relative masses of engines together with their systems, auxiliary power unit (APU), rotors, and transmission.

3.12.1. Determining Relative Masses of Engines Together with Their Systems and Auxiliary Power Unit

Relative masses of engines together with their systems and auxiliary power unit are determined as

$$\overline{m}_{eng-sys}(p) = \left(\gamma_{eng} + K_{sys}\right) \cdot \tilde{N}_{0max} \cdot g + K_{fuel-sys} \cdot \overline{m}_{fuel} + \overline{m}_{APU}, \tag{3.44}$$

where γ_{eng} is the engine specific mass, kg/kW; K_{sys} is a coefficient accounting for mass increase of the power plant due to its systems, namely cooling system, fire protection system, starting system, engine mounts, oil systems and main reducer, oil, kg/kW; $K_{fuel-sys}$ is a coefficient accounting for mass increase of the power plant due to fuel system; $\overline{m}_{APU}$ is the APU relative mass, $\overline{m}_{APU}$ = 0.005…0.008 [13]. Table 3.10 gives the values of γ_{eng} for some piston engines [2.3]:

Table 3.10

Piston Engine Specific Mass

Engine Grade	γ_{eng}, kg/h.p.
VAZ 426	0.667
Lycoming H10-360	0.7
Rotax 914F	0.53
Lom Praha M 332A	0.807
Lom Praha M 337	0.786
Hirt F30	0.484

To approximate individual masses of power plant units incorporating piston engines as well as K_{sys} and $K_{fuel-sys}$, divide the following masses (Table 3.11) by the engine rated brake power [13]:

Table 3.11

Relative Specific Masses of Piston Engine Systems

Power plant systems	K_{sys}, kg/kW
Engine bed with mounts	0.034…0.047
Fan for cooling engine	0.030…0.041
Cowling and deflectors	0.027…0.041
Feeding system	0.058…0.084
Exhaust and suction system	0.020…0.034
Oil ducts and gasoline lines	0.016…0.032
Control system	0.011…0.016
Starting system	0.020…0.045

High values of specific mass correspond to the helicopters fitted with low-power engines, and low values of specific mass correspond to helicopters fitted with rather powerful engines.

For modern helicopters which gas turbine engines develop power **N** ≈ 500…800 kW: γ_{eng} ≈ 0.2…0.24 kg/kW; K_{sys} is about 0.04…0.05 kg/kW [12]; $K_{fuel-sys}$ = 0.07…0.09 for fuel systems with protected fuel tanks; $K_{fuel-sys}$ = 0.06…0.07 for fuel systems with integral fuel tanks [12].

3.12.2. Determining Relative Rotor Mass

Rotor relative mass can be counted as

$$\overline{m}_{rotor} = \overline{m}_{MR} + \overline{m}_{TR}, \tag{3.45}$$

where $\overline{m}_{MR}$, $\overline{m}_{TR}$ are relative masses of main and tail rotors; $\overline{m}_{MR} = \overline{m}_{\Sigma blade} + \overline{m}_{hub}$, $\overline{m}_{TR} = \overline{m}_{\Sigma blade\_TR} + \overline{m}_{hub\_TR}$; $\overline{m}_{\Sigma blade}$, $\overline{m}_{\Sigma blade\_TR}$ is a total relative mass of blades and hubs of both main and tail rotors; $\overline{m}_{hub}$, $\overline{m}_{hub\_TR}$ are relative masses of main and tail rotor hubs:

$$\overline{m}_{\Sigma blade}(P) = a\,\frac{K \cdot \sigma}{\lambda^{0.7} \cdot P}; \tag{3.46}$$

$$\overline{m}_{hub}(p) = a_{hub} \cdot 10^{-5} \cdot K_{hub} \cdot K_z \cdot z \cdot \overline{m}_{blade}^{1.35} (\omega R)^{2.7} \cdot p^{0.35}; \tag{3.47}$$

$$\overline{m}_{\Sigma blade\_TR}(p) = \frac{\sigma_{TR}}{\sigma}\left(\frac{\lambda}{\lambda_{TR}}\right)^{0.7}\left(\frac{R_{TR}}{R}\right)^{2.7} \cdot \overline{m}_{\Sigma blade}; \tag{3.48}$$

$$\overline{m}_{hub\_TR}(p) = \frac{K_{zTR}}{K_z} \cdot \frac{z_{TR}}{z} \cdot \left(\frac{\omega_{TR} \cdot R_{TR}}{\omega \cdot R}\right)^{2.7} \cdot \left(\frac{R_{TR}}{R}\right)^{0.65}\left(\frac{\overline{m}_{blade\_TR}}{\overline{m}_{blade}}\right)^{1.35} \cdot \overline{m}_{hub}, \tag{3.49}$$

where $a_{blade} = 23.62 \cdot R^{0.7}$, $m^{1.7}/s^2$; $a_{hub} = 2.34 / R^{0.65}$, $\mathbf{s^{0.7}}$/m are coefficients of relative masses for MR blades and hub; K_{blade} is a coefficient characterizing structural features of MR blades; σ_{TR} is the tail rotor solidity ratio, $\sigma_{TR} = (1.7...2.3) \cdot \sigma \approx 2\sigma$; λ_{TR} is the tail rotor (TR) blade aspect ratio, $\lambda = \mathbf{z} / \pi\sigma$, $\lambda / \lambda_{TR} = 2z / z_{TR}$; $\overline{m}_{blade} = \dfrac{\overline{m}_{\Sigma blade}}{z}$, $\overline{m}_{blade\_TR} = \dfrac{\overline{m}_{\Sigma blade\_TR}}{z_{TR}}$ are relative masses of MR and TR blades; K_{hub} is a hub coefficient, K_{hub} = 0.0527 kg/kN$^{1.35}$; $\mathbf{K_z}$, K_{zTR} are coefficients accounting for the influence exerted by the blade number of main rotor **z** and tail rotor z_{blade} on the their respective masses; $K_z\,(K_{zTR}) = 1$ when $z\,(z_{TR}) \le 4$, $K_z\,(K_{zTR}) = 1 + 0.05[z(z_{TR}) - 4]$ when $z\,(z_{TR}) > 4$.

For modern blade structures with steel tubular spars or duralumin extruded spars, K_{blade} = 12.6...13.8 kg/m$^{2.7}$; for composite (glass fibre) blades, K_{blade} = 11.5...13.6 kg/m$^{2.7}$ [5].

Approximately, $z_{TR} = 2/3 \cdot z$, $\omega R \approx \omega_{TR} \cdot R_{TR}$.

3.12.3. Determining Relative Transmission Mass

We determine relative transmission mass as follows:

$$\overline{m}_{trans} = \overline{m}_{m-red} + \overline{m}_{int-red} + \overline{m}_{tail-red} + \overline{m}_{tr-shaft}, \tag{3.50}$$

where $\overline{m}_{m-red}$, $\overline{m}_{int-red}$, $\overline{m}_{tail-red}$, $\overline{m}_{tr-shaft}$ are relative masses of main (m-red), intermediate (int-red), tail (tail-red) reducers and transmission shaft (tr-shaft) respectively:

$$\overline{m}_{m-red}(p) = a_{m-red} \cdot K_{m-red} \cdot \xi^{0.8} \left(\frac{\tilde{N}_{0max}}{\omega R} \right)^{0.8} \cdot \frac{1}{p^{0.2}}; \tag{3.51}$$

$$\overline{m}_{int-red}(p) = a_{int-red} \cdot K_{int-red} \cdot (1-\xi)^{0.8} \left(\frac{\tilde{N}_{0max}}{\omega R} \right)^{0.8} \cdot \frac{1}{p^{0.2}}; \tag{3.52}$$

$$\overline{m}_{tail-red}(p) = a_{tail-red} \cdot K_{tail-red} \cdot L_{TR} (1-\xi)^{0.8} \left(\frac{\tilde{N}_{0max} \cdot R_{TR}}{\omega\ R} \right)^{0.8} \cdot \frac{1}{p^{0.2}}; \tag{3.53}$$

$$\overline{m}_{tr-shaft}(p) = a_{tr-shaft} \cdot K_{tr-shaft} \cdot L_{TR} (1-\xi)^{2/3} \left(\frac{\tilde{N}_{0max}}{\omega_{TR}} \right)^{2/3} \cdot \frac{1}{p^{1/3}}, \tag{3.54}$$

where $a_{m-red} = 7.8 \cdot R^{0.4}$, $m^{1.4}/s^2$; $a_{int-red} = 7.8 / R^{0.4}$, $m^{0.6}/s^2$; $a_{tail-red} = 7.8 / R^{0.4}$, $m^{1/3}/s^2$; $a_{tr-shaft} = 6.7 / R^{2/3}$, $m^{1/3}/s^2$ are coefficients of relative mass for main (m-red), intermediate (int-red), tail (tail-red) reducers and transmission shaft (tr-shaft) respectively; $K_{m-red} = 0.0748 kg / (N \cdot m)^{0.8}$, $K_{int-red} = 0.137\ kg/(N \cdot m)^{0.8}$, $K_{tail-red} = 0.105\, kg / (N \cdot m)^{0.8}$, $K_{tr-shaft} = 0.0318\ kg^{1/3} \cdot s^4/m^{7/3}$ are mass coefficients formain (m-red), intermediate (int-red), tail (tail-red) reducers and transmission shaft (tr-shaft) of a single-rotor configuration helicopter; ω_{TR} is a transmission shaft angular speed, $\omega_{TR} \approx 314$ 1/s.

Power loss coefficient ξ under varying **p** is accepted for those values $\tilde{\mathbf{N}}_{0\max}(\mathbf{p})$, W/N which are determined by corresponding flight conditions.

3.13. Helicopter Equipment

3.13.1. General

We decided to include this section to the tutorial in order to show future helicopter designers approximate layout of a helicopter. Since helicopters of different weight categories have different takeoff masses and therefore have different equipment, we write hereabout an approximate set of equipment provided an average light, medium, or heavy helicopter.

Helicopter equipment provides the most effective realization of its purpose; secures flight under normal and complicated weather conditions, day or night; make it convenient for operation and overhaul on ground and in air.

Helicopter equipment can be divided into many groups including electrical equipment (EE), instrumentation equipment (AI), radio equipment (RE), oxygen equipment (OE), cargo compartment equipment (CCE), operational items (OI), heating and ventilation equipment (HVE), anti-ice equipment (AIE), external sling equipment (ESE) etc.

Electrical equipment includes current sources (DC and AC generators, accumulators), converters, transformers, switchgear as well as many consumers. In fact, electrical equipment facilitates the operation of all systems and all equipment groups of a modern helicopter.

To give an example: Mi-6 is equipped with a DC generator of 24 kW, AC generator of 180 kW, and its electrical system incorporates more than 1400 consumers.

Instrumentation equipment comprises flying instruments providing control for the helicopter flight. They measure altitude, speed, heading, vertical rate of climb and descend, position with respect to horizon, distance covered, etc. The most important instruments are redundant for the sake of reliability, and the same parameters are measured in different ways to get more precise values. For example, heading is measured with magnetic, gyromagnetic, radio, and celestial compass.

Instruments control the helicopter systems.

Radio equipment encompasses communication equipment used to communicate with ground and other aircraft; radio navigation equipment (radio compass, radioaltimeter, radio-range station, low speed Doppler navigator, etc.); radio means providing radar landing; interphone communication systems for crew members; special radio systems providing for the execution of special flight missions and others.

Oxygen equipment is used to supply crew members with oxygen when at altitudes higher 4000 m. As a rule, helicopter flies at low altitudes so the oxygen equipment is simple and sometimes even omitted.

Cargo compartment equipment varies depending on the precise mission.

Rigging arrangement is used when the helicopter transports cargoes inside the compartment. It includes hoists for loading cargoes and tie-down fittings to attach the cargo to the floor (cables, meshes, shoes and spring braces for wheeled machinery, etc.).

When arranging the cargo inside the compartment, those in charge must strictly observe the instructions on CG position in relation to the cargo weight. If neglected, they lead to the incorrect helicopter CG positioning and bring down the helicopter controllability. Cargoes must be reliably attached to the floor. Otherwise, they could move in flight and break the CG position that would result in dangerous consequences.

Sanitary appliances are employed if the helicopter transports injured persons. They include stretchers arranged in several stages, working places of medical personnel, crockery for the wounded (cans, drinking bowls, vacuum flasks, bed-pans, etc.); lavatory, medical oxygen equipment and others.

Wounded persons must also be arranged in the compartment so as not to affect proper CG position.

Aerial delivery equipment is used while carrying assault parties. It includes seats for paratroopers, cables to be attached to parachute static lines, devices to signal the drop. Paratroopers must leave the helicopter strictly according to special instructions so as not break the CG position.

Rescue equipment provides rescue operations related with distress. It includes hoist and rope-ladder to embark people on board when the helicopter hovers; safety boats; belts; first aid appliances, etc.

Depending on the helicopter purpose, other kinds of equipment can be installed, e.g. fire extinguishing or agricultural equipment.

Passenger helicopter has a passenger compartment equipped with passenger seats, lavatory, thermal-heat insulation, and other facilities providing comfort flight conditions (sufficient temperature, humidity, ventilation).

Operational items provide convenient operation and maintenance on ground and in flight. They include on-board and ground equipment such as: inspection holes, stairs, cowls, tie-down fittings, turbine generator plants for autonomous start, etc. Ground equipment includes tie-down fittings to control the units and systems suspended when the helicopter is in hovering; hydraulic plants to test the helicopter’s hydraulic system; electrical plant to power helicopter and its equipment providing for quick and easy routine maintenance and installation.

3.13.2. General Airworthiness Requirements for Helicopter Equipment

Type and amount of equipment is defined by the helicopter purpose, self-sufficiency, operating conditions, reliability, special tasks and other factors.

Helicopter equipment is a combination of complexes which concentrate data coming from multiple sensors to generalized indicators and instruments representing environment and helicopter attitude. It secures automated control for helicopter and its power plant in air for all flight stages under all weather conditions (including the most severe of them). Designers take no rest to develop new equipment which would be of minimal mass providing reduced share in helicopter total mass including built-in radar antenna with radomes, de-icers, and equipment bays accessible outside the helicopter.

When developing pilot project, it is necessary to select the necessary equipment, arrange it reasonably and incorporate into the helicopter structure observing the principle of minimal mass; keeping aerodynamic shape; meeting the requirements for reliability and redundancy.

Despite variety of helicopter operating conditions, equipment layout is based on the same principles with only slight variation.

When laying out the equipment in the flight compartment, it is necessary to reach maximum possible comfort for flight crew, to provide good visibility at any stage of flight including takeoff and landing, to arrange instruments and control panels so that they are accessible from the work stations with tightened seat belts.

Controls must follow the usual order.

Hygiene and sanitary conditions (temperature, air composition and humidity, pressure, noise level, ventilation, air speed) must answer accepted norms. Flight compartment must be protected against the sun light by blinds and light filters. Pilot seats must have two positions, for work and for rest. Sharp edges must be covered with soft lining. Floor must be covered with carpets with electrical heating. Compartments and panels nesting the equipment must be painted in neutral colours in order not to tire eyes of the crew. Compartment must be illuminated with white and coloured light. Flight documentations need proper placement as well as food containers, vacuum flasks, first aid kits, lavatory, and paraphernalia.

Special technical bays are organized inside the fuselage so as to improve ground maintenance, reject heat, reduce in different places mass of wiring, pressurize without interfering with loading/unloading procedures.

Proper layout of technical bays secures easy entrance to and exit out of the bay no matter whether the helicopter is loaded or not; convenient ground maintenance; easy installation or dismantling, connection to test equipment. It must also provide night-time illumination and interphone communication terminal as well as heat rejection from technical bays, air circulation, cooling, and ventilation of compartment.

All equipment used for routing engineering facilities must guarantee shock-absorption and be well-bonded. By bonding, we understand interconnecting all individual movable and immovable units as well as all equipment elements with the helicopter body using wire jumpers, metal buses, or metal (shielding) braids of minimal resistance. All connection places must be easily accessible for inspection.

Based on draft design, arrangement, mock-up, thoughtful installation of equipment in load-bearing set of airframe allows it to essentially reduce mass of the structure, keep aerodynamic shape of the helicopter, enhance its performance characteristics.

At the stage of preliminary design, when helicopter takeoff mass is calculated in the first approximation, it includes the calculation of equipment mass as well. Here absolute dimensions, lengths of joining elements, completed equipment complex are not fully defined, and calculation is performed based on approximate statistical dependences for different types of helicopters. At a later stage, equipment complexes and their layout in helicopter are developed and centre of gravity as well as equipment mass are precisely determined.

Refer to Danilov [28] for detailed description of equipment characteristics.

All kinds of on-board equipment must meet a number of requirements, necessarily including flight and navigation equipment together with the instruments monitoring power plant; navigation and landing radars; radio communication; electrical; lighting.

On-board equipment must be designed, manufactured, and installed so as to perform main functions at the required level of reliability and efficiency under expected operating conditions and to exclude failures, malfunctions, and emergency situation such as fire, smoke, odour or toxic gases, etc. On-board equipment and its components must be approved for this precise helicopter.

Equipment which consumes electrical power from the centralized electrical system must be designed, manufactured, and installed so as to function normally provided the high-quality electrical power. Failure or malfunction of any consumer must not cut electrical supply of other consumers.

Equipment controls must be designed so as to a) secure correct operation at any usual and arbitrary positions and b) prevent any damage to this or another equipment in case the normal sequence of working operations was broken.

Controls which can cause emergencies must be arranged so that no one can switch them on/off inadvertently when in flight. Blocking devices in a form of caps, latches, etc. can be used to prevent their unintentional switching. At the same time, these blocking devices must not impede to use these controls in proper situations.

Controls and regulators which are not used in flight must be inaccessible for crew members.

Equipment must be fitted with built-in testing features or secure easy external testing and regulation on the ground.

Equipment layout inside the helicopter must provide optimal operation of both the equipment itself and flight crew as well as secure easy ground servicing. If used by crew members in flight, operative equipment controls must be placed on the every work station so that they are clearly visible, their scales and marking legible, and working operations executable without standing up or moving.

Helicopter equipment must be perfectly compatible. Designers must pay great attention to prevent equipment, cables, and wires from any damage coming from baggage, cargo, transport facilities, maintenance personnel, or passengers. At the same time, equipment, cables, and wires must not become a source of burns, electrical shock, or any other kind of trauma.

Equipment must be thoroughly tested on ground and in flight, with all the test results confirmed.

Equipment must be properly marked for clear identification and supplied with necessary documentation so that to provide adequate operation, servicing, repair, storage, and transportation.

3.13.3. Approximate Helicopter Equipment

Table 3.12

Helicopter Equipment Approximate Arrangement

No.	Equipment of Helicopter and Its Variants	Components of Equipment		
		light helicopter	medium helicopter	heavy helicopter
		Place of Installation		
1	2	3	4	5
1	Electrical equipment: – DC; – AC; – 400 Hz AC; – converters, transformers, switchgear, etc.; – storage battery bays.	in CS* in NS*	in CS* in NS*	in CS* in NS*
2	Aircarft instrumentation: – instruments including engine monitoring, transmission control, and control system; – instruments monitoring the helicopter systems	in NS	in NS	in NS
3	Radio equipment: – communication with ground and other aircraft; – radio navigation (radio compass, radio altimeter, radio distance measuring equipment, Doppler computer); – ILS; – interphone communication, etc.	in CS	in CS	in CS
4	Oxygen equipment	in CS	in CS	in CS
5	Equipment arranged in cargo compartments, depending on fulfilled tasks:			
5a	Rigging arrangement: – hoists to loading cargoes; – tie-down devices to secure cargoes position on the floor (nets, shoes, braces, etc.)	—	in CS	in CS
5b	Fire extinguishing equipment: – glass-fibre-reinforced plastic tanks with protecting devices for fire extinguishants; – equipment supplying fire extinguishants	—	in CS	in CS

Table 3.12 (Continued)

1	2	3	4	5
5c	Medical equipment: – stretchers arranged in several tiers; – work places of medical personnel; – dishes for servicing the wounded; – lavatory; – oxygen equipment	—	in CS	in CS
5d	Aerial delivery: – seats for paratroopers; – cables to attach parachute halyards; – signalling devices	—	in CS	in CS
5e	Rescue equipment: – hoist and rope ladders to lift people when in hovering; – medical facilities for victims	in CS	in CS	—
5f	Agricultural equipment: – spraying system; – pollination system	in CS	in CS	—
5g	Equipment for external cargo transportation: – external suspensions	—	in CS	in CS
6	Operational items: – inspection holes, stairs, cowls, tie-down fittings, turbine generator for autonomous engine start	—	in CS	in CS

Note: *NS*, *CS*, *TB* stand for *nose section*, *central section*, and tail boom.

After processing the statistical data by the technique of least squares, we obtained regression dependence

$$m_{equip} \approx 0.017 \cdot m_o^{1.2235}, \qquad (3.55)$$

where m_{equip}, m_o are the mass of equipment and takeoff mass of current approximation, kg.

(3.56) allows helicopter designers to determine m_{equip} for helicopters with takeoff mass $m_o = 2000...6000$ kg with acceptable error level.

3.14. Calculating Takeoff Mass of Helicopter and its Units under Varying Disk Loading

In order to select helicopter parameters and determine its masses, designers start with calculating helicopter takeoff mass in the first approximation. After calculating takeoff mass of five helicopters, they compare the obtained results to accept $\mathbf{m'_{o\,min}}$. This value is used to for calculations in the second approximation. Designers follow this algorithm to continue calculations until the fifth approximation and then plot the diagrams (Fig. 3.9) of dependences $m_0^i = f(p_i)$, where i = 1, 2, …, 5. Using these diagrams, they find out the values of $\mathbf{m_{o\,min}}$ as well as MR disk loading $p*$.

Depending on the given performance requirements, function $m_0'(p)$ can behave in three ways (Fig. 3.10):

– have a clear minimum;

– decrease monotonously;

– increase monotonously.

For the first variant, the solution is evident: the minimum of takeoff mass defines the value of optimal (reasonable) specific disk loading $\mathbf{p}*$.

Solutions for variants 2 and 3 require additional conditions, e.g. be limited by minimal permissible $p_{min\,perm}$ and extreme p_{extrem} disk loading. In these cases, you assume that $p \le p_{extrem}$ for $m_{0\,min}^I$ monotonously decreasing and $p \ge p_{min\,perm}$ - for $m_{0\,min}^I$ monotonously increasing.

Limitations by minimal permissible values of disk loading $p_{min\,perm}$ under selected MR solidity ratio (blade area) the conditions of permissible blade deflection **h** while parking, allowed blade aspect ratio λ and blade mass characteristic γ_0, may be set using data given in publication [5].

You determine the sufficient number of approximations by comparing the values of ξ and initial ξ_3 using the equation

$$\left(m_0''',\,''' - m_0',\,''\right) / \, m_0''',\,''' = \xi \le \xi_3 \, . \qquad (3.56)$$

If the condition (3.56) is not met, then increase the number of approximations.

Calculated values are tabulated (Table 3.3) and plotted as a set of diagrams (Fig. 3.11).

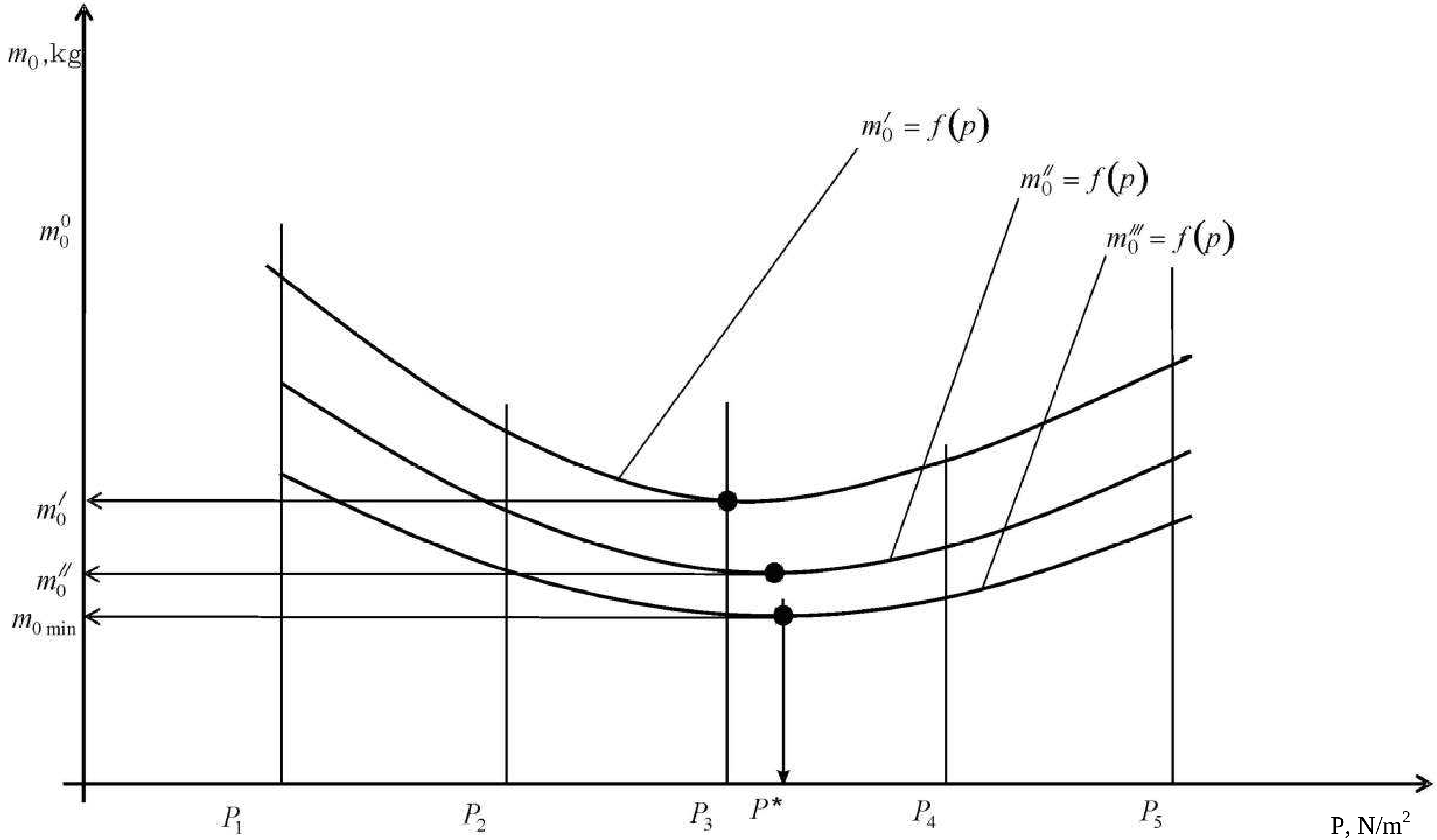


Fig. 3.10. Determining **p*** by the Value of ***$m_{0\,min}$*** for the Helicopter under Design

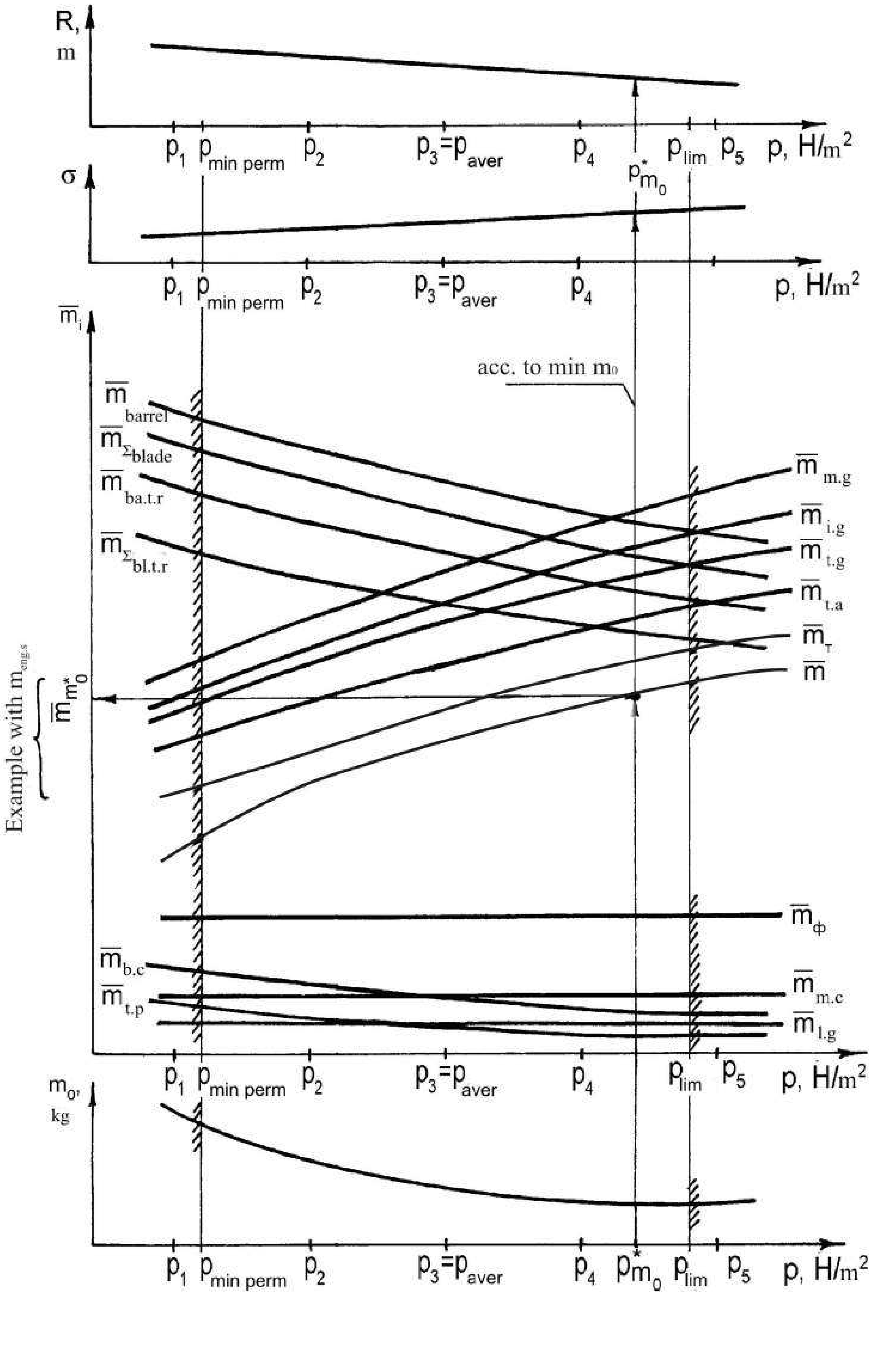


Fig. 3.11. $m_0 = f(p)$ and $\overline{m}_{unit} = f(p)$ Dependences

3.15. Calculating Maximal Permissible MR Radius of Helicopter under Design

To meet the condition of (3.56), helicopter designer must check the limit of maximum blade radius $R_{y\,max\;perm}$ corresponding to permissible blade tip deflection y_{Rperm}.

When developing helicopters of any configuration and weight category, you must account for these limitations and assume the main rotor radius $R_{y\,max\;perm}$ at the most.

In many cases, the limitation comes from the layout of the disk loading **p** corresponding to maximal permissible blade tip deflection $y_{Rperm} = 0.12 \cdot R$ which does not allow you to increase blade radius more than $R_{y\,max\;perm}$.

Radius R should not exceed $R_{y\,max\;perm}$, that is $R \le R_{y\,max\;perm}$. If $R > R_{y\,max\;perm}$, then R is accepted as equal to $R_{y\,max\;perm}$. Remember that $R_{y\,max\;perm}$ is calculated for $m_{0\,min}$.

Maximal blade radius limited by the value of its relative permissible deflection $\bar{y}_{Rperm} = y_{Rperm} / R = 0.12$ is calculated as [12]

$$R_{y\,max\;perm} = 0.0585 \left[\frac{\bar{y}_{Rperm}}{K_{y\,R} \cdot K_{л}} \right]^{0,189} \cdot \left[\frac{m^{I,II}_{0\,min} \cdot g}{\bar{t} \cdot z} \right]^{0.435}, \qquad (3.57)$$

where K_{yR} is the coefficient characterizing the perfection of the blade layout by the value of minimal (provided) deflection: K_{blade} is the coefficient calculated by (3.58); $\bar{t}$ is a relative parameter; $\bar{t} = \frac{t_{y0}(\omega R)^2}{7502}$, t_{y0} is the MR thrust coefficient at H = 0 and ωR = 220 m/s, $t_{y0} = C_T / \sigma$ (Fig. 3.6).

K_{yR} depends on the blade structure, namely spar material, distribution of inertia moments and linear mass over blade length. Approximate values of K_{yR} are:

– $K_{yR} \approx 0.353 \cdot 10^{-6}$ m^2/kgs for blades with duralumin extruded spar;

– $K_{y\,R} \approx 0.388 \cdot 10^{-6}$ m^2/kgs for blades made of glass fiber (according to statistics).

$$p_{min\;perm} = \frac{m_0^I \cdot g}{\pi\, R^2_{y\,max\;perm}}. \qquad (3.58)$$

$R_{y\,max\;perm}$ and other parameters included in (3.58) come from Fig. 3.12.

In general, given conditions are definitive for the determination of MR radius for medium and heavy helicopters. Moreover, they are provided without special effort (often automatically) for light and superlight helicopters with metal MR blades and hubs. Composite blades (i.e. made of fiber glass) possess less rigidity in bending due to essentially (2-3 times) lower module of elasticity of a blade in tension/compression and because of high material flexibility in bending due to interlaminar shear of glass fiber layers.

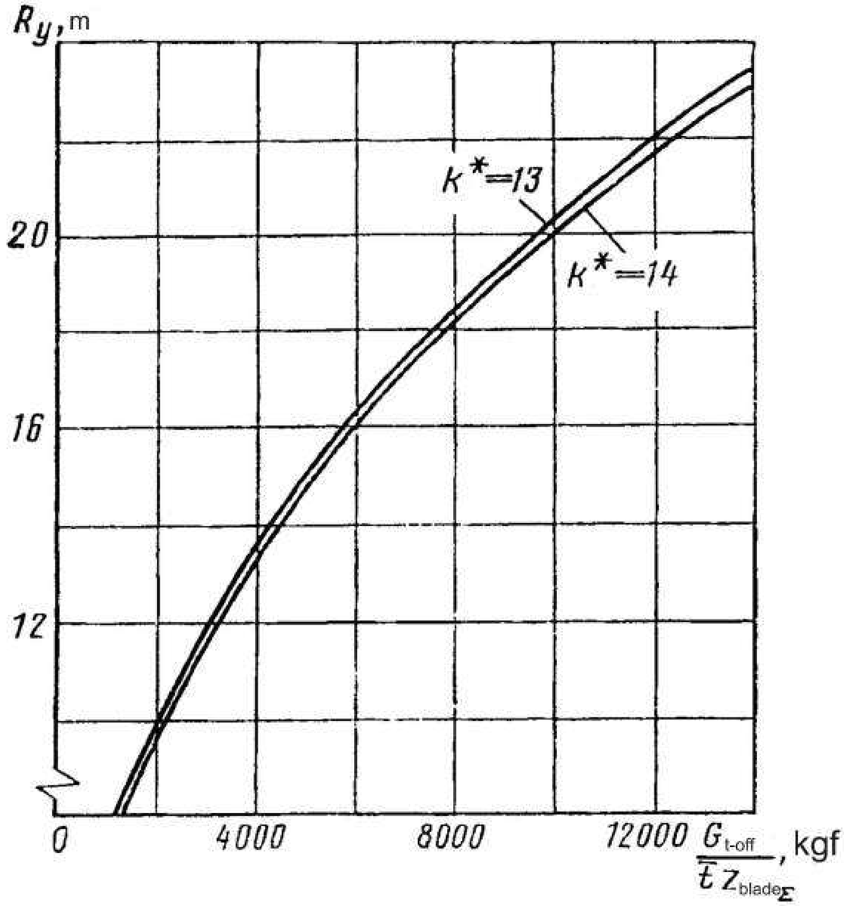


Fig. 3.12. Maximum Permissible MR Radius according to Droop Value

3.16. Selecting Engine

3.16.1. General Issues

Modern helicopters with mechanical MR drive power plants are usually equipped with engines of two types: piston engines and gas turbines.

Helicopter power plants consist of:

1) engine(s) and power plant systems;
2) external engine fuel systems (tanks, booster pumps, piping, valve-filters, filling units);
3) cooling systems (for piston engines);
4) air suction systems (together with dust protection devices);
5) exhaust systems (together with screen exhaust unit);
6) engine control systems;
7) engine mounts and cowlings;
8) start-up systems.

When designed and chosen for helicopter operation, power plants must meet the following requirements:

1) provide convenient access;
2) prove easy for installation/dismantling;
3) achieve effective cooling under any weather and flight conditions, allow for necessary adjustments with cooling system; sustain low power consumption;
4) prove easy for repair and manufacturing;
5) achieve low specific weight;
6) provide good shock absorption (vibrations to be absorbed directly inside the power plant but not transferred to other units);
7) secure necessary strength of load-bearing members for all loading cases according to the strength standards.

Distinguishing features of a power plant are defined by its scheme, layout, type, and engine design.

When selecting the engine, helicopter designers must account for the following factors:

– specific weight γ_{eng}, kg/KW;
– specific fuel consumption $\mathbf{C_e}$, kg_{fuel}/KW h;
– reliability and durability;
– convenience in operation;
– engine balance and uniformity of torque;
– acceleration capability, sec;
– possible start without loading of resistance and rotating masses playing role of flywheel;
– cost.

Helicopter power plant transforms energy of combusted fuel into mechanical work of rotation for main and tail rotors as well as uses it to drive helicopter units and systems. Most of the power coming power plant (up to 85...90 %) is consumed for driving main rotor.

Power plants can be equipped with either mechanical or jet-type drives. We do not cover jet drives here because they are not applied.

Mechanical drives need transmission system to transmit rotation from the engine to the main rotor and other helicopter units.

First helicopters with gas turbine engines used to have shorter range and flight duration because of essential fuel consumption. Modern gas turbines provide the level of affordability comparable with that of piston engines that is why the characteristics of flight range and duration has become essentially better today.

Low engine mass and specific consumption are especially important for helicopters as their thrust-to-weight ratio is higher and their power plants are heavier in comparison to airplanes. With that, flight range is essentially less than that of an airplane because of lower lift-to-drag ratio of the carrying system. For example, lift-to-drag ratio of Sikosky S-65 is K = 4.5; lift-to-drag ratio of Boeing-Vertol V-114 is K = 3.9. Modern subsonic airplanes have lift-to-drag ratio K = 15…18, and supersonic airplanes have K = 8...12.

For another thing, modern helicopters must have a high static ceiling which means that they need engines providing higher engine critical altitude and higher power.

If helicopter designer fail when selecting a proper engine, this would results in the helicopter seasonality; e.g., this helicopter will be able to hover and climb vertically only in winter.

3.16.2. Power Plants with Piston Engines

Up to 1950s, piston engines were the main type of power drive as gas turbine engines excel piston ones were at the very beginning of their development. Today gas turbine engines excel piston in specific mass, unit power of one, and certain other characteristics.

At present, piston engines are used mainly in small helicopters which is conditioned by several reasons:

– absence of serial gas turbine engines of relatively low power (gas turbine engines with power less than 250 horse power are practically absent);

– relatively low power transmitted by transmission system;

– relatively low reduction degree due to low number of revolutions that make it possible to use drive belting.

Distinguishing feature of power plants equipped with piston engines include availability of cooling system with a fan; master clutch and freewheel clutch which can be combined to form one unit; transmission with hinged and elastic couplings of shafts and reducers.

In most cases, piston engines are used with air cooling systems, but sometimes water cooling systems are provided.

Engine power produced by a power plant with mechanical transmission system feeds not only main rotor but other helicopter units and systems as well: ~5 % drive the fan; ~3...5 % go to overcome friction in reducers and transmission; about 7...8 % are consumed to drive the tail rotor in hovering, and 3...4 % – in cruising flight. Total efficiency of the drive system of a single-rotor helicopter is about 0.82...0.89.

To provide required number of MR revolutions, total gear ratio of reducers equals 7...14 for power drives of helicopters with mechanical transmission.

If a helicopter uses a piston engine, it is desirable for this engine to be multicylinder with the lowest degree of non-uniformity of torque in order to reduce shifting jerks and vibrations of the transmission.

If it is necessary to analyse loading caused by engine torque, then operational value of the torque is assumed to be equal to

$$M_{trq}^{oper} = k \cdot M_{trq}^{aver}, \qquad (3.59)$$

where M_{trq}^{aver} is an averaged value of engine torque at preset power; k is a corrective coefficient depending on the engine type used (Table 3.13).

Table 3.13

Coefficient k in Relation to Engine Type

Engine Type	Number of Cylinders	k
Piston Engine	2	4
	3	3
	4	2
	≥ 5	1.5
Gas Turbine Engine	—	1.25

Engine design with a vertical crankshaft is preferable as it reduces the mass of a power plant due to simplifying helicopter transmission. At the same time, vertical crankshafts have a number of drawbacks including bad lubrication, complicated cooling fan system and tail rotor drive, uneasy CG positioning, etc.

Transmission and reducers constitute an essential part of power plant total mass. According to statistical data on light (up to 2000 kg) helicopters, their transmission mass amounts to about 25...50 % of the engine mass. For helicopters with the takeoff mass exceeding 5000 kg, transmission mass is approximately equal to the mass of the engine.

Besides transmission, engine mass includes masses of all units attached to it and required for its operation as well as mass of the grease inside the crank case. Mass of the engine mount system (engine frame) is taken accounting for the engine suspension dampers.

The engine control system includes control panel in the flight compartment and all control system links upstream the engine.

Cooling system includes fan, cowlings, heat exchanger (for liquid cooling system), and executive device which regulates the cooling process. Its mass includes masses of the following units: oil tanks, oil coolers, fittings, oil piping, drainage facilities all of them together with attachment fittings and oil mass in the lubricating system.

At the stage of draft design, mass of a power plant, its parts and units can be selected based on statistical data for similar power plants used on airplanes. Moreover, power plants usually have ready-made units and parts with known masses which helps us to calculate their mass at the high level of precision.

As is evident, essential disadvantages of power drive include high mass; design and manufacturing complexity; a lot of time needed for service and ground maintenance under winter conditions. Low fuel consumption is the advantage.

It is unreasonable to mount two engines on a helicopter with low mass and loading less than **p** $\approx$ 200 N/m^2 as the specific mass and fuel consumption of low-power engines is less in comparison with double-power engine. Moreover, with two engines, power plant system becomes more complicated and its cost increases.

3.16.3. Power Plants with Turboshaft Engines

Today, light helicopters are mainly equipped with double-shaft (with a free turbine) turbine engines are used which, as compared to piston engines, have smaller overall dimensions and less mass, are simpler and more reliable in operation, possess good stability under slight (within 10...12 %) variations of turbine number of revolutions.

If we compare turboshaft helicopter engines with gas turbine engines used for airplanes, we shall find out that in addition to gas turbine placed on the same shaft with the compressor, they have the second (free) turbine placed behind (downstream gas flow). The free turbine has no mechanical link to the compressor shaft and the first (gas) turbine. Output shaft of the free turbine is linked to the main rotor by means of transmission system and reducer. When used to drive the main rotor, free turbine secures the best number of MR revolutions irrespective of the engine compressor's angular speed; achieves optimal fuel consumption under different engine operating conditions; facilitates spin-up of the turbine compressor at the engine start; allows us to go without combined master clutch in the helicopter power plant, etc.

Gas turbine engines

– increase weight efficiency of helicopters;
– provide for growing cruising speeds;
– raise power plant relative power;
– simplify operation, enhance reliability (e.g. of the engine start at low temperatures);
– make it possible to install two engines,
– enhance flight safety.

Moreover, these engines are good for low engine specific mass; low power consumption for engine cooling (they need no special cooling system); simple operation, especially under low temperatures of ambient air; relatively low vibration due to high balance of engine rotors; enhanced aerodynamics (small engine midsection).

Their disadvantages include more complex reducers with greater mass due to high reduction factor for the torque transmitted to main rotor; relatively long acceleration time (up to 15 s) which worsens helicopter manoeuvrability;

dependence on environmental conditions (air dustiness, moisture, temperature, etc.).

Turboshaft engines must satisfy many requirements, namely reliability and durability; high economic efficiency; reliable start; specified acceleration time; low level of noise and vibration; low mass and small dimensions; simple and convenient operation and maintenance; low production and repair costs.

Vilensky and Strukov [23] extend this list of requirements to include:

– longer service life under severe conditions;

– stable operation under conditions close to maximum power, during long periods of time, and under complicated weather conditions (rain, snow) at low altitudes;

– protection from destructive factors (for special purpose helicopters);

– simple operation with no highly qualified technical staff required;

– survivability and flight safety if one engine fails (this means that n_{eng} = 2 engines are required for a power plant).

In comparison to piston ones, turboshaft engines are better due to their:

– low specific weight (2.5 times less than that of piston engines);

– possible use of little thrust from exhaust gases to increase maximum flight speed (~2 %);

– no special engine cooling system;

– high reliability (less failures);

– simpler operation (especially in winter);

– lower vibrations;

– smaller midsection;

– possible use of heavier (simpler) fuels (kerosene instead of benzene).

Disadvantages of turboprop engines come from

– necessarily higher reduction ratio for transmitting rotation from the engine to the rotor (5-10 times higher than for piston engines);

– relatively long engine length which troubles helicopter layout;

– impossible reduction of fuel consumption with altitude.

Turboshaft engines with a free turbine surpass single-shaft engines in drive system which requires no master clutch and freewheel clutch as there is no mechanical link between the engine and the main rotor.

With turboshaft engine starting and spinning, the free turbine first remains stationary (due to high inertia of main rotor); hot gases pass through the free turbine blades. Only when engine begins to work in a stable mode and develops the power required, the gases begin to gradually spin the free turbine and rotate transmission and main rotor. If the engine fails, the main rotor – free turbine system continues rotating in the autorotation mode.

Helicopters equipped with gas-turbine engines have higher mass efficiency due to essentially lower mass of a power plant even though their transmission mass somewhat increases because of the increased gear ratio.

Thanks to less operation non-uniformity, gas-turbine engines do not need a special fan for cooling. The noise developed by the gas-turbine engine is less than that of the piston engine as well.

3.16.4. Selecting Engine

Maximum required specific reduced power (required thrust-to-weight ratio) is the main criterion, when selecting engine:

$$\tilde{N}_{max} = max\left(\tilde{N}_{Hstat}, \tilde{N}_{Hdyn}, \tilde{N}_{V_{max}}, \tilde{N}_{continued}\right). \tag{3.60}$$

Thrust-to-weight ratio is one of the main characteristics of a helicopter as a aircraft using rotors as moving (propulsive) units. It is expressed as a ratio of power plant takeoff thrust to the helicopter takeoff weight:

$$\tilde{N} = N_{p-pl} / m_o g . \tag{3.61}$$

Thrust-to-weight ratio affects main characteristics of a helicopter performance: maximum speed, vertical rate, manoeuvrability. When takeoff power is minimal, thrust-to-weight ratio is determined by the power plant thrust. With that, thrust-to-weight ratio must be limited by the given power of power plant engines so as it would not be too high.

In order to make thrust-to-weight ratio $\tilde{\mathbf{N}}_{\mathbf{max}}$ meet the conditions differing from the international standard atmosphere, it is reduced accounting for the change of power in relation to altitude $\bar{\mathbf{N}}_{\mathbf{H}}$, flight speed $\bar{\mathbf{N}}_{\mathbf{V}}$, power plant throttling coefficient $\bar{N}_{throt}$, and power losses ξ. Therefore,

$$\tilde{N}_{0max} = \frac{\tilde{N}_{p-pl}}{\bar{N}_H \cdot \bar{N}_V \cdot \bar{N}_{thrt} \cdot \xi} \cdot \frac{1}{m_0 \cdot g} = \frac{N_{0\,p-pl}}{m_0 \cdot g}, \tag{3.62}$$

$$N^I_{eng} = \frac{\tilde{N}^I_{0max} \cdot m^I_0 \cdot g}{n_{eng}}, \tag{3.63}$$

2here ^{I} is a number of approximations; n_{eng} is a number of engines for the helicopter power plant; $N^I_{0\,p-pl}$ is a power plant reduced power, $N^I_{0\,p-pl} \approx n_{eng} \cdot N^I_{eng}$; N'_{eng} is a design value of an engine takeoff power.

To produce the required amount of power, helicopter power plant must have two engines. This increases power plant mass but secures better a flight safety.

For helicopters with one engine, flight condition of a continued takeoff with one engine failed is not specific.

According to Johnson [18], medium helicopters require thrust-to-weight ratio equal to $\tilde{\mathbf{N}}_{0\,\mathbf{max}} \approx 0.015$ kW/N. however, these data seem obsolete and lowered as statistics shows that $\tilde{\mathbf{N}}_{0\,\mathbf{max}} = 0.020…0.025$ kW/N is required for light helicopters.

At early stages of design, reduced value of $\tilde{\mathbf{N}}_{\boldsymbol{0}\,\mathbf{max}}$ allows us to determine composition and payload at specified flight range and duration based on the number of engines and power to be produced by the power plant.

Thus, we use (3.63) to select the engine which would secure proper helicopter operation under prescribed conditions and meet specified performance requirements. Calculations are accomplished with regard to $\boldsymbol{m}_{payload}$, $\boldsymbol{H}_{static}$, $\boldsymbol{H}_{dynam}$, $\mathbf{V}_{\mathbf{max}}$, $\mathbf{L}$, extreme value of efficiency, and corresponding limitations of the helicopter operation. For that, we lay the value of $\mathbf{p*}$ along the $\mathbf{p}$ axis (Fig. 3.10) corresponding to the extreme value of efficiency criterion of the helicopter operation as well as its reduced thrust-to-weight ratio $\tilde{\mathbf{N}}_{\boldsymbol{0}\,\mathbf{max}}$, and takeoff mass $\mathbf{m}_{\mathbf{o}}^{\mathbf{i}}$ in $\boldsymbol{i}$ th approximation, the latter values helping us to design the value of engine power.

Type and model of the engine is selected from the engine catalogue or other documents which provide the necessary data on its overall and installation dimensions, arrangement in the helicopter, specific parameters (specific mass and fuel consumption), etc.

For training purposes, future helicopter designers can refer to foreign brands which parameters are given in descriptions (Appendix 7).

If the catalogue includes several types of engines with identical or close values of power and other parameters, then preference is given to the engine type with the lowest specific fuel consumption and air consumption, the smallest mass and overall dimensions.

3.17. Specifying Helicopter Performance

In this paragraph, we shall consider helicopter external aspect formation from the viewpoint of mutual space linkage of helicopter parts, their form and structure-force diagrams which serve as a precondition for the general view and layout drawing implementation.

Helicopter's performance and geometry are selected after you calculated $\mathbf{R}$, σ, and $\omega\mathbf{R}$ and chose engine and $\mathbf{m}_{\boldsymbol{0}}$ providing required level of efficiency.

Helicopter performance is calculated based on aerodynamic design as a part of general design. Obtained results must conform to the initial data.

3.17.1. Determining Main Rotor Parameters and Location

At a characteristic radius, blade width is $(\overline{\boldsymbol{R}} = 0.7)$

$$\boldsymbol{b}_{0.7} = (0.0471...0.0628)\cdot \boldsymbol{R} \approx 0.055\cdot \boldsymbol{R}, \tag{3.64}$$

whence $\lambda_{av} = \boldsymbol{R} / \boldsymbol{b}_{0.7} \approx$ 18.182 and $\boldsymbol{z} = \pi \cdot \lambda \cdot \sigma$.

Such parameters as blade twist, its planform shape, profile, and the number of blades are selected to compromise different operational conditions of helicopter flight so that to provide optimal aerodynamic characteristics of the rotor. At the stage of predesign, these and other parameters are taken into account with regard to their effect on mass and helicopter performance.

The place to locate the main rotor hub above the fuselage is determined by (Fig. 3.15):

– hub cutout deflection angle down on restraint of the centrifugal droop stop (~1.5...2°) [20];

– static deflection of blade ($y_R \approx 0.12 \cdot R$) [24];

– vertical distance between the nonrotating blade tip and the tail boom of the helicopter structure $(0.05...0.07) \cdot R$ [25].

At the cruise flight, fuselage axis is directed to the flight path by locating the main rotor shaft axis not perpendicularly to the fuselage longitudinal axis but is tilted forward at the angle of 4...6° (Fig. 3.13). Besides, helicopter fuselage could be arranged so that the main rotor shaft axis would deflect from the normal axis to the right by 2...3° (if looking in the flight direction from behind). Due to this, the roll of helicopter to the right decreases on hovering and at low speeds, and the vertical takeoff is achieved with coincidental lift-off of the main landing gear (or landing/touchdown on both main landing gear units).

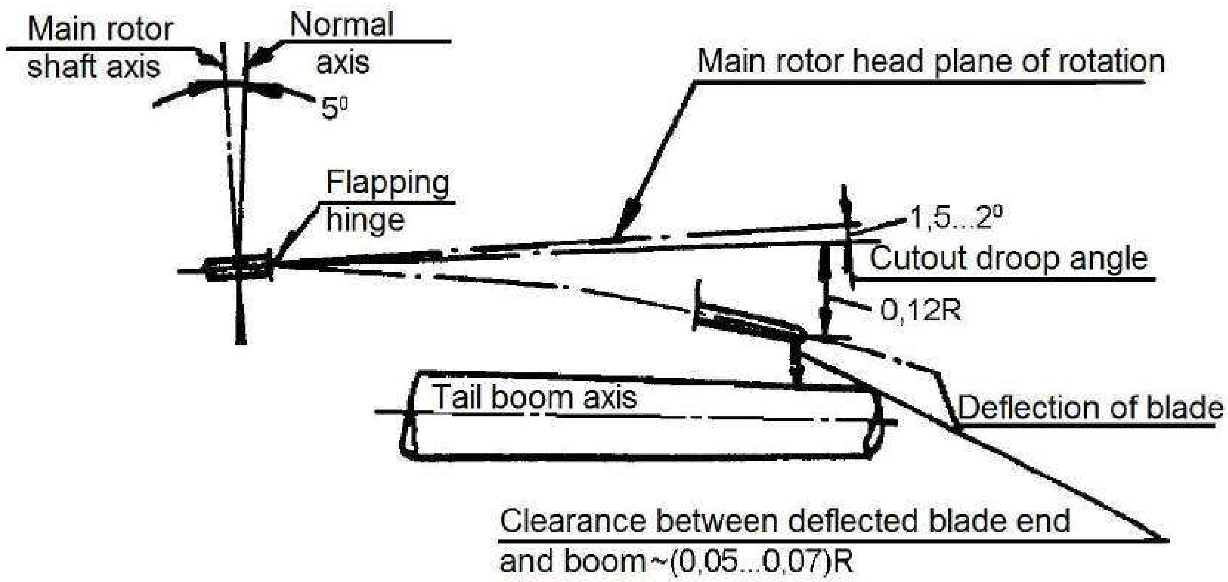


Fig. 3.13. Location of Hub and Main Rotor above Fuselage

3.17.2. Selecting Tail Rotor Parameters and Location

Tail rotor is installed on the tail boom beyond the area of windmilling and intensive aerodynamic interference of the main rotor. With these, the overall view of the tail rotor arm is

$$L_{TR} = R + R_{TR} + \delta. \tag{3.65}$$

All superlight helicopters as well as some light ones have no tail rotor pylon by reason of relatively low thrust of tail rotor.

In most helicopters, tail rotor is a pusher. It is installed on the tail rotor pylon on the right of flight direction. In case of relative distance $\overline{Z}_{TR} = Z_{TR}/R_{TR}$ between the tail rotor disc plane and vertical stabilizer over the range 0.3 < $\overline{Z}_{TR}$ < 0.5, pusher rotors suffer the relative loss of thrust several times less [15] than puller rotors (Fig. 3.14). Besides, this rules out additional dynamic loading of the tail rotor pylon by the pulsating airflow rejected by the tail rotor.

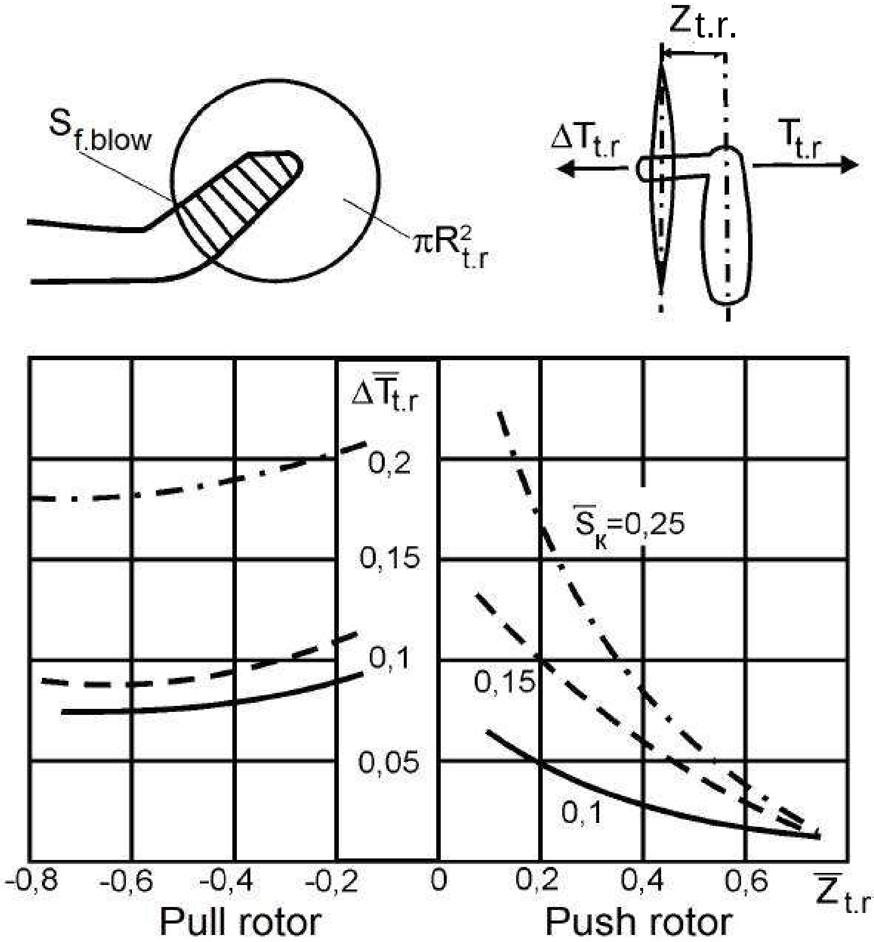


Fig. 3.14. Impact of Tail Rotor Model and Parameters on Respective Thrust Loss

Rough number of the tail rotor blades is determined by the ratio $z_{TR} = 2/3 \cdot z$, the solidity factor of the tail rotor $\sigma_{TR} = (1.7...2.3) \cdot \sigma \approx 2\sigma$. As a rule, tail rotor blades are not twisted, they have a squared shape in the layout.

According to Vildgube [19], maximal starting torque of the main rotor is modulo equal to reactionary torque effecting the helicopter frame:

$$M_t = M_P = \frac{n_{eng} \cdot N_{eng}^{cat} \cdot R \cdot \xi}{\omega \cdot R}. \quad (3.66)$$

In this case, under condition of helicopter balance at the aerodynamic design point, the tail rotor thrust is:

$$T_{TR} = M_P / L_{TR}. \qquad (3.67)$$

Approximately, $T_{TR} \approx (0.06...0.07) \cdot G_0$, where G_0 is normal take-off weight of a helicopter.

All-movable or non-all-movable stabilizer can be used to provide longitudinal trim features and helicopter stability. The stabilizer is tapered in planform and, often, symmetrical in cross-section. It is mounted at the end of the tail boom for the purpose of ensuring maximal distance from the centre of inertia and in order to reduce harmful inductive influence of the main rotor. The parameters and the stabilizer's mounting position are selected after the stability and controllability analysis of the helicopters (including statistical prototype analysis) is fulfilled. At this stage,

$$S_{HT} \approx 0.004 \cdot \pi \cdot R^2. \qquad (3.68)$$

3.17.3. Selecting Fuselage Parameters

For superlight helicopters, power fuselage frame is usually performed in the integrated truss structure. The integrated truss structure accommodates passenger cabin, cockpit, armchairs, control elements, propulsion reduction gear with a rotor, power unit, tail boom with tail rotor, and skid. Skin can take up external loads (e.g. surface aerodynamic ones) and transfer them to the integrated truss structure. This structure stands all kinds of loads, both external (mass, aerodynamic, inertial) and internal (torque, twist, shear forces). As the skin is not a part of the fuselage power structure, extra reinforcement is not needed in the cuts.

Helicopter compartment has to include considerable glass-covered area in order to provide the best multidirectional view for the crew, minimum drag, great occupancy, and design modularity (for convenient removal, installation, and maintenance).

Tail boom could be designed as the integrated truss structure made from thin-walled duralumin tube or light alloys monocoque. Though, the most reasonable construction is a one-piece construction, as every stud of integrated truss structure produces oscillation and concentrates the in joints. If you use uniform section solid tube for tail boom, you will surely get weight losses.

The selection of creating a superlight helicopter may be individual. Fuselage type (frame, truss, or mixed construction) is selected so that to balance fuselage construction deflection rate and the loads acting on helicopter channels and control elements. You must also pay attention to helicopter purpose, when selecting the fuselage type.

Helicopter statistics analysis shows that integrated truss fuselage structure (as in Bell 47G-3B-1 and Alouette II) is not used even in modern superlight helicopters (Fig. 3.15). It is replaced by the frame structure with relatively thick low stressed skin (as in Gazelle helicopter) or by the frame structure with thin stressed skin (as in Puma helicopter).

Beam system consists of the frame, stressed skin, and load-bearing units. Framework consists of two sets, transverse (frames and diaphragms) and longitudinal (stringers, beams and stiffened elements).

According to statistical data, the pitch of the normal frame is 450…500 mm for most structures, 150…200 mm for stringers, 250…350 mm for ribs. For tail parts of the central fuselage and tail beam, the pitch and the stringer length is shown by the stringer distribution diagram.

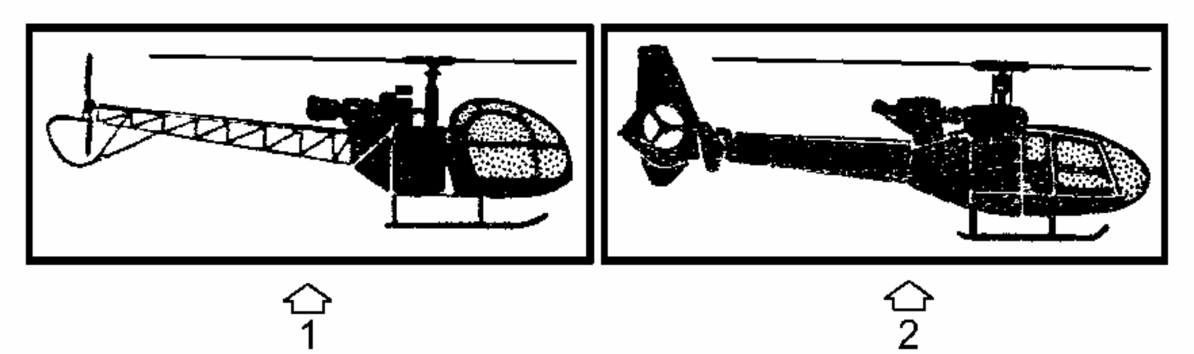


Fig. 3.15. Evolution of *Aerospasial* Light Helicopter Fuselages:

1 – Alluet II, made the first flight on March, 12, 1955, take-off weight equals 1650 kgf; $C_x S$ = 1.85 m^2;

2 – Gazelle, made the first flight on April, 7, 1967; take-off weight equals 1900 kgf; $C_x S$ =0.8 m^2

For light helicopters, the forward fuselage ratio (from the tip to the main rotor's axis of rotation of) to the length of the fuselage L_f is recommended as 0.36...0.40, and aspect ratio of fuselage ($\lambda_f = L_f / h_f$) - 4.5...5.5, where h_f is a constructional depth of the fuselage.

To reduce the fuselage drag at the angle of attack $\alpha_f \approx 0$, it is desirable to set the largest cross section at a distance $0.435 \cdot L_f$ from the nose fuselage.

3.17.4. Selecting Landing Gear

Landing gear (LG) is an alighting gear designed to absorb the kinetic energy during take-off, and to secure stable position of a helicopter during taxiing and parking on the ground. These include gear and tail wheel equipped with a liquid-gas shock absorbers. Shock absorbers with pneumatics wheels absorb the energy of shock loads affecting the helicopter landing and movement along the ground. Tail skid block protects the blade tail rotor and tail boom during landing with a large pitch angle. Usually, dampers absorb 65...75 % of the kinetic energy of impact, and 25...35 % are absorbed by tires.

Fig. 3.16 shows main parameters of the landing gear with the nose wheel together with their conventional symbols:

- distance from the front wheel to the centre of the helicopter, **a**;
- distance from main LG wheels to the centre of the helicopter, **b**;
- base, **c**;
- track, **B**;
- pull over angle, θ;
- landing gear height, **h**;
- nose over angle, γ.

Landing gear track **B** influences the nose-over protection angle γ_{no} and the characteristics of the earthly resonance. Earthly resonance can be avoided virtually at any values of landing gear track using shock absorbers with a small amount of preload force and dampers of the MR vertical hinges. To exclude a nose over (turning over) manoeuvre of helicopter rotated relative to 1-3 (see Fig. 3.16), it is required that the slope of the resultant lateral force and the weight of the helicopter landing site surface does not exceed β.

To exclude nose over manoeuvre of helicopter,

$$\boldsymbol{tg}\,\gamma = \frac{\boldsymbol{aB}}{2\boldsymbol{Y}_{mc}\sqrt{\boldsymbol{c}^2 + 0.25\boldsymbol{B}^2}} \geq \mu \tag{3.69}$$

where μ is the tire-against-runway friction coefficient ($\mu_{\mathbf{max}}$ = 0.6…0.8 for rubber and dry concrete with surface irregularities taken into account); $\boldsymbol{Y}_{MC}$ is the height of the helicopter CM above the ground.

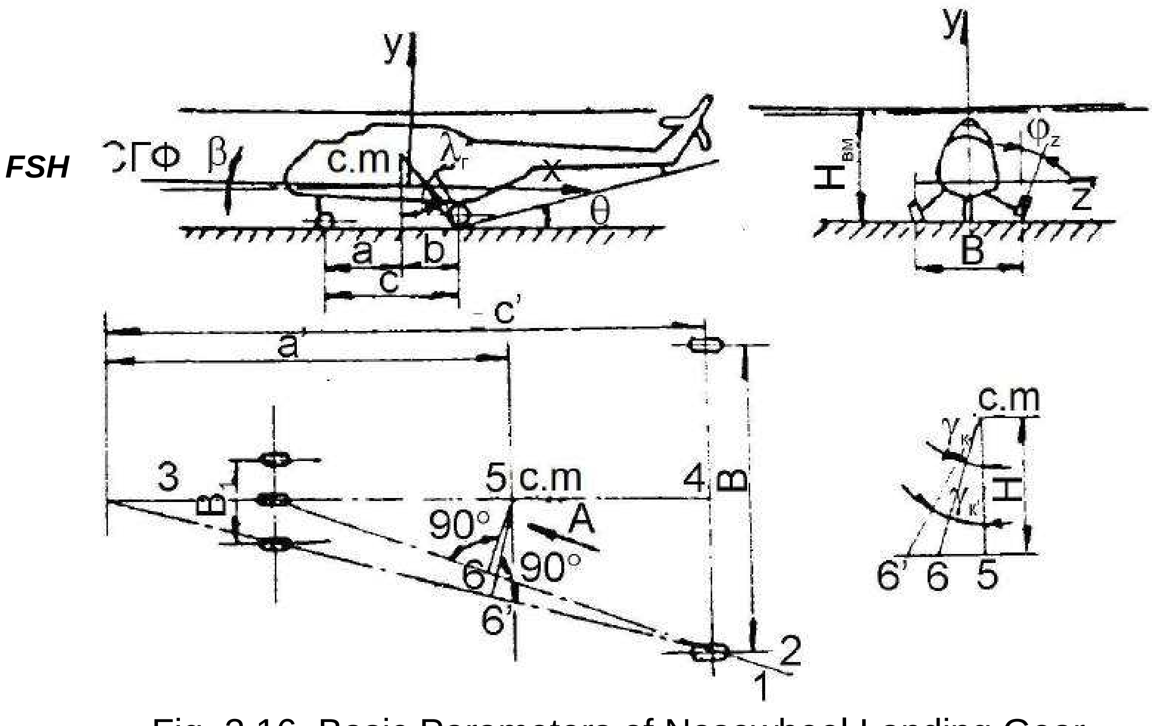


Fig. 3.16. Basic Parameters of Nosewheel Landing Gear

As a rule, a nose-over prevention angle γ_{no} shall be 30-40°. Modern helicopters tend to decrease the tricycle landing gear nose-over angle to the value of about 25-30° which conforms to $tg\gamma_{no} = 0.46 \ldots 0.58$.

For the tricycle LG arrangement, $\boldsymbol{B}$ is determined by the ratio

$$\boldsymbol{B} \geq \frac{2\boldsymbol{tg}\gamma_{no} \cdot \boldsymbol{H} \cdot \boldsymbol{c}}{\sqrt{\boldsymbol{a}^2 - \boldsymbol{tg}^2\gamma_{no} \cdot \boldsymbol{H}^2}} \qquad (3.70)$$

If the design value γ_{no} is smaller, then you should choose quadricycle landing gear, replacing the nose landing gear by two landing gear legs placed at the distance of 0.5 x B_1 from the centreline so that the angle γ_{no}' meets the requirement. In this case, B_1 is defined by the condition.

$$\boldsymbol{B} \geq \frac{2\boldsymbol{tg}\gamma_{no}' \cdot \boldsymbol{H} \cdot \boldsymbol{c}'}{\sqrt{\boldsymbol{a}'^2 - \boldsymbol{tg}^2\gamma_{no}' \cdot \boldsymbol{H}^2}} \qquad (3.71)$$

In this case, it is highly desirable to mount the main landing gear, the frame and the main gear box units to strong frames.

Figs. 3.17, 3.18, 3.19 below show typical designs of the helicopter wheeled landing gear elements.

3.18. Developing Units Structure for Light Helicopters

Here we shall consider helicopter structure to provide an example how specific equipment structure can be simplified for ultra-light and, in some cases, for light helicopters.

The units exposed for further development and facilitation include main rotor, tail rotor, skid landing gear, control system, transmission, and power plant.

Materials on the main and tail rotor are adduced involving some of the theory clauses needed to design units.

Even though some simplifications are applicable for ultra-light helicopter structures, it is extremely difficult to create a light rotorcraft which would have simple structure, be reliable and inexpensive.

Fig. 3.17. Main Landing Gear Shock Absorber:
1 – strut; 2 – nut; 3 – axle box; 4 – bronze axle box;
5 – piston ring; 6 – inner cylinder; 7 – base cylinder;
8, 10 – bronze boxes; 9 – duralumin hub; 11 – base sliding cylinder; 12 – charging valve; 13 – end piece (lug); 14 – nut; 15 – strut seal pack; 16 – steel valve; 17 – hub;
18 – cylinder; 19 – mechanical stop; 20 – thrust bush

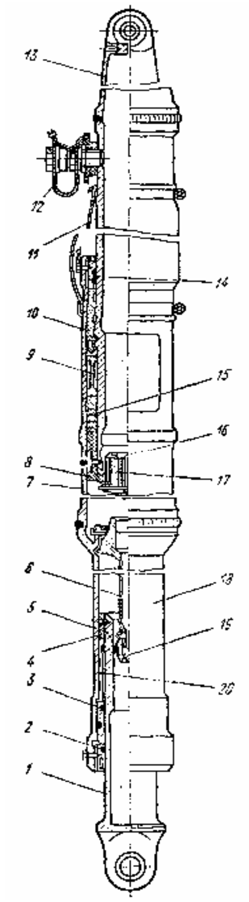
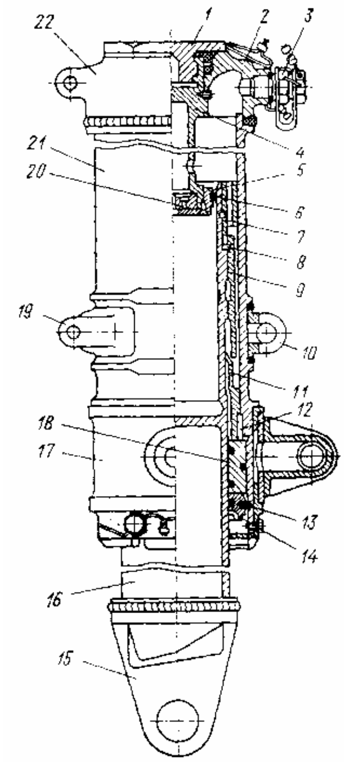


Fig. 3.18. Nose landing Gear Shock Absorber:
1 – nut; 2 – door;
3 – charging valve;
4 - inner cylinder;
5 – piston; 6 – piston ring; 7 - bronze box; 8 – rim;
9 – upper stop;
10 – eye lug;
11 – catch lock;
12 – lock pin;
13 – bronze hub; 14 – nut; 15 – (landing gear) fork;
16 – sliding cylinder;
17 – scissor; 18 – box;
19, 22 - eye lugs;
20 – floating valve;
21 - cylinder



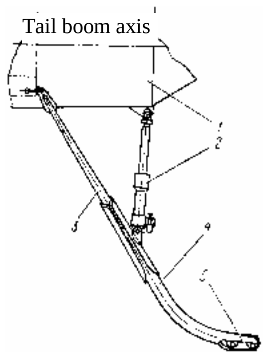


Fig. 3.19. Tail Bumper:
1 – tail boom; 2 – shock absorber;
3 – struts (two pieces); 4 – tube; 5 – bumper

3.18.1. Main Rotor

Today many new technical solutions are introduced to helicopter engineering which significantly simplify its structure and operation, reduce maintenance time and operating costs, etc.

A lot of improvements were put in the structure of main rotor and its heads to which new load-bearing structures were implemented.

3.18.2. Main Rotor Hubs

Third-generation helicopters widely employ main rotor hubs with hinged (Fig. 3.20), hingeless rigid (Fig. 3.21), semi-rigid (Fig. 3.22) and elastic (Fig. 3.23) blade mounting.

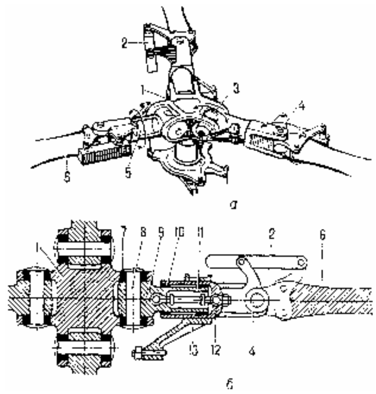


Fig. 3.20. Articulating Hub:
a – general view; b – cutout;
1 – hub body; 2 – elastomeric damper;
3 – flapping hinge; 4 – drag hinge;
5 – feathering hinge, 6 – blade;
7 – flapping hinge head; 8 - flapping hinge pin;
9 - feathering hinge trunnion;
10 – feathering hinge heads; 11 – wire torsion;
12 – feathering hinge case; 13 – push rod

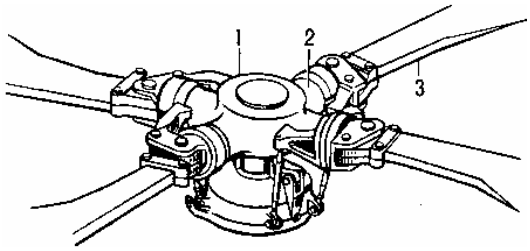


Fig. 3.21. Rigid Hub:
1 – hub body; 2 – feathering hinge;
3 – elastic blade segment

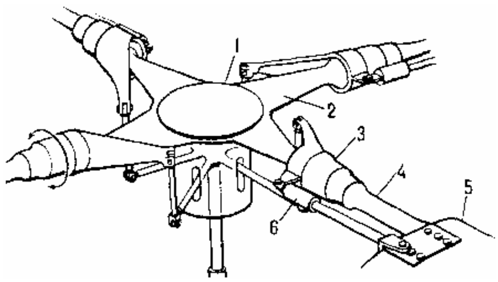


Fig. 3.22. Semi-Rigid Hub:
1 – hub body; 2 – elastic segment of case;
3 – feathering hinge;
4 – outer head elastic segment; 5 – blade;
6 – damper

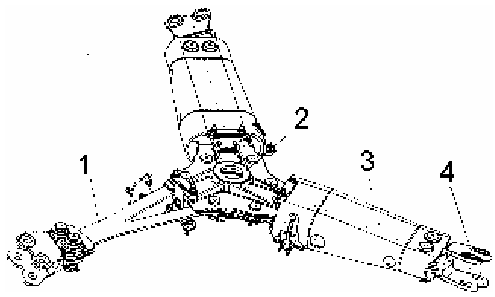


Fig. 3.23. Elastic Hub:
1 – flexbeam; 2 – hub of main rotor;
3 – sleeve; 4 – blade fitting

One way to simplify the main rotor head construction is to introduce elastic members instead of standard hinges. Flapping hinges are more often replaced with elastic members. However, elastic members have fragility in flapping plane which can cause blade flapping movement and high stiffness in the plane of rotation. Such members can consist of a set of elastic plates. In this case, they can twist like flexbeam and fulfil the functions of feathering hinge.

If hinge attachment is provided for blades, their vibration in the plane of rotation as well as rotation around the fore-and-aft axis is secured by using flapping, drag, and feathering hinges. In the case of hingeless attachment, elastomeric spherical angular contact bearings allow us to replace flapping, drag, and feathering hinges. Elastic elements with elastic plates (torsion plates) can be used instead of flapping and feathering hinges with vibration damping in plane of rotation related to drag hinge dampers.

Vibration of blades is damped in drag hinges both by friction dampers and due to friction in their bearings.

Elastic plates which connect opposite blades take up centrifugal loads and act as elastic elements in cases of blade flapping in thrust plane and their rotation relative to fore-and-aft axis.

Statistical analysis of ultralight helicopter heads shows that today blades are usually connected to the main rotor head by means of hingeless elastic scheme due to

– less items involved (no “sleeves” with hinges);

– less greasing points which means that less maintenance time is needed, and operating costs go down;

– less factory labour hours,

– smaller takeoff weight, etc.

that is why we advise future helicopter designers to use elastic blade connection in ultra-light helicopters (Fig. 3.24).

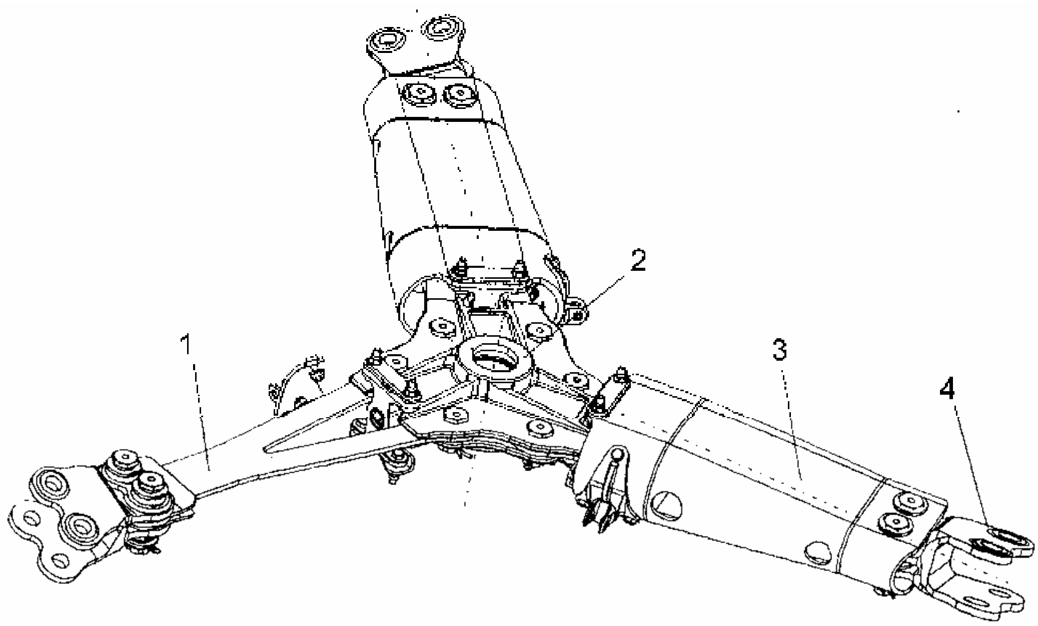


Fig. 3.24. Load-Bearing Structure of Elastic Hub of Helicopter Main Rotor (based on laminated torsion bar):

1 – laminated torsion bar; 2 – main rotor hub; 3 – sleeve; 4 – blade attachment fitting

3.18.3. Main Rotor Blades

Designers of main rotor blades wok with the following source data:
- main rotor diameter D_{MR}, m;
- solidity of rotor σ;
- number of blades z;
- blade shape in planview;
- sections of blade cross-section;
- blade twist;
- manufacturing materials;
- main rotor tip speed ωR, m/s (number of MR revolutions, 1/s);
- maximal helicopter flight speed V_{max}, km/h;
- main rotor blade weight, kg.

When selecting the blade form, helicopter designers account for aerodynamics requirements, durability, current level of production technology, helicopter takeoff weight and service conditions. For the most applied main rotor blades in helicopter industry, the character of variation of their geometrical features ($\bar{b}$ и $\bar{c}$) of section on the radius of main rotor is shown in Fig. 3.25.

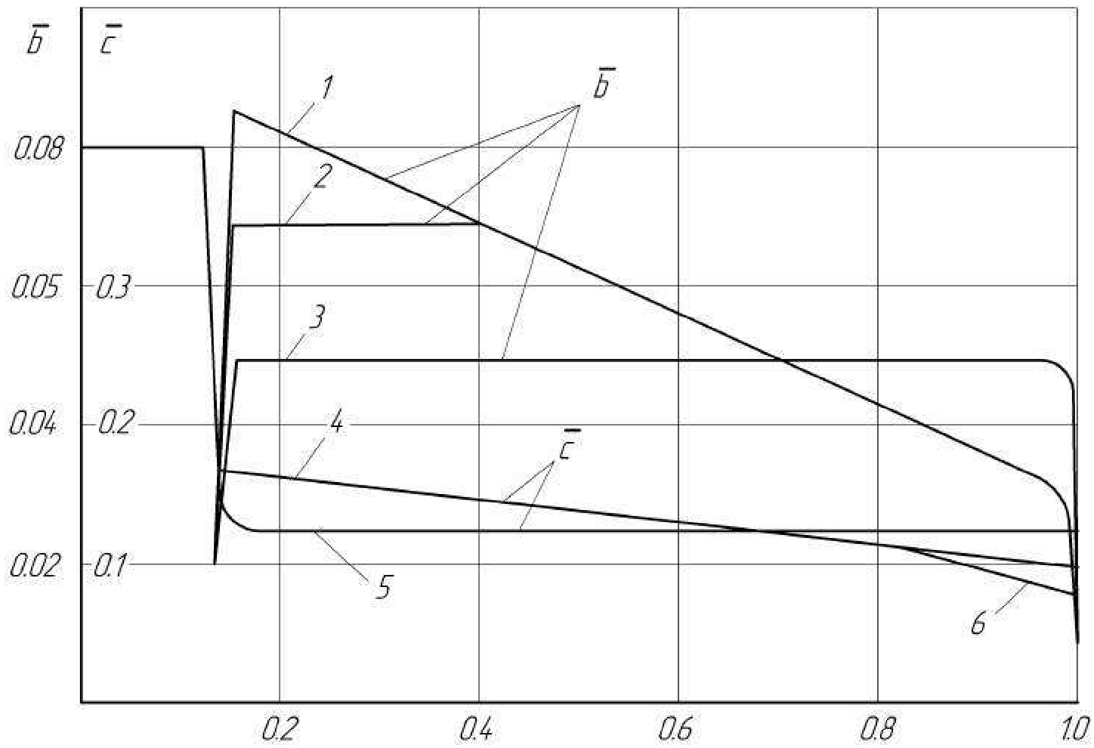


Fig. 3.25. Nature of Changes in Geometric Characteristics of Main Rotor Blades depending on its Radius R:
1 – trapezoidal; 2 – semitrapezoidal; 3 – rectangular;
4 – with variable relative thickness of the profile; 5 – with constant relative thickness;
6 – with variable relative thickness and vibration-resistant tip

Consider the effect of the blade geometric parameters on its form drag in cases when the coefficient of profile-drag power $m_{airfoil}$ exceeds half of the total losses for blades.

Fig. 3.26 shows the coefficient of profile-drag power for blades of different geometric forms as the coefficient of propulsive thrust changes. The trapezoidal twisted blade with high-speed airfoil at the tip has the smallest value $m_{airfoil}$. Rectangular twisted blade with the high-speed airfoil at the tip gets a little behind the trapezoidal one, but it has significant advantages in terms of design and manufacturing technology. The highest airfoil drag (by 20% and more) is common for rectangular blades, flat blades, blades without high-speed airfoils at the tips as their airfoil drag is 20 % and more higher that the airfoil drag of a trapezoidal twisted blade with high-speed airfoil at the tip at a hovering mode.

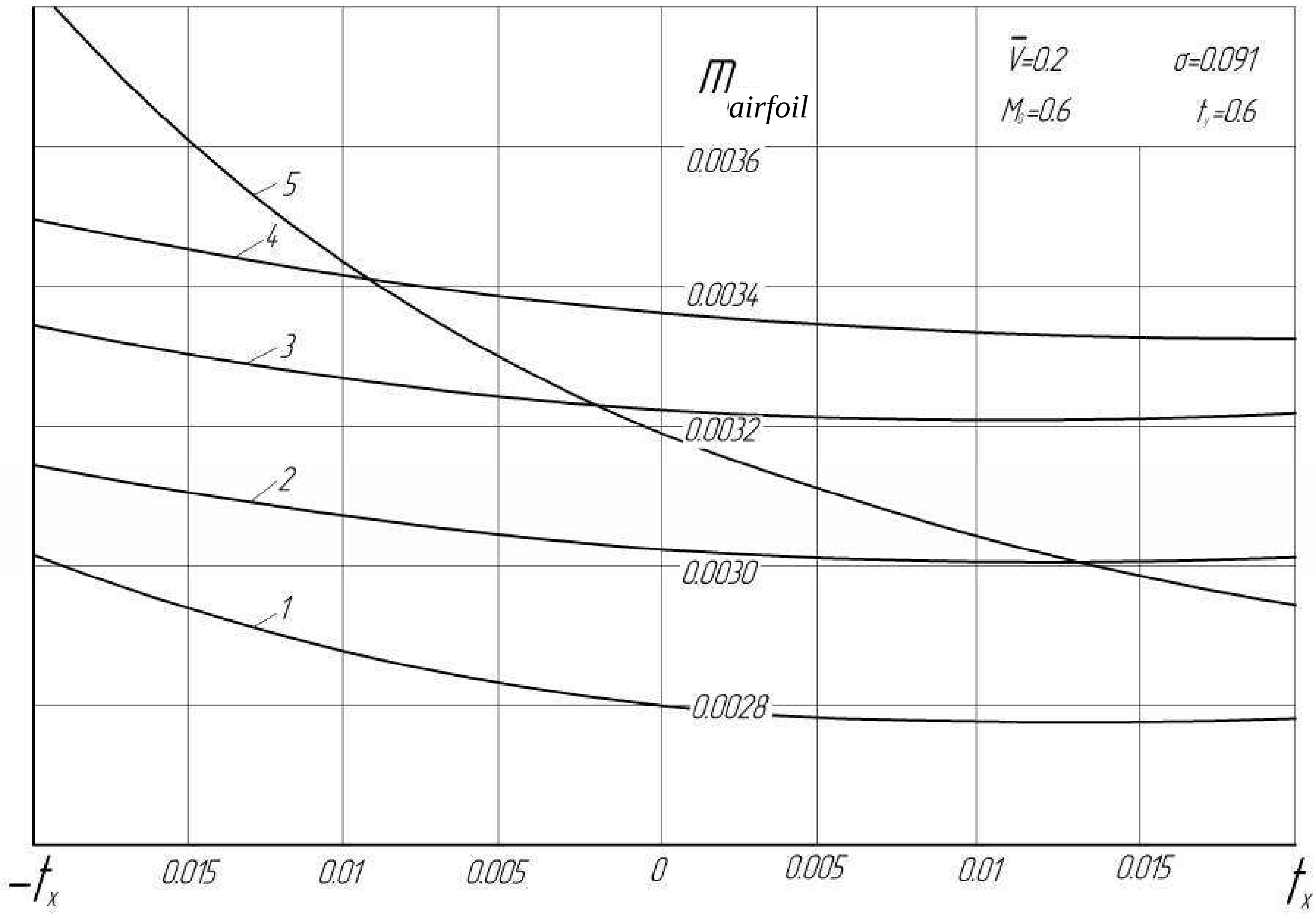


Fig. 3.26. Effect of Main Rotor Blades Shape on Coefficient of Profile-Drag Power:

1 – trapezoidal twisted blade with high-speed airfoil at the tip;
2 – rectangular twisted blade with high-speed airfoil at the tip;
3 – rectangular twisted blade with airfoil NACA 230-12;
4 – rectangular twisted blade with symmetrical airfoil NACA 0012;
5 – trapezoidal flat blade with high-speed airfoil at the tip

At $\mathbf{M_0} \leq \boldsymbol{0.5}$, trapezoidal twisted blades give the 8 %-decrease of $m_{airfoil}$ as compared to rectangular twisted ones, thus reducing the necessary engine capacity by about 4 %.

At $M_0 \geq 0.6...0.7$ and $\omega\mathbf{R} = \boldsymbol{230...240}$ m/s, high-speed airfoils are preferable at the blade tips because they provide 40 %-decrease in drag, thus reducing the necessary engine capacity by about 20…25 %.

At autorotation mode, twist is more important than the blade shape because twisted blades prove to be much worse than the flat ones at autorotation regime even though they are better at hovering and take off modes.

When calculating the blade width at the characteristic radius,

$$b_{av} \approx b_{0.7} = \frac{\sigma\pi R}{z} \quad \text{or} \quad \bar{b}_{0.7} = \frac{b_{0.7}}{R} = \frac{\sigma\pi}{z} . \tag{3.72}$$

In modern helicopters, $\frac{\sigma}{z}$ of the MR blades vary from 0.015 to 0.02, so $\bar{\mathbf{b}}_{0,7} = (0{,}015\ldots0{,}02)\boldsymbol{\pi}$ is 0.0471… 0.0628 and 0.055 in average. This means, that approximately

$$b_{0.7} \approx \bar{b}_{0.7} R \approx 0.055 \cdot R . \tag{3.73}$$

The relative thickness of blade airfoils $\bar{\mathbf{c}}$ is selected based on their lifting ability, smoothness of stall, minimal airfoil drag, and minimal wave losses.

The most common airfoils are NACA 63A012, NACA 63A015, NACA 230, symmetrical airfoils NACA-000, high-speed airfoil TsAGI P-57-9, etc. Also, airfoils can be manufactured specially for precise helicopter blades.

When choosing blade cross airfoils, helicopter designers give preference to airfoils which provide relatively low airfoil drag, high lifting ability, M_{tw} , and α_{tw}; change lifting ability smoothly at post-stall angles of attack; and change centre of pressure when the angle of attack and **M** grows. Do not forget here that you can increase the weight-lift ability at the same takeoff weight by increasing the load ratio coefficient, reducing blade aerodynamic losses, and taking down airfoil drag.

As airfoil losses are proportional to the third degree of speed, and thrust force is proportional to its second degree, it is advantageous to create the thrust force by the part of blade close to its butt (as it operates at low speeds) thus reducing the angle of attack at the blade tips.

Blade twist plays a positive role at hovering and takeoff because it achieves uniformity of inductive speeds along the radius and, thereby, increases the blade efficiency. During the forward flight, twist improves the rotor efficiency by cutting down the impact of compressibility.

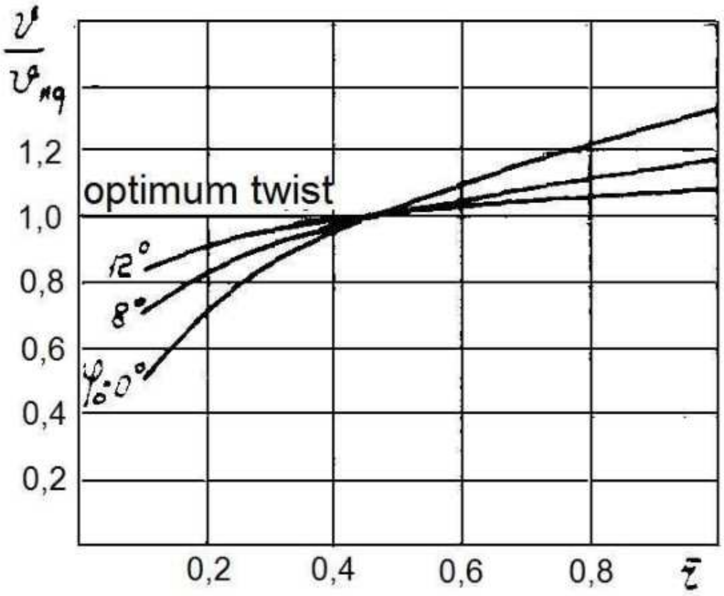


Fig. 3.27. Relationship between the induced velocities by the radius and the induced velocity with optimum twist on the blade twist value φ_0

Fig. 3.27 shows how induced velocity by the rotor radius relates to the induced velocity of the rotor with optimal twist. 12 %-twist brings the distribution of induced velocities close to an optimal one. Further increase proves unreasonable as it derates stall performances on the retreating blade and decreases its auto-rotating properties.

Blade mass is an important parameter due to the fact that a centripetal force affecting joints and rotor hubs depends on mass. Mass-saving blade design involves certain restrictions for blade flapping, lock number, and minimal implemented structural and technologic dimensions.

Alternating stresses, flap-pitch and chord flutters, and wing divergence depend on the blade lock number

$$\gamma_0 = \frac{c_y^\alpha \rho b_{0.7} R^4}{2 J_{fh}}, \tag{3.74}$$

where c_y^α is a coefficient derivative of the lift on the profile angle of attack; ρ is the air density at a height H = 0; J_{fh} is a blade moment of inertia relative to flapping hinge axis.

As $J_{fh} = k_1 m_{bl} R^2$, then $\gamma_0 = \dfrac{c_y^\alpha \rho b_{0.7} R^4}{2 k_1 m_{bl}}$, where m_{bl} is a blade mass, kg.

Total mass of blades $\Sigma m_{bl} = k_{bl} \sigma R^3$.

If we substitute total mass of blades for γ_0, we get

$$\gamma_0 = \frac{c_y^\alpha \rho b_{0.7} R^2 z}{2 k_1 k_2 \sigma R^3}. \tag{3.75}$$

After substituting σ for $\sigma = \dfrac{z b_{0.7}}{\pi R}$,

$$\gamma_0 = \frac{c_y^\alpha \rho \pi}{2 k_1 k_{bl} z} \tag{3.76}$$

and

$$k_{bl} = \frac{c_y^\alpha \rho \pi}{2 k \gamma_0 z}. \tag{3.77}$$

For $m_{bl} = \dfrac{k_{bl}\sigma R^3}{z}$,

$$m_{bl} = \frac{k_{bl} b_{0.7} R^2}{\pi} \quad \text{or} \quad m_{bl} = 0.0175 \cdot k_{bl} R^3 . \tag{3.78}$$

Fig. 3.28 shows how k_{bl} depends on the blade radius.

Blade design experience proves that most conditions that prevent standard operation of the main rotor can be easily eliminated for the lock number $\gamma_0 < 4\ldots5$ as compared to lighter blades with $\gamma_0 < 6\ldots7$ [19]. Mass-saving of the structurally technologic blade mass is relatively easily reached for blades of high dimensions and leads to large values of lock number $\gamma_0 > 4\ldots5$. Mass of small blades is a minimal when $\gamma_0 < 3\ldots4$.

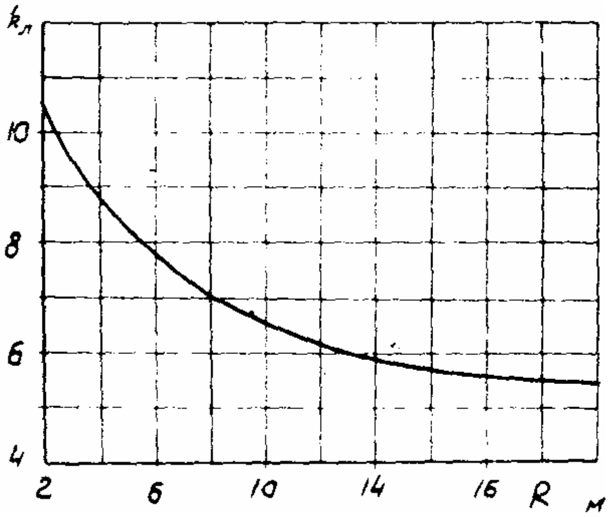


Fig. 3.28. Dependence of Blade Radius R on Mass Ratio k_{bl}

When designing blades, remember that active stresses in the spar caused by centrifugal force and its proper weight must lie within tolerable limits (Table 3.14):

$$\sigma_{cf} \le (\sigma_{cf})_{add}; \qquad \sigma_{man} \le (\sigma_{man})_{add}, \tag{3.79}$$

$$(\sigma_{prod})_{add} = \sigma_{br}/f\delta, \tag{3.80}$$

where $\mathbf{f}$ is a safety coefficient equal to 1.5; δ is an augmentation ratio of peak stresses in case of blade fall on the droop stop in relation to stresses of dead weight of the blade.

Table 3.14

Permissible Stresses of Various Materials

Material	Stress level of centrifugal force
Steel	$(\sigma_{cf})_{add} = (22\ldots26)\cdot10^7$, Pa
Avial	$(\sigma_{cf})_{add} = (7\ldots8)\cdot10^7$, Pa
Titanium	$(\sigma_{cf})_{add} = (15\ldots16)\cdot10^7$, Pa
Glass-fibre laminate (1:1)	$(\sigma_{cf})_{add} = (6\ldots8)\cdot10^7$, Pa
Glass-fibre laminate (10:1)	$(\sigma_{cf})_{add} = (14\ldots15)\cdot10^7$, Pa

In order to lower weight of the blade, increase its stiffness, and reduce active stresses, it is necessary to put the spar material as far as possible from zero-twist axes of airfoil. Actually, the best form of spar is the one in which the material is located on the internal circuit of the airfoil.

Growing blade stiffness tends to increase natural frequency of the blade. Sometimes it allows us to reduce resonant vibrations. However, increased stiffness can move one frequency away from the excitatory harmonic and make another frequency approach them.

The stiffness of a flap blade can be increased by increasing the airfoil thickness ratio of blade cross-section; at the same time, it will increase airfoil drag and, consequently, reduce lift-drag ratio of the rotor.

In the disk plane with extruded spars and spars (made of composite material), stiffness can be increased by increasing the spar width ratio, though at $\boldsymbol{b_{bl}} > 0.4$, centring control of blade misaligns far back and calls for a heavier antiflutter weight (dead-weight).

Let us consider a typical scheme of changes in mass per unit length, spar height, and static stresses depending on the blade radius (Fig. 3.29).

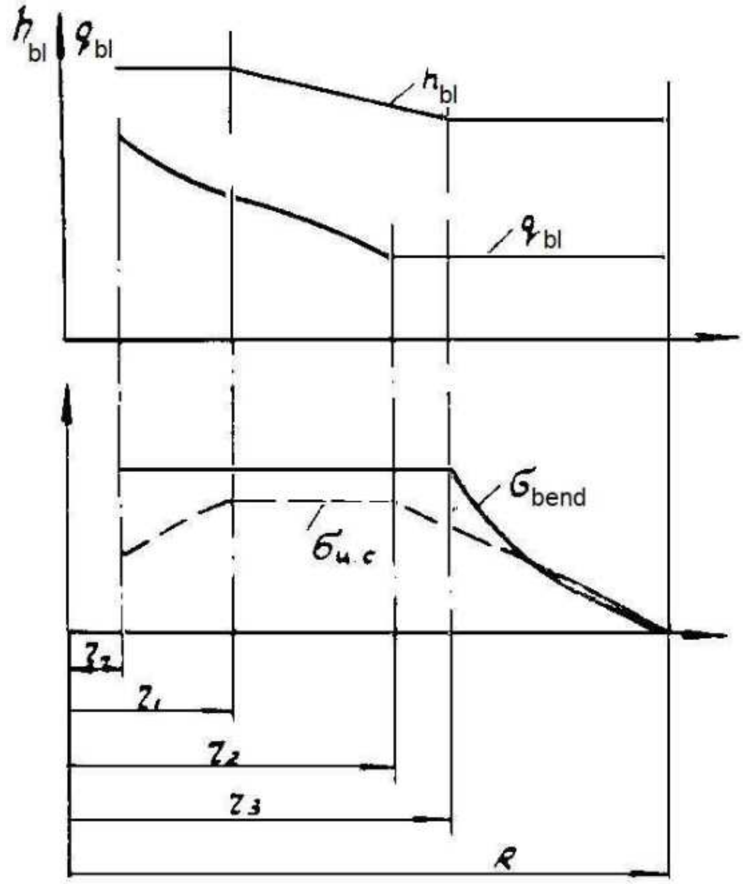


Fig. 3.29. Typical Scheme of Changes in Mass per Unit Length, Spar Height, and Static Stresses depending on the Blade Radius

Let us call the area from $\mathbf{R}$ to $\mathbf{r}_2$ a ballast area as here the balancing weight is arranged. Stresses caused by bending and centrifugal stiffening increase from 0 to allowable values. In the area from $\mathbf{r_2}$ to $\mathbf{r_1}$, allowable values of stress keep constant due to increased spar web height and thickness; and in the area from $\mathbf{r_1}$ to $\mathbf{r_0}$, σ_{bend} at constant $\mathbf{h}$ keep constant due to regular thickening of spar webs. At the same time, stresses from centrifugal stiffening are somewhat reduced.

Using the formulae from structural mechanics and strength analysis, we can perform preliminary strength calculations of the designed blades, namely calculate $\mathbf{q}$, $\mathbf{i}$, GJ_w, $\mathbf{EJ}_y$, $\mathbf{EJ}_x$, N_{eng}, and σ relative to the rotor radius.

Alternating stresses acting on the operating blade spar can be reduced by reducing γ_0, though this is actually impossible because in this case spar web and skin would become too thin to preserve their stability. For existing blades, mass of blade tips varies from 10 to 17 % of the whole blade weight and can be determined as [25]

$$m_{tip} = k_{tip}(N_b)^{3/2}, \tag{3.81}$$

where N_{bl} is the centrifugal force of the blades measured in thousands of daN (decanewtons); for blade tips made of steel tubes $k_{tip} = 0.055$, for the pressed spar tips $k_{tip} = 0.135$.

The history of designing main rotor blades provides us with four groups of blade structures which basically differ in material and spar manufacturing procedures:

1. Wooden blades.
2. Metal blades.
3. Mixed blades:
   – blades with steel tubular spar, wooden ribs, fabric or plywood skin.
   – blades with steel tubular spar or with pressed aluminium alloy with fiberglass frame and glass-cloth or metal tail sections.
   – combined reinforced plastic blades.
4. Blades made of composite materials, executed by tissue packing or by winding strands of glass, boron, carbon fibres, or by their combinations.

Currently, blades of the second and fourth types are the most common.

Ultralight helicopters usually require blades made of composite materials due to their small size, simple construction, shape, and size as well as relatively simple manufacturing methods (Figs 3.30, 3.31).

Fig. 3.30. Rotor of Ultralight Helicopter

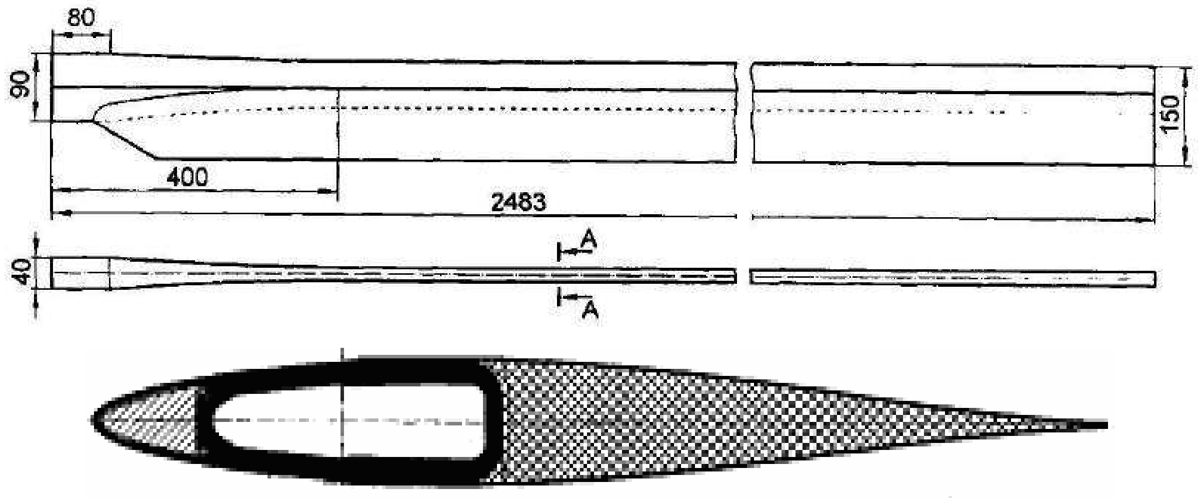


Fig. 3.31. General Layout of Helicopter Main Rotor Blade

3.18.4. Tail Rotor Blades

Tail rotor serves to balance the torque reaction and as a rudder to provide the yaw control. Since the tail rotor thrust direction of operation can change at the autorotation mode, its blades are usually flat (untwisted). Blade profile can be either symmetrical or similar to that of the main rotor. Thickness ratio of the tail rotor blade profile is calculated as

$$\bar{c}_{tr} = 0.9\bar{c}_{mr}\frac{(\omega \boldsymbol{R})_{mr}}{(\omega \boldsymbol{R})_{tr}}. \tag{3.82}$$

Tail rotor solidity ratio σ_{tr} is 1.5-2 times as big as the main rotor solidity ratio while the former has less blades. At the same time, tail rotor blades are considerably greater in relative width and harder in the plane of rotation as compared to the MR blades. Their anti-torque rotor hubs have a simpler structure, with two hinges (feathering and flapping ones).

The absence of drag hinges creates additional cyclic forces from the Coriolis forces on the blade. They come to the anti-torque rotor hub in the form of the cyclic moment, which requires a harder bedding-in of the flapping hinge.

Anti-torque rotor hub design is similar to that of the main rotor head. Ball, needle, roller, elastomeric, and metal-fluoroplastic bearings can also be widely used, and pitch bearings can be used instead of thrust ones.

Tail rotor diameter d_{tr}, its solidity σ_{tr}, and number of blades can be found approximately:

$$d_{tr} = \frac{142}{\sqrt{p\eta_{0x}N_{tr}}} \cdot \left(\frac{N_{mr}}{L_{tr}n_{mr}}\right)^{3/2}, \tag{3.83}$$

where η_{ox} is a efficiency coefficient of the tail rotor, $\eta_{0x} \approx 0.7$; N_{tr} is a power consumed; $N_{tr} \approx (0.05...0.12)N_{mr}$ (small values of N_{tr} refer to helicopters with light take-off mass); N_{mr} и n_{mr} stand for the MR capacity of rotation and frequency of rotation; L_{tr} is the distance between the main and tail rotor axes.

$$\sigma_{tr} \approx (1.7...2.3)\sigma_{mr} \approx 2\sigma_{mr} . \tag{3.84}$$

The number of tail rotor blades is usually smaller than the number of the main rotor blades. It can be approximately determined as

$$z_{tr} = \frac{2}{3} z_{mr} . \tag{3.85}$$

The moment of the Coriolis forces applied to the blade is

$$M_{CO} = 2\frac{G_{bl}}{g} r_i^2 \omega^2 \frac{\beta\beta'}{\omega} = \int_0^k r \cdot \cos \beta dk , \tag{3.86}$$

where r_i is the inertia radius of the blade; $\left(\frac{\beta\beta'}{\omega}\right)_{max} = 0.022$ for $\psi = 150°$ and $\psi = 270°$.

Coriolis forces moment can be approximately found by a formula

$$M_{CO} \approx \frac{2}{z} M_{tr} \approx 1948\frac{N_{tr}}{zn_{tr}} , \tag{3.87}$$

where N_{tr} and n_{tr} – are tail rotor capacity and frequency of rotation correspondingly.

Additional forces applied to the bail cheek and bearings of the flapping hinge are found from

$$p_{co} = \pm\frac{M_{co}}{ez} = 1948\frac{N_{tr}}{en_{tr}z} , \tag{3.88}$$

where e is the distance between bail cheeks or centres of bearing supports.

Design procedures for tail rotor blades are similar to those for the main rotor providing for specific character of its operation. Main construction differences include greater width of tail rotor blades and absence of certain sections on the blade tail section (i.e. the tail section skin is continuous).

3.18.5. Skid Landing Gear

Landing gear absorbs kinetic energy during landing and keeps the helicopter stable during taxiing and parking on the ground. It consists of a gear and a tail wheel with liquid-gas shock absorbers. Together with wheels tyres, shock absorbers absorb energy of load pulse which affects the helicopter during landing and in ground motion. Tail skid block protects tail rotor blade and tail boom during landing with a large pitch attitude. Usually shock absorbers absorb 65 ... 75 % of the kinetic impact energy, and tyres take up 25 ... 35 %.

Skid landing gear is better used in helicopters with load and takeoff weight nearing 5.000 kg as it is simple in structure, has smaller weight ($\boldsymbol{m_o}$ decreases to 1.5 ... 2 %) and aerodynamic drag. At the same time, it prevents from roll-on landing, running takeoff and taxiing which means that this type of landing gear is appropriate only for light helicopters.

Below we shall discuss the main features of skid landing gear and provide you with the algorithm of calculating shock absorbers.

Usually, helicopter skid landing gear consists of two springs and slides tightened to the ends of the springs (Fig. 3.32). Slides are made of metal pipes.

Fig. 3.32. Spring Deformation of Skid Landing Gear during Landing

In contrast to wheel landing gear which has hydro pneumatic shock absorbers, skid landing gear absorbs kinetic energy of helicopter landing due to elastic deformation, and energy dissipation is the result of slides friction force applied to landing gear on the surface of runway at transverse displacement.

Basic parameters of skid landing gear (Fig. 3.33) include:

– wheel base $\boldsymbol{c}$;
– distance between slides (track) $\boldsymbol{B}$;
– tail clearance angle $\boldsymbol{\theta}$;
– landing gear height $\boldsymbol{h}$;
– crash angles y_{f}, y_{r}, y_{s}.

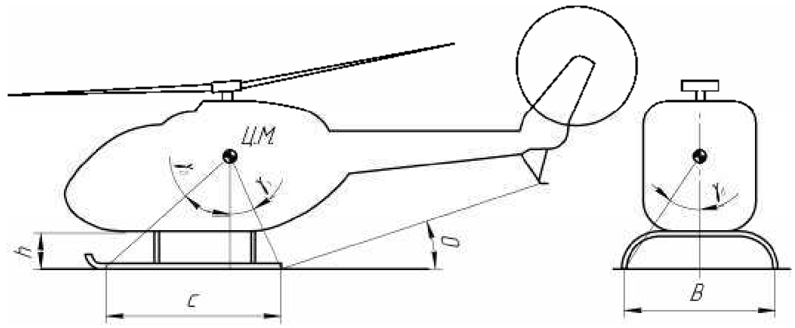


Fig. 3.33. Basic Parameters of Skid Landing Gear

Wheelbase is equal to the length of the slides' straight area limited by the front y_f and rear y_r crash angles which should be like no roll over forward or backward in any possible case. Distance ***B*** between the slides is limited by the assumed value of lateral crash angle determined by $tg\gamma_S = B / 2\gamma_{cm} \geq \mu$. According to statistical data, $\gamma_f = 50\ldots55^{\circ}$, $\gamma_r = 50\ldots55^{\circ}$ and $\gamma_S = 35\ldots40^{\circ}$. Skid landing gear should not be lower than 200 mm; its tail clearance angle is determined by conditions of the helicopter landing at autorotation.

Dimensions and side thickness of the spring as well as their distribution on the spring length rely on two factors.

1. During landing, impact kinetic energy A_l is equal to the potential energy E_p stored by the spring due to their full compression.

A_l calculations for skid landing gear are the same as for wheel LG. To find E_p, you integrate the force which arises vertically on the spring tip in the direction of spring twisting δ_S (the tip deflection) (Fig.3.34):

$$\boldsymbol{E_p} = \int_0^{\delta_{p\,max}} \boldsymbol{P_y d\delta_s} \,. \qquad (3.89)$$

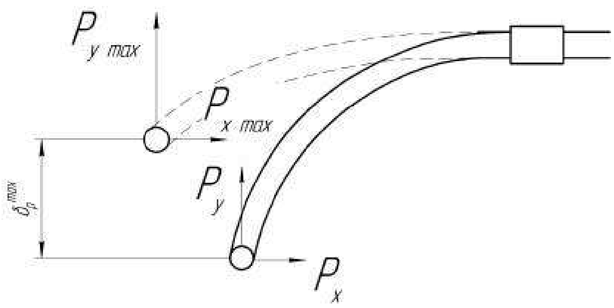


Fig. 3.34. Forces Acting on the Spring in the Course of its Compression

$\boldsymbol{A_l} = \boldsymbol{E_p}$ is the very first equation used to calculate the spring.

2. Force $\mathbf{P_y}$ increases as the spring sags and reaches its maximum $\mathbf{P_{ymax}}$ at the end of the spring travel. When $\mathbf{P_{ymax}}$, bending stresses occurring in the spring shall not exceed the breaking stresses:

$$\sigma \le [\sigma]. \tag{3.90}$$

This is the second equation of a system that calculates the spring. The allowable value of $P_{y\,max}$ is equal to the ultimate load P_u. If you know permanent spring load P_{perm}, you can determine P_u by the given limit load factor n_l and the safety coefficient $\mathbf{f} = \mathbf{1.2}$ for LG structural elements:

$$P_u = P_{perm} n_l f. \tag{3.91}$$

After you have selected the spring parameters, you must check that there are no plastic deformations in the cross section of a spring under the limit load $P_{ylimloadfactor} = P_{perm}\ n_{limloadfactor} f$ ($n_{lim\ loadfactor}$ is a limit load factor). The stresses caused by the bending inside the spring under $P_{ylimloadfactor}$ must not exceed the ultimate strength of spring material.

In real life, operation wear of the spring is usually calculated according to permanent spring deformation. Remember that if the separation of skids exceeded the allowed value, it must be replaced with a new one.

Let us consider the case when the helicopter lands on the both skids at once.

We chose PZL SW-4 as an example (Fig. 3.35).

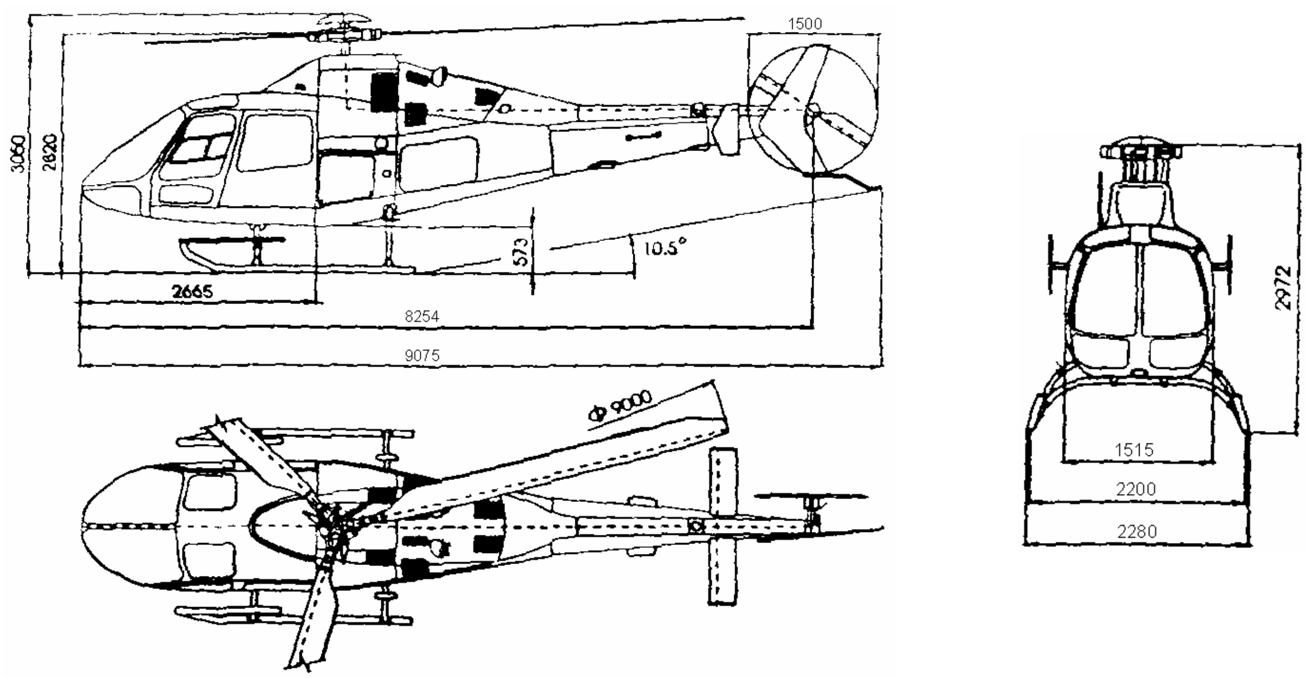


Fig. 3.35. Helicopter PZL SW-4

Initial data:

$$\mathbf{B} = 2200 \text{ mm};$$
$$\theta = 10.5°;$$
$$\delta = 200 \text{ mm}.$$

$$\text{Statistics: } \begin{cases} \gamma_f = 55°; \\ \gamma_r = 30°; \\ \gamma_s = 40°. \end{cases}$$

$\boldsymbol{m_0}$ = 1700 kg.

$\alpha = \frac{\mathbf{d}}{\mathbf{D}}$, where d and **D** are inner and outer diameter of the springs;

$$\delta_s \geq 200 \text{ mm.}$$

Diagram of full compression of spring (Fig. 3.36) and diagram of PZL SW-4 skid landing gear (Fig. 3.37) are given below.

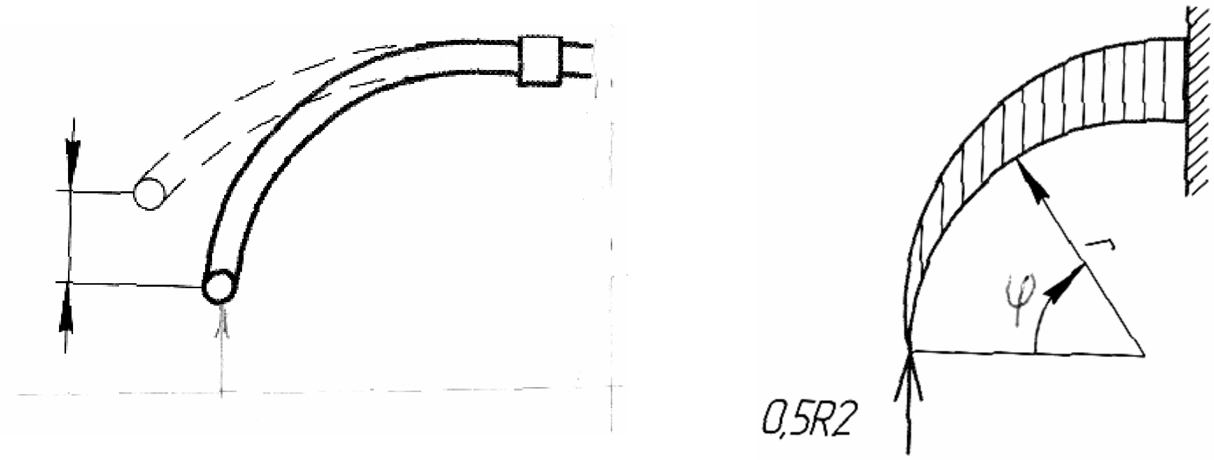


Fig. 3. 36. Full Compression of Spring

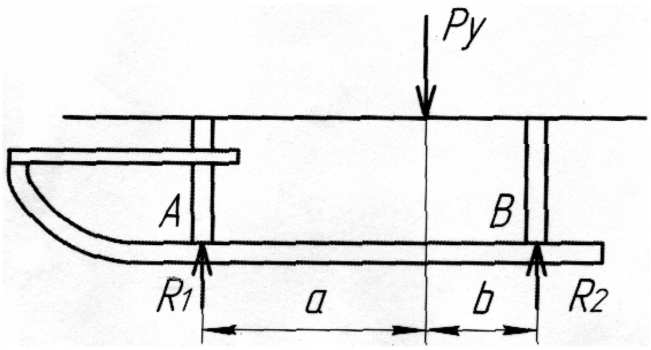


Fig. 3. 37. Diagram of PZL SW-4 Skid Landing Gear

If $\mathbf{c} = \mathbf{a} + \mathbf{b}$ = 1.1 + 0.48 = 1.58 m, $\boldsymbol{P_y = m_0 \cdot g \cdot n_э \cdot f}$ where $\boldsymbol{n}_{limloadfac\ tor}$ is a limit load factor, $\boldsymbol{n}_{limloadfac\ tor}$ = 3; **f** is a safety coefficient, **f** = 1.2.

We calculate the forces acting on one side of the spring:

$$\sum \boldsymbol{M}(\boldsymbol{A}) = \boldsymbol{R}_2(\boldsymbol{a} + \boldsymbol{b}) - \boldsymbol{P}_y \cdot \boldsymbol{a} = \boldsymbol{0};$$

$$\boldsymbol{R}_2 = 1700 \cdot 9.81 \cdot 3 \cdot 1.2 \cdot \frac{1.1}{1.1 + 0.48} = 48.764 \text{ кN};$$

$$\boldsymbol{R}_1 = \boldsymbol{P}_y - \boldsymbol{R}_2;$$

$$\boldsymbol{R}_2 = 1700 \cdot 9.81 \cdot 3 \cdot 1.2 - 48764 = 21.28 \text{ кN}.$$

Then we define the moment in the spring for $\boldsymbol{R = 0.5R_2}$.

$M = 0.5 \cdot R_2 \cdot r(1 - \cos\varphi)$; φ = 90°; **M** = 17.0674 кNm;

$$\sigma = \frac{M}{W},$$

where W is a section modulus.

For a hollow tube,

$$W = \frac{\pi \cdot D^3}{32(1-\alpha^4)}.$$

We accept α equals 0.9,

$$\frac{M \cdot 32(1-\alpha^4)}{\pi \cdot D^3} \le [\sigma];$$

$$D = \sqrt[3]{\frac{M \cdot 32(1-\alpha^4)}{\pi \cdot [\sigma]}};$$

$$D = \sqrt[3]{\frac{17067.4 \cdot 32 \cdot (1-0{,}9^4)}{3.14 \cdot 800}} = 4.213 \text{ mm.}$$

Tube dimensions are D = 45 mm, side thickness is 2 mm.

3.18.6. Control

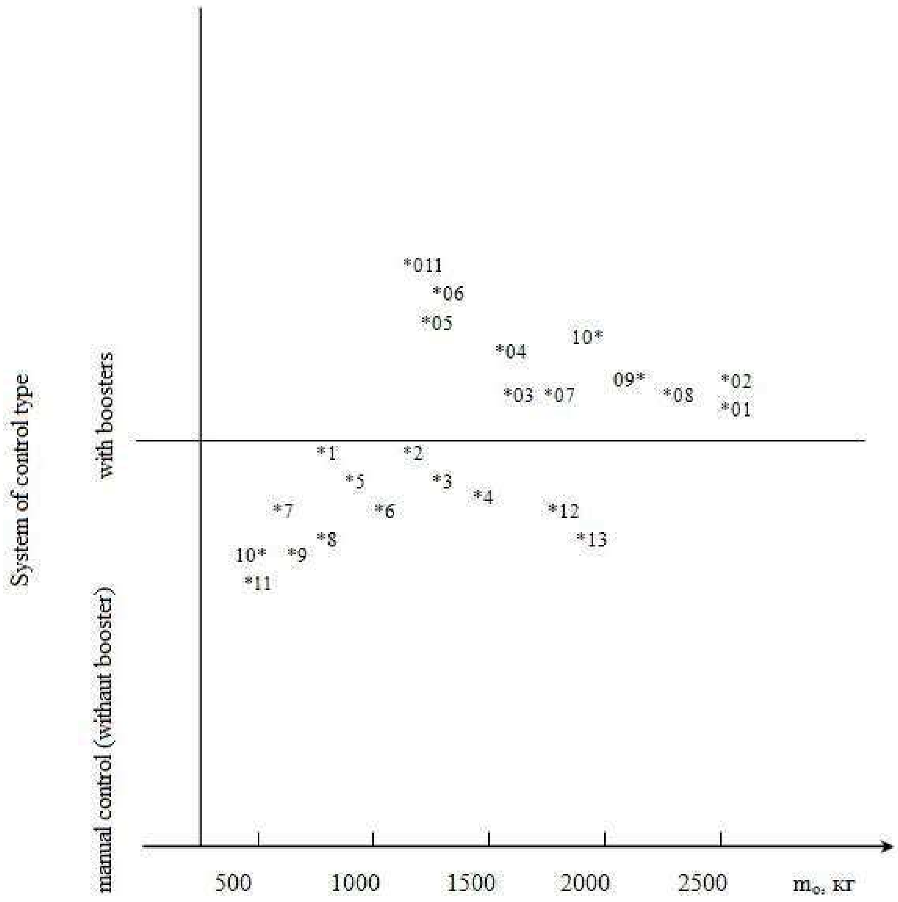
In helicopters, pressure in the flight control channel of flight controls does not exceed the standards which permit only manual control (control linkage from knobs and pedals to flight controls). This means that they do not need power control (main actuators with their attachments, control system from these actuators to a swash plate and main hydraulic system, etc.).

Power control requires not only low pressure in the control channels but also hydraulic system, hydraulic actuators, heavier structural elements, etc.

Statistical analysis provides us with certain recommendation on to develop the control for ultralight helicopters keeping in mind their mass that influences the pressure in control channels.

Manual unpowered control is recommended for helicopters with the takeoff mass under $m_0 = 1500$ kg: Otherwise, power control is required. Obtained results are approximate and can be applied only at the stage of preliminary design (Fig. 3.38).

Final decision on the type of control is made only after actual loads in control channels are calculated and they do not exceed the standard ones.



Helicopters:

01 – Bo.105CBS;
02 – Bo.105LS;
03 – EC.135;
04 – OH-58D;
05 – MD500E;
06 – MD530F;
07 – SA/315B "Lama" ($m_{п.н.}$=1950 *кг*);
08 – SA315B "Lama" ($m_{п.н.}$=2300 *кг*);
09 – AS316B " Aluette ' III;
010 – AS.342M "Gazella";
011 – RH.1100.

Helicopters:

1 – R.22 "Beta";
2 – R.44;
3 – Enstrom 280FX;
4 – Enstrom 480;
5 – 300C;
6 - 330SP;
7 – Ultrasport 496;
8 – Exec 162F;
9 – AKI-3;
10 – Dragon 334;
11 – Masquito M80;
12 – MD52ON;
13 – MD600N.

Fig. 3. 38. Control System Types

3.18.7. Transmission

Until 1950s, piston engines were the main element of the rotor power drive as there were no helicopter gas turbine engines. Over time, gas turbine engines exceeded piston ones, first and foremost in unit weight and capacity.

Today, piston engines are mostly used in ultralight helicopters. There are several reasons for this, proven by statistical data:

1. There are no mass-produced low-capacity gas turbine engines (there are little to none gas turbine engines with capacity under 250 kW) [10]).
2. Piston engines have relatively low transmission rating.
3. Piston engines have relatively low reduction rating due to low rotation speed that enables belt transmission.

Despite clear advantages of using gas turbine engines for helicopters with the required power under 250 kW, piston engines are preferred with two types of transmission: combined (gear + belt) or gear transmission.

Combined transmission and gear transmission differ by their mass. This difference can be crucial for ultralight helicopters which can lose such specific properties as hovering and vertical takeoff because of overweight.

When choosing the type of transmission, helicopter designer compares masses of known transmission layouts and considers whether they meet the layout requirements.

There are many layout options which cannot all be reviewed here. We shall consider the procedure of choosing the transmission among different layouts with the same design factor.

It is clear that to choose the transmission type, you must formalize each layout by its mass. The reasonable objective function is mass $\mathbf{m_p}$ of layout elements with the parameters of engine-to-drive-axes distance (layout requirement) and equal reduction rate. Preferred will be the transmission of smaller mass.

When formalizing layout design, remember this:

– V-belt drive overall dimensions is determined by the axe-to-axe distance **a**,

$$a_{min} = 0.55(d_1 + d_2) + T_0, \qquad (3.92)$$

$$a_{max} = d_1 + d_2, \qquad (3.93)$$

where $\mathbf{d_1}$ and $\mathbf{d_2}$ are pulleys pitch diameters; $\mathbf{T_0}$ is a belt cross-sectional height;

– the ratio of the bevel gear speed reducer is $\mathbf{u} = 1\ldots6.3,$ but if it should not be applied at $\mathbf{u} > \mathbf{5}$;

– for geometrical and design reasons, cogwheels must have no less than 10-13 and no more than 100-130 cogs (use series of gear pairs to transmit the rotation that exceeds these limits);

– get a V-belt transmission of a bigger gear ratio (up to 7 or even 10);

– consider the reducer frame side thickness to be 5 mm.

If both transmission layouts have bevel gear speed reducers of the same design, you choose the transmission layout depending on masses of belts and cylindrical gear drives. This does not mean though that real helicopters have separate main reduction gears and bevel gear speed reducers.

To calculate the transmission mass with belt and gear transmission for superlight helicopters, follow this algorithm.

Deadload belt transmission $\boldsymbol{m_{b.t}}$ is conventionally a sum of components:

$$M_{b.t} = m_p + m_b + m_s + m_{t.m} + m_c + m_f, \tag{3.94}$$

where $m_p + m_b + m_s + m_{t.m} + m_c + m_f$ are masses of pulleys, belts, shafts, tensioning mechanism, cheeks, and fittings correspondingly.

$$m_b = L_b \cdot z_b \cdot q, \tag{3.95}$$

where $L_{\mathbf{b}}$ is a belt length at the neutral line, m; z_b is a number of belts, pcs; q is a mass of belt per metre, kg/m.

$$m_b = \boldsymbol{L_p} \approx 2\boldsymbol{\alpha} + \boldsymbol{\pi d_{cp}} + \frac{\boldsymbol{\Delta}^2}{\boldsymbol{a}}; \ \boldsymbol{\Delta} = \frac{\boldsymbol{d}_2 - \boldsymbol{d}_1}{2}; \ \boldsymbol{d_{cp}} = \frac{\boldsymbol{d}_2 + \boldsymbol{d}_1}{2}. \tag{3.96}$$

For a V-belt drive, angle of contact on the small pulley is $\alpha_1 = 120^\circ$; however, transmission operation is reliable enough at the angle of 90°.

3.18.8. Transmission without Intermediate Gearbox

Ultralight helicopter may have no intermediate gearbox because of the small thrusting torque moment of the main rotor relative to the tail rotor axis ($M_{\mathbf{TR}}$) and the fact that it is installed on the tail. Approximately, thrusting torque moment

$$T_{TR} \approx (\mathit{0.06 \ldots 0.07}) \cdot G_o, \tag{3.97}$$

where $\mathbf{T_{TR}}$ is the thrust of antitorque (tail) propeller; G_o is a helicopter takeoff weight (Fig. 3.39, Table 3.15).

The moment is estimated on the basis of statistical data, with the scale obtained by measuring the shoulder between the axes of both tail rotor and tail boom for each helicopter. Leverage shoulder is measured from disc plane of MR to the tail boom axis. Results for $T_{TR} \approx \mathit{0.065} \cdot G_o$ are shown in Table 3.15.

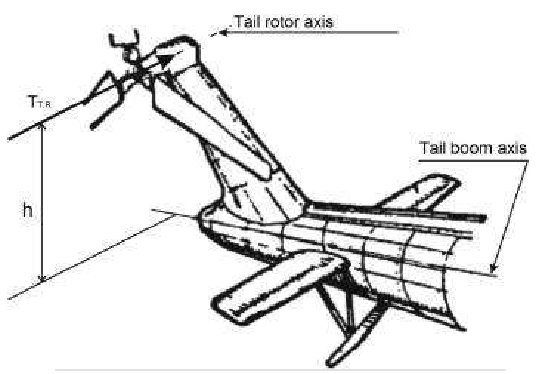


Fig. 3.39. Thrust Moment of Tail Rotor relative to Tail Boom Axis

Table 3.15

Statistical Data on Characteristics of Helicopter Tail Rotor

Helicopters	Statistical Data on Tail Rotor			
	G_o, *H*	T_{tr}, *H*	h, *m*	$M_{tr}=T_{tr}\cdot h$, *H·m*
CH-7 "Angel"	3531.6	229.5	0.96	220.3
Mini 500	3433.5	223.2	0.726	162.04
M-80 "Masquito"	4414.5	286.9	0.96	275.4
"Dragon Fly 333"	4414.5	286.9	0.83	238.1
R-44	10673.3	693.8	1.0	693.8
"Exec 162F"	6670.8	433.6	0.94	407.6
Hughes 269A	7426.2	482.7	0.84	405.5
Schweizer 300CB	7779.3	505.6	1.18	596.6
Enstrom F-28A	11566	751.1	0.98	736.1
Aktai	10300.5	669.5	1.2	803.4
Mi-34	12556.8	816.2	1.17	954.9
Hughes 500D	13341.6	867.2	1.01	875.9

3.18.9. Power Plant with Vertical Arrangement of Engine Cranked Shaft Axis

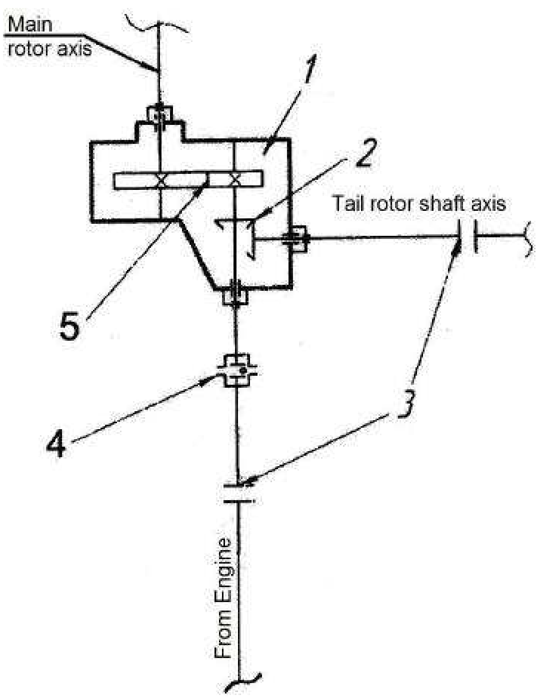


Fig. 3.40. Main Gearbox with Vertical Engine Input to Main Rotor:
1 – casing; 2 – gear (taper) transmission; 3 – disc clutch; 4 – overrun clutch; 5 – gear (cylindrical) transfer

Ultralight helicopters may employ piston engines with a vertical axis of the engine crankshaft. It essentially simplifies helicopter transmission by bringing rotary gear functions to zero, reducing transmission weight and volume, and moving the helicopter's centre of mass down. This positively influences balancing and controllability of the helicopter though makes lubrication and tightness of the main gearbox much more complicated.

Main gearbox of an ultralight helicopter with vertical engine input to the main rotor is shown in Fig. 3.40.

3.19. Layout Diagram and General Arrangement of a Helicopter

3.19.1. Helicopter Layout

General arrangement of a helicopter includes mutual space linkage of its parts, their shapes and load-carrying structures together with the arrangement of engines, crew, basic cargoes and operating equipment.

Arrangement is a complex three interrelated processes performed at the same time: aerodynamic, weight space, and structure-force configurations.

Aerodynamic configuration is intended to identify shape, dimensions, and relative position of helicopter parts, surrounded by air flow.

Weight space arrangement aims at distributing the space and arranging assemblies and helicopter parts within the expected contours as well as at collecting balance data for calculation of the helicopter operating load.

Structure-force arrangement is supposed to create such a load-bearing structure of the helicopter that will provide the required capacity, strength and stiffness of construction, manufacturability and operational ergonomics as well as ensure minimal deadload.

Helicopter arrangement must meet the following requirements:

1. Helicopter provides enough space to arrange cargo and passengers.
2. Passengers and crew fell convenient on-board.
3. Flight compartment (cockpit) secures good vision.
4. Helicopter is easily escapable if crashed.
5. Cargo can be loaded and unloaded rapidly and is lashed safely;
6. Equipment can be easily accessed, installed, and removed.
7. Helicopter arrangement secures necessary level of communication between flight and passenger compartments (in multi-seated helicopters).
8. Overall dimensions can be reduces by removing or folding the blades and the TR tail boom when the helicopter is in limited space.
9. Engine and transmission accessories are arranged so that to secure easy installation and removal as well as convenient inspection.
10. Oil tanks and coolers are close to the engine.
11. Helicopter shapes and extensions comply with aerodynamics requirements.
12. Fuel tanks are installed within allowable limits in regard to the helicopter centre of mass.
13. Payload is located near the centre of mass so as to secure centre of gravity position in the prescribed range.

The place for the system of cargo lashing on the external load sling or dropped cargoes is near the helicopter centre of mass.

Iterative procedure is used to obtain the required centre of gravity from several variants of helicopter arrangement.

Before helicopter designers start the arranging procedures, they must get engine installation drawings as well as drawings which show overall dimensions of the cargo and dedicated equipment with centres of gravity plotted on them. It is also desirable to know approximate overall dimensions of the main rotor gearbox and other transmission accessories. Helicopter arrangement depends on its diagram, function, type, the number of engines and many other factors.

When you develop the layout scheme, pay particular attention to the view from the cockpit (is usually regulated by statutory documents). The view should provide an opportunity for a helicopter pilot to operate in all phases of flight. View from the cockpit is the important feature of the pilot’s workplace that determines flight safety and target achievement.

Helicopter layout is represented as a drawing or a 3D-space allocation model. The drawing can be done in electronic form in the 1:1 S scale or on a sheet of drawing paper (at least A1 or A2). The drawing must depict a helicopter longitudinal section view and planform view as well as cross-section in the places where cargo (passenger) cabin is located or engine and the main gearbox is mounted. The drawing must show location of cargoes, passengers and the crew; all structural members of the framework and contours of the basic units; attachment scheme of the load-bearing frame. When included in the general layout drawing, construction parts must be sized according to the designed standard helicopter size.

Both layout drawing and helicopter main view (three projections) are the basis for the lines drawing plan of the fuselage and its connections with the other helicopter parts at the draft stage of design.

3.19.2. Centre of Gravity

Helicopter centre of gravity is calculated after the tolerated error of takeoff weight calculation was reached, helicopter parameters and characteristics were selected, and its effectiveness was evaluated as well as the engine efficiency.

The desired position of the helicopter centre of mass (CM) at all flight conditions is found based on the centre of gravity. Centre of gravity position is calculated for three cases at the very list, namely for takeoff with gross load, landing with 5 % fuel reserve on board, and parking of empty helicopter (Table 3.16):

$$\mathbf{x}_0 = \frac{\sum_i \mathbf{m}_i \cdot \mathbf{x}_i}{\sum_i \mathbf{m}_i}; \qquad \mathbf{y}_0 = \frac{\sum_i \mathbf{m}_i \cdot \mathbf{y}_i}{\sum_i \mathbf{m}_i}. \tag{3.98}$$

Table 3.16

Weight Report

Name of Assembly	Weight $\mathbf{m_i}$, kg	CG Position on $\mathbf{x}$ Axis, m	Static Moment $\mathbf{m_i \cdot x_i}$, kg·m	CG Position on $\mathbf{y}$ Axis, m	Static Moment $\mathbf{m_i \cdot y_i}$, kg·m
1. Main Rotor: - blades, - head					
2. Steering System: - booster, - manual					
3. Transmission: - main gearbox, - intermediate gearbox, - transmission shaft					
4. Tail Rotor: - blades, - head					
5. Power plant					
6. Fuel system					
7. Fuselage: - nose section, - centre section, - rear section, - gearbox attachment, - cowl panels					
8. Landing Gear: - main, - nose, - tail gear					
9. Electrical Equipment					
10. Equipment: - cabin instruments, - radio equipment, - hydraulic equipment, - pneumatic, - auxiliary					
	$\sum \mathbf{m_i}$		$\sum \mathbf{m_i x_i}$		$\sum \mathbf{m_i y_i}$

Centre of gravity is often calculated by the φ_{CM} angle to that of the main rotor axis and the line connecting the main rotor centre head with helicopter CM:

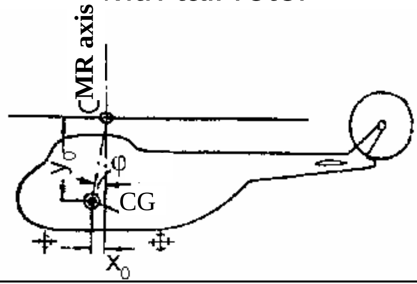
$$tg\ \varphi_{CM} = \frac{x_0}{y_0}. \tag{3.99}$$

The centre of gravity calculation is a part of the body-mass layout as well as related publications (helicopter drawing scale, centring sheet, etc.).

Table 3.17 provides CM positions for cases of helicopter normal, maximum permissible forward and aft centre of gravity.

Table 3.17

CG Position of Single-Rotor Helicopter

Configuration of Helicopter	CG Position			
	CM Angle	Normal	Allowable Limit	
			Forward	Aft
Single-propeller with tail rotor 	φ_{cm}	-3°	-6° with stabilizer $S / \pi R^2 = 0{,}004$ (at $\frac{y_0}{D} = 0.105...0.125$)	2°

3.19.3. General View of Helicopter

General view helps helicopter designers to interlink main helicopter sizes and set its external forms. General view is necessary to produce workshop drawings of helicopter parts and model drawings for aerodynamic researches. Helicopter drawing or 3D space allocation model is required for locking device and centre-of-gravity positioning, aerodynamic design, stability and controllability analysis, etc. in turn, their results are used to revise the helicopter general view and put additions into its drawing.

At the stage of preliminary design, helicopter general view is based on its main parameters and geometrical sizes by taking into account initial data and world experience in helicopter design.

Helicopter general view is drawn in three projections on A2 paper (ГОСТ 2.301-68 (ST SEV 1181-78)) applying overall and most typical sizes.

The right corner of the drawing (in the table above the stamp) indicates main parameters of the helicopter under design:

– takeoff mass, kg;
– payload, kg;
– maximal flight speed at the altitude of **H** = 500 m, km/h;
– maximal flight altitude, m;
– stabilized ceiling, m;
– course flight speed at the altitude ***H*** = 500 m, km/h;

– economic flight speed at the altitude ***H*** = 500 m, km/h;
– flying range with regular filling at the altitude ***H*** = …m, km;
– mission time at the altitude ***H*** = 500 m, h;
– engine type;
– maximum engine power, kW;
– course engine power, kW.

Drawings of the helicopter general view must be performed according to the requirements of Unified system for design documentation and national standards. They are supported by layout and text design documentation which provide necessary analysis and decision grounds.

3.20. Helicopter Layout

Fig. 3.41 provides an example of helicopter layout [29].

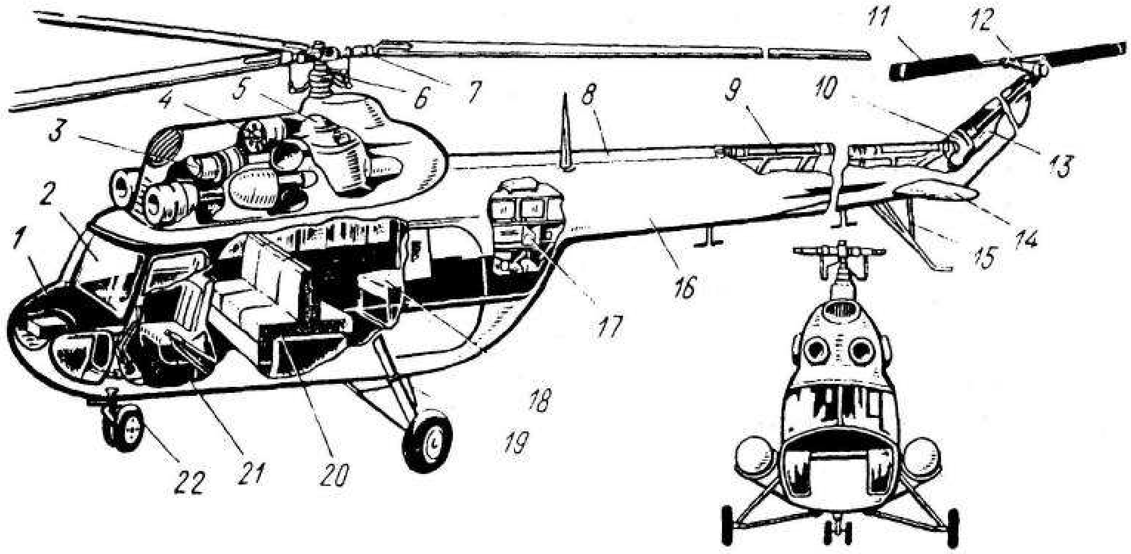


Fig. 3.41. Layout Diagram of Light Helicopter:
1 – battery compartment; 2 – front transparency; 3 – engine compartment; 4 – fan system; 5 – main gearbox; 6 – swash plate; 7 – main rotor; 8 – casing of tail rotor drive shaft; 9 - tail rotor drive shaft; 10 – accessory gearbox; 11 – tail rotor; 12 – tail gearbox; 13 – tail boom pylon; 14 – horizontal stabilizer; 15 – tail unit; 16 – tail boom; 17 – avionics compartment; 18 – back passenger seat; 19 – main landing gear; 20 – passenger bench-type seat; 21 – pilot seat; 22 – nose landing gear

The helicopter (Fig. 3.41) comprises the following main structural members and systems: fuselage, landing gear, pneumatic system, power unit, transmission gearbox, main rotor, tail rotor, deicing system, control system, hydraulic system, heating and ventilation system, rigging mooring equipment, furnishings, aeronautic equipment, and avionics.

Helicopter fuselage consists of nose and central parts, tail boom, and tail boom pylon. Cockpit is situated in the nose part, cargo-passenger cabin is in the central part. Control levers, cockpit panel, electrical control panels, pilot seat are in the nose section. The cockpit is entered through the front door on the right side of the helicopter; sliding blister is on the left nose side. Cockpit door and blister are equipped with emergency release mechanisms. Cockpit transmittance provides enough surveillance in front of the helicopter. Lidded container (upfront the nose part) hosts two 12–CAM–28 storage batteries.

Cargo-passenger compartment occupies the whole central part of the fuselage and is equipped depending on the helicopter design purpose. The compartment is furnished with a bench-type seat with a combined back for six passengers as well as a one-type seat on the back right side (in aeromedical configuration, it is replaced by a table and a seat for medical attendants). Behind the frame, there is an avionics compartment accessible through the door in the wall of this frame. Central part of the fuselage has rectangular windows on both sides and back entrance door on the left side. At the bottom of the central part of the fuselage, main fuel reservoir is installed in a special container.

A tail boom is attached to the central part. A transmission tail rotor drive shaft extends over the tail boom, covered with an easy-removable cover unit. Tail boom hosts a horizontal stabilizer, tail unit, and radio stations antennae. The boom ends with a device to which a tail boom pylon is attached through an accessory gearbox. Tail gearbox is installed on the flange of its rear part.

Centre and main non-retractable landing gears are attached below the fuselage nose and centre sections. Landing gears and skid block are fitted with gas-fluid shock struts. Nose landing gear has a centring cylinder; main gear's wheels are fitted with break gears which are driven by pneumatic system.

Two turboshaft engines ГТД-350 are located above the cargo-passenger compartment as well as main gear box BP-2, accessories to an air-cooling system, accessories and communication lines of a hydraulic system, oil system, fire-protection system, and helicopter control systems. Such arrangement makes it possible to considerably increase cargo-passenger compartment. Two external fuel tanks are installed on the helicopter sides to support long-distance flights.

Power comes from the engines to main and tail rotors and accessories are driven with the help of the rotor drive system consisting of main gearbox, intermediate gearbox, tail gearbox, two main drive shafts, tail rotor drive shaft, and main-rotor brake. Main gearbox transmits drive to the main rotor, antitorque rotor drive, fan layout, hydraulic unit, AC generator, volumetric compressor, and other accessories. Torque effect is transmitted to the tail rotor through the tail rotor drive shaft, intermediate and tail gearboxes. In order to secure autorotation mode in case of the engine failure, main rotor is fitted with freewheel clutches that disconnect it from the engines. Engines, main rotor, and accessories are shut down with a cowl panel which provides for the rounded form of the dorsal section.

Main rotor consists of a hub and three all-metal blades. Hub is fitted up with spaced-apart flapping, drag and feathering hinges as well as hydraulic dampers. Blades have an annunciator system of spar crash and electric anti-icing. Tail rotor is two-bladed, pushing, with controllable pitch in flight. Rotor hub has a common flapping hinge and feathering hinges individual for each blade. Blades are supplied with heaters of electrothermal action.

To prevent icing of MR and TR blades, left sight glass of a cockpit thermoelectric anti-icing system is provided.

Helicopter control consists of fore-aft (arm) actuation, directional (foot pedal) control, combined control (collective pitch – horizontal stabilizer), decoupled control of engines, MR brake control, and engine stoppage. Collective and cyclic pitches of the main rotor are changed by a swash plate that is installed on a MR transmission, and a pitch of a tail rotor is changed by pitch-change actuator installed in a tail gearbox.

Helicopter has a hydraulic system provided for hydraulic boosters which take part in lateral-longitudinal and collective pitch control of the main rotor. Hydraulic boosters are installed on the irreversible scheme and take all loads from the rotor aerodynamic forces that are transmitted to the control. The lateral-longitudinal control system includes feel mechanisms that create required gradient of the effort on the cyclic pitch handle and allow balancing these efforts at normal operation.

Heating and ventilation systems supply cockpit and cabin with heated or cold air thus maintaining normal temperature and air quality in the helicopter.

Finally, helicopter has instrumentation system, avionics, and accessories equipment which secure its operation daytime and night under all weather conditions.

3.21. Main Helicopter Development and Usage Trends

Growing demand for helicopters is explained by the following factors:

– helicopters are able to perform a slow controlled vertical flight;

– currently improvements in production technology can significantly improve performance and reliability of helicopters (flight safety, pilotage relief through the use of advanced main rotors and transmissions, new vibration reduction systems, stability enhancement systems, etc.);

– many possible application fields.

1. Newest Technology of Main Rotor Design

Resent years saw improvements in the field of main rotor design, including aerodynamics, blades and heads, materials, etc.

New aerodynamic profiles are developed based on the vortex theory. Aerodynamic configurations of blades are based on their operation conditions.

Modern computing methods studied interrelations between elastic and mass characteristics; between blade configurations and flight modes and loads. This increased the main rotor quality from 6 to 8…9 and kept hinge moments with the 1.5-times increase in flight speed.

Composite materials were developed which provide necessary characteristics, geometry fidelity of profile section, and the quality of blade surface as well as determine blade production technology (filament-wound method or pressing method in the negative matrix). Blades made of composite materials have considerably higher levels of reliability and survivability in case of the rotor damage or failure. Moreover, composite blades have greater resources: Glass fibre blades mounted on Ka-26 have flown 2.5 millions of hours by the middle of 1984 and still don't have any resource constraints. It was proved experimentally that the thrust of the main rotor blades made of polymeric composite materials grows by 10 % compared to metallic ones.

Potential capacities of composite materials found their use in main rotor hubs, control rods, and swash plate assemblies. An opportunity has appeared to switch to the condition-based maintenance.

Current progress in MR hubs design and manufacturing technologies brought out hingeless hubs and hubs with hinges which do not require lubricating oils. In its turn, simplified hub design reduced the parasite drag of the main-rotor-bearing-to-hub chain which can today reach up to 30 % of the total helicopter drag.

2. Reduction of Non-Bearing Elements Parasite Drag

Research shows that today it becomes possible to reduce parasite drag of helicopter non-bearing elements by 3.5 to 4 times. Combined with improved main rotors, this will increase course speeds to about 350 km per hour in future.

3. Composite Materials in Helicopter Frame

Composite materials cannot be employed everywhere in a helicopter because they accumulate static electricity, are transparent for radio waves, pose danger at lightning stroke, lack common electric potential between different structural members, etc. However, Russia has already designed helicopters in which composite materials make up to 35 % of airframe weight.

4. Engine Technology

Historically aircraft progress meant that airplanes and helicopters were developed of large takeoff mass and dimensions, designed to carry a lot of passengers and cargoes. Aircraft designers strived to improve expendable, specific and mass characteristics of the engines, their electronics and automation.

Today engine designers aim at building light and fuel-efficient engines producing dozens to hundreds kilowatt. If they succeed, helicopter designers would never be forced to opt for non-optimal engines or adapt helicopters for the available engines.

5. Electronic Systems

All-weather helicopters and helicopter airlines will result in new compartments which will reduce pilot workload. These compartments will be equipped with microprocessors, cathode-ray indicators, etc.

Standard helicopter equipment will include systems of blind flight, automatic landing. These systems will allow the crew to work confidently even in the areas which do not have good navigational devices. First such systems have already been created and installed on Ka-32.

6. Lower Operational Costs

Helicopter customers call for not only higher efficiency and lower initial costs but also lower operational costs. However, the increased quality of helicopters does not pull down their costs.

Only simple construction, profound failure prevention, new maintenance procedures can decrease operational costs for helicopters of any weight.

7. Structural Changes in Helicopter Types

Modern helicopters are characterized by:

– higher flight speeds due to high-qualified rotors with the higher capacity of a power plant at the increased speed and drag;

– greater relative payload mass due to improved rotor in hovering and lower mass of an empty helicopter due to new structural materials;

– lower kilometric fuel consumption due to improved airflow of the fuselage and lower specific engine fuel consumption;

– greater flight task efficiency, increased survivability and reliability due to the use of composite materials;

– smaller pilot workload due integrated application of electronics;

– smaller flight and maintenance staff.

With power plants of the same capacity, helicopter payload depends on the ratio of structural mass to takeoff mass and on the efficiency coefficient of a rotor. Structural mass of the first helicopters was relatively large. Since the first serial helicopter has been manufactured, the ratio of their payload to the structural mass increased from 0.55 (Alouette provided with piston engine) to

0.75 (Alouette II and SA.341 Gazelle provided with turboprop engine Turbomeca Astazou XIV).

Today the ratio of a helicopter payload mass to its structural mass is about 1.5.

Maximal speed can grow only at the cost of more powerful engines mounted on the helicopter and, thus, greater fuel consumption. The increase in helicopter maximal speed without increased engine power can be achieved by decreasing the parasitic drag as 45 % of the maximal drag power is spent on its elimination. Main rotor hub and landing gear have the highest drag (about 50 % of the parasitic drag).

Important construction problems include helicopter-generated noise (for the ICAO regulations, (Fig. 3.42) [18]).

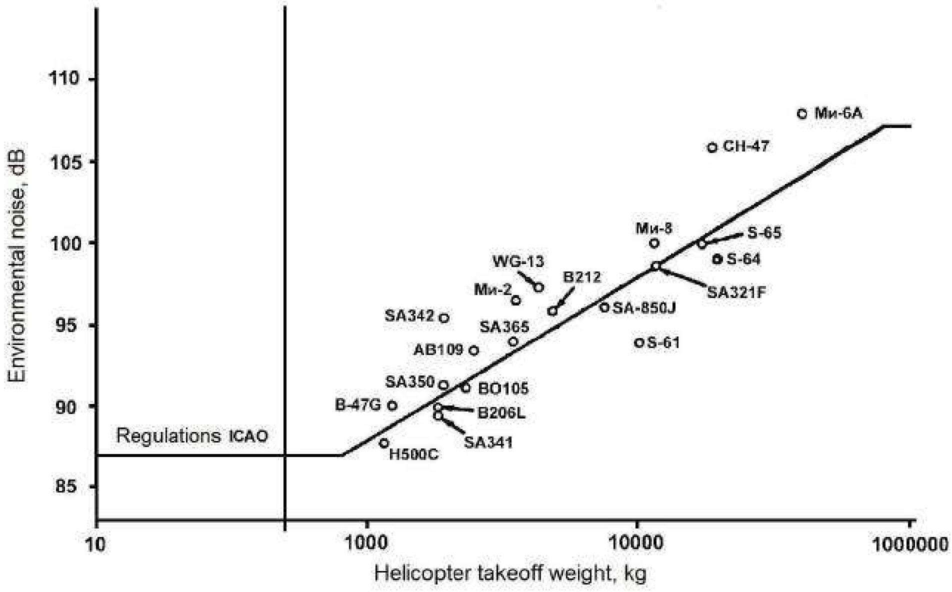


Fig. 3.42. ICAO Noise Regulations

Noise generated by the helicopter is measured both inside and outside the aircraft. Inner noise level is regulated by empiric standards considering pilots' comfort and their physiological factors, possible oral communication, etc. Outer noise level is regulated by normative documents because helicopters are used for transport needs in densely populated regions (e.g., as an air-taxi in megapolises).

Blade rotational noise usually sounds like dim beats that can turn into distinct claps. It is generated by periodic blades force impact on air in each of the fixed propeller disk points due to the lift force and drag force rotating with the blades. The noise range depends on the blade geometry and propeller mode of operation. Rotational noise mostly fits in the low-frequency spectral region (up to 150 Hz) with the upper limit of 10-20 Hz. Heightened ultraharmonics can turn this noise into blade slaps if there is small number of blades and high circular velocity. Rotational noise can cause frame vibrations and fatigue damage. Besides, low-frequency noise spreads easily in the atmosphere. Thus, rotational noise and blade slaps will be crucial at the longer distances from the helicopter.

In respect to their speed, flight range and altitude, helicopters are considerably inferior to modern airplanes. But helicopters do not compete with aeroplanes due to their specific appearance, purpose, and steadily growing scope of application (Fig. 3.43).

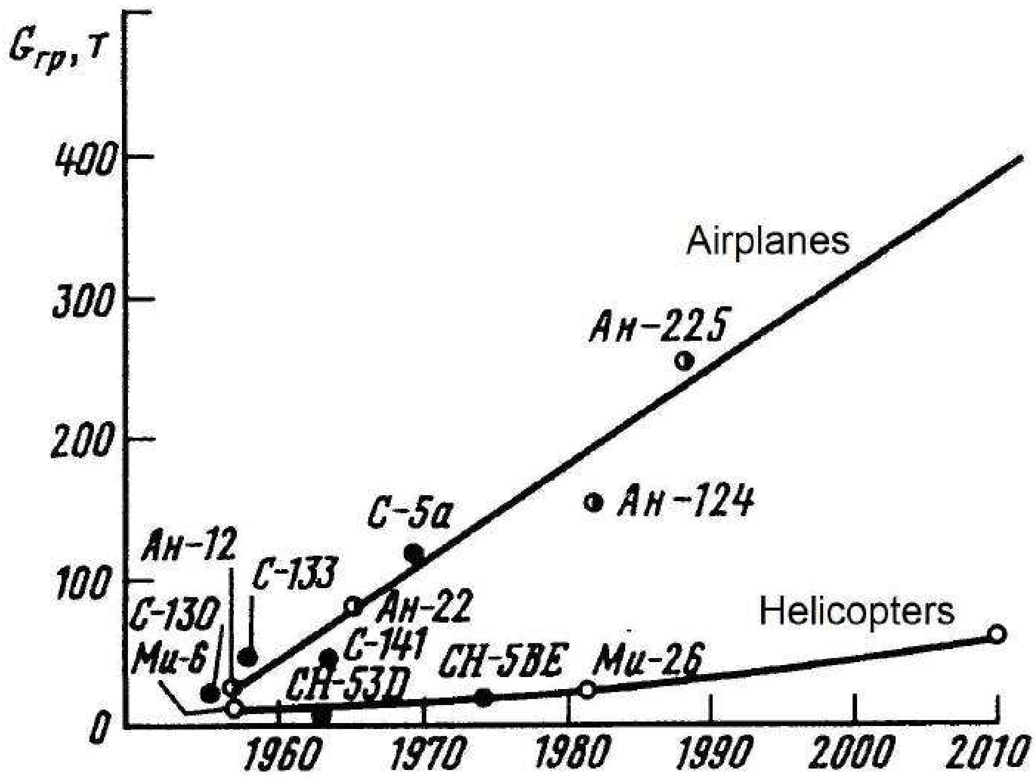


Fig. 3.43. Heavy-Payload Airplane and Helicopter Development Over the Years

4. EXAMPLES OF HELICOPTER PARAMETERS SELECTION

Helicopter parameters (4.1) – (4.4) are chosen with regard to a minimum criterion of the takeoff mass. We based our selection on the rules provided in Chapter 3 according to weight categories of Table 1.3. Calculation results are in Tables 4.1, 4.2, 4.3 and 4.4.

We verified our selection of helicopter parameters by examples of Robinson R-22 (USA), Swidnik SW-4 (Poland), MI-2 (Russia), and MI-8 (Russia). Paragraph 4.2 goes for a light helicopter.

We took parameters of real helicopters as our initial data. Low differences in calculated and actual parameter values are specified by range and other coefficients.

4.1. Selecting Parameters for Light Helicopter with Single Rotor

4.1.1. Initial Data

1. $\boldsymbol{m}_{cr}$ = 80 kg – crew mass.
2. $\boldsymbol{m}_{pl}$ = 115 kg – payload mass.
3. $\mathbf{V_{max}}$ = 190 km/h – maximal flight speed.
4. $\mathbf{L_{max}}$ = 320 km – maximal range.
5. $\boldsymbol{H}_{hov.}$ = 1586 m – hovering ceiling.
6. $\boldsymbol{H}_{dyn.}$ = 4265 m – dynamic ceiling.
7. $\boldsymbol{m}_{equip}$ = 43 kg – mass of equipment.
8. $\bar{\boldsymbol{m}}_{con.} = \bar{\boldsymbol{m}}_{man.\ con.}$.
9. Weight category – light communications.

4.1.2. Parameter Selection Completing Initial Data

1. Statistical table includes Brantly B-28, Dragon Fly 333, Exec 162F, Hughes 269A, Us496.
2. Airfoil portion NACA 23012 is taken for rotor blades.
3. Tip speed ωR is taken equal to 184 m/s.
4. $\mathbf{m}_{\boldsymbol{00}}$ = 590.9 kg is taken at zero approximation of the takeoff mass.
5. $\mathbf{p_1}$ = 190, $\mathbf{p_2}$ = 215, $\mathbf{p_3}$ = 240, $\mathbf{p_4}$ = 265, $\mathbf{p_5}$ = 290 N/m^2 are taken as a variability range of the specific load $\mathbf{p}$:.
6. Takeoff mass at zero approximation $\boldsymbol{m}_{takeoff}$ is determined on the basis of the statistical data by the formula

$$\boldsymbol{m}_0^0 = \frac{\boldsymbol{m}_{pl} + \boldsymbol{m}_{cr}}{\bar{\boldsymbol{K}}_{full\ \ l} - \bar{\boldsymbol{m}}_{fuel}}, \qquad (4.1)$$

were $\bar{K}_{full\ l}$ is a full load coefficient determined through the average value of empty helicopter mass ratio $\bar{m}_{emp}$ = 0.562 according to statistical sampling:

$$\bar{K}_{full\ l} = \frac{m_0 - m_{emp}}{m_0} = 1 - \bar{m}_{emp} = 1 - 0.562 = 0.438. \tag{4.2}$$

Specific fuel mass is determined by the statistical data,

$$\bar{m}_{fuel} = \bar{q}_{fuel} \cdot L + 0.33 \cdot \bar{Q}_V = 0.275 \cdot 320 + 0.33 \cdot 0.061 = 0.108. \tag{4.3}$$

Calculations are summarized in Table 4.1.

Takeoff mass $\mathbf{m_0}$ = 617.324 kg in second approximation, and the error is

$$\xi = \frac{|617.324 - 613|}{613} \cdot 100 = 0.705\ \%.$$

Chosen parameters and characteristics are given in Tables 4.1 and 4.2.

The helicopter under design has a single reciprocating engine Rotor Way R1162 of 113 kW (152 HP) (the same as in Rotor Way Exec 162F).

4.1.3. Conclusions

Our helicopter is close to Robinson R-22 (Fig. 4.1) in its tactical-technical requirements, performance, mass characteristics, and dimensions.

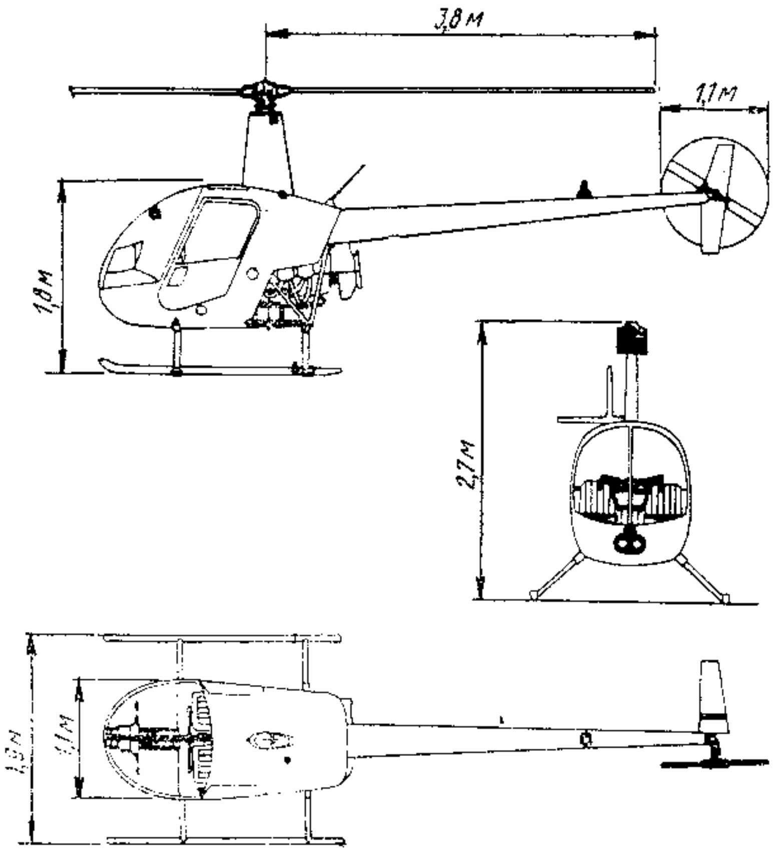


Fig. 4.1. Robinson R-22 Beta

Table 4.2

Parameters of Helicopter Under Design

Sr. No.	Parameters and characteristics of a helicopter and its assemblies	Unit	R-22	Parameters values
1	Normal takeoff mass	kg	621	617.324
2	Airframe mass including:	kg		116.657
	- fuselage	kg		78.813
	- landing gear	kg		12.346
	- tail unit	kg		1.044
	- manual control	kg		24.454
3	Power plant mass including:	kg		185.278
	- engines	kg		71.1
	- service systems	kg		10.466
	- main rotor blades	kg		23.801
	- main rotor hub	kg		20.465
	- anti-torque rotor blades	kg		0.859
	- anti-torque rotor hub	kg		0.049
	- gearbox	kg		43.420
	- tail gearbox	kg		9.995
	- transmission shaft	kg		5.123
4	Fuel mass	kg	52	77.390
5	Main rotor diameter	n	7.67	6.745
6	Rotor solidity ratio	–	0.039	0.053
7	Main rotor blades number	pcs.	2	3
8	Main rotor tip speed	m/s	217	190
9	Main rotor blades chord	m		0.186
10	Main rotor blades aspect ratio	–	–	18.175
11	Main rotor angular speed	°/c		3228
12	Main rotor speed per minute	rpm.		538
13	Anti-torque rotor diameter	M		1.147
14	Anti-torque rotor solidity ratio	–	–	0.105
15	Anti-torque blades number	Pcs.	2	2
16	Anti-torque rotor tip speed	m/s		190
17	Anti-torque rotor blades chord	M		0.095
18	Anti-torque rotor blades aspect ratio	–	–	6.058
19	Anti-torque rotor angular speed	°/s		18988
20	Anti-torque rotor speed per minute	rpm.		3165
21	Equipment mass	kg		43
22	Calculated thrust-weight ratio	kW/N		16.774
23	Engine		R1162	R1162
24	Engine takeoff capacity	kW	119	113
25	Specific load	N/m^2	131.9	168.291
26	Full-to-takeoff load ratio	–	–	0.688

4.2. Selecting Parameters for Light Multipurpose Helicopter with Single Rotor

4.2.1. Initial data

1. Crew mass $\boldsymbol{m}_{cr}$ = 80 *kg.*
2. Payload mass $\boldsymbol{m}_{pl}$ = 600 *kg.*
3. Maximal speed $\boldsymbol{V}_{max}$ = 245 *km/h.*
4. Maximal flight range under normal takeoff mass $\boldsymbol{L}_{max}$ = 600 *km.*
5. Hovering ceiling (out of ground effect) $\boldsymbol{H}_{hov}$ = 2200 *m.*
6. Service ceiling $\boldsymbol{H}_{serv}$ = 5000 *m*.
7. Helicopter mass category – light multipurpose helicopter.

4.2.2. Parameter Selection Completing Initial Data

1. Statistical table includes Aeropasial SA-318C, Aeropasial SA-342L, Aluette II, Bell 206A L-1 Long Ranger, Gesel, Hughes 500E, MVV Belkov Bö-105A.
2. Airfoil portion NACA 23012 is taken for main rotor blades.
3. Tip speed is $\omega\mathbf{R}$ = 205 m/s.
4. Takeoff mass is at zero approximation

$$\mathbf{m_o^o} = \sigma \boldsymbol{V}_{max} = \frac{\boldsymbol{C}_{fuel\ O\ V\,max}}{(\boldsymbol{C}_{fuel\,/\sigma})} = \frac{600+80}{0.55-0.2008} = 1947 \ kg.$$

5. **p** = 160…360 with pitch of 50 N/m^2 is taken as a variability range of specific load **p**.

Dependence diagram $\boldsymbol{m}_o(\boldsymbol{p})$ is based on Table 4.3 (Fig. 4.2).

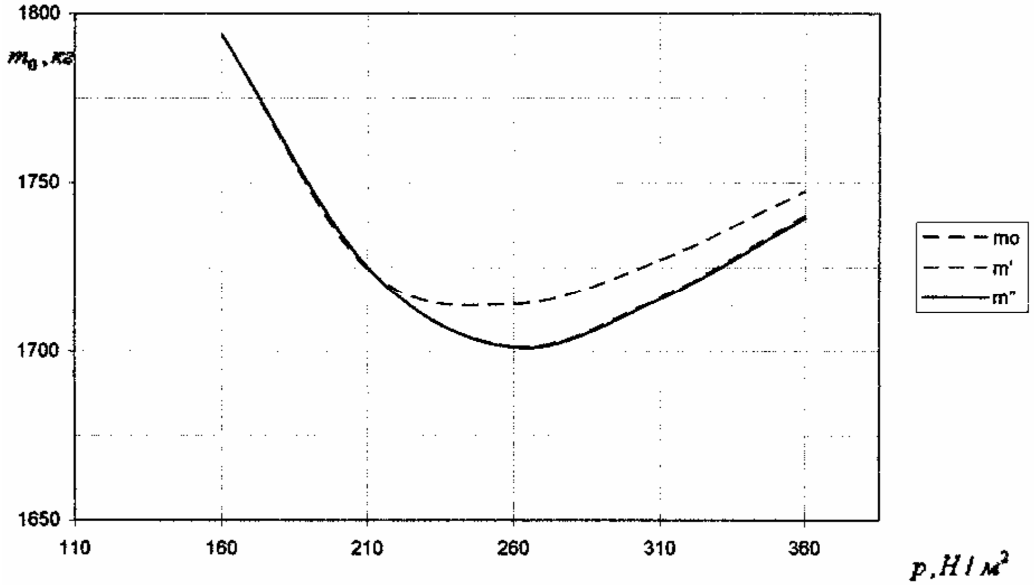


Fig. 4.2. Influence Diagram of Specific Load **p** on $\boldsymbol{m}_0$

Table 4.3

Parameter Calculation for Helicopter under Design

Sr. No.	Parameters, mass ratio, coefficients under $\omega R = 205\ m/s$	Unit	Specific load $\boldsymbol{p}$, N/m^2				
			160	210	260	310	360
			Values of parameters, mass ratio, coefficients				
1	$\sigma_{V_{max}}$	—	0.0349	0.0458	0.0567	0.0677	0.0786
	$\left(\bar{V}^{cr}_{H serv} = 0.2\right).\ \sigma_{H_{serv}}$	—	0.0459	0.0602	0.0746	0.0889	0.1032
	$\sigma^{allow}_{\min}$	—	0.0459	0.0602	0.0746	0.0889	0.1032
	$\sigma_{serv.} = 1.03 \cdot \sigma^{allow}_{min}$	—	0.0473	0.062	0.0768	0.0916	0.1063
2	$R = \sqrt{\dfrac{m_0 \cdot g}{\pi \cdot p}}$	0	6.16	5.38	4.83	4.43	4.11
		I	5.78	5.04	4.53	4.15	3.85
		II	5.76	5.03	4.52	4.14	3.84
3	$Z = 57.1 \cdot \sigma_{serv.}$	0, I, II	2.7	3.54	4.38	5.23	6.07
4	$\bar{m}_{fus}(i)$ $S_{fus} = 30,\ K_{fus} = 1.5$	0	0.1021	0.1021	0.1021	0.1021	0.1021
		I	0.1124	0.1124	0.1124	0.1124	0.1124
		II	0.1129	0.1129	0.1129	0.1129	0.1129
5	$\bar{m}_{LG}(i)$	0, I, II	0.01	0.01	0.01	0.01	0.01
6	$\bar{m}_{HE}$ $\bar{S}_{HE} = 0.0044, K_{HE} = 131$	0, I, II	0.0036	0.0027	0.0022	0.0019	0.0016
7	$\bar{m}_{man.con.}(i)$ $K_{man.con.} = 7\ kg/m$	0	0.02216	0.01934	0.01739	0.01592	0.0148
		I	0.02363	0.02062	0.01853	0.01697	0.0158
		II	0.0237	0.0207	0.0186	0.0170	0.0158
8	$\bar{m}_{pow.\ con.}(i)$ $a_{pow.con.} = 30.8 \cdot R;\ K_{pow.con.} = 13$	0	0.01296	0.01131	0.01017	0.00931	0.00864
		I	0.01216	0.01061	0.00954	0.00873	0.0081
		II	0.01212	0.01058	0.0095	0.0087	0.00808
9	$\bar{m}_{fr} = \sum \bar{m}_i$	0	0.1508	0.14548	0.14185	0.13917	0.13709
		I	0.16176	0.15635	0.15266	0.14994	0.14783
		II	0.16237	0.15695	0.15326	0.15054	0.14842
10	$\tilde{N}_{H_{dyn.0}}$ $S_e = \dfrac{0.0107}{m_0^{0.4354} \cdot g};\ \xi_{Vcr} = 0.865;$ $\bar{N}_{Hdyn.} 0.6525;\ \bar{N}_{nom.} = 0.92;$	0	15.982	18.12	20.128	22.038	23.868
		I	16.063	18.219	20.244	22.17	24.016
		II	16.067	18.224	20.25	22.177	24.024
11	$\bar{N}_{HV\,max} = 0.965;\ I_e = 1.118;$ $\xi_{V\,max}\ 0.875;\ \tilde{N}_{V_{max\,0}}$	0	18.237	18.674	19.11	19.547	19.984
		I	18.752	19.19	19.625	20.062	20.499
		II	18.779	19.216	19.653	20.09	20.526
12	$\tilde{N}_{Hhov._o}$ $\bar{N}_{Hhov.} = 0.8471;\ \eta_0 = 0.7;$ $\xi_{Hhov.} 0.96;\ T = 1$	0	15.81	18.113	20.154	22.007	23.715
		I	15.81	18.113	20.154	22.007	23.715
		II	15.81	18.113	20.154	22.007	23.715

Table 4.3 (Continued)

Sr. No.	Parameters, mass ratio, coefficients under $\omega R = 205\ m/s$	Unit	Specific load p, N/m^2				
			160	210	260	310	360
			Values of parameters, mass ratio, coefficients				
13	$\tilde{N}_{0_{max}}$	0	18.237	18.674	20.154	22.038	23.868
		I	18.752	19.19	20.244	22.17	24.016
		II	18.779	19.216	20.25	22.177	24.024
14	$\bar{m}_f$ $\bar{C}_{eH} = 0.995; \bar{C}_{eV} = 0.987;$ $K_{Ce} = 0.71$	0	0.09573	0.978	0.10475	0.11352	0.12197
		I	0.09943	0.10151	0.10653	0.1156	0.12423
		II	0.09963	0.10171	0.10663	0.11572	0.12435
15	$\bar{m}_{eng.s}$ $K_{fuel.s} = 0.017; \bar{m}_{APU} = 0.005;$ $\gamma_{eng} = 0.25; K_s = 0.02$	0	0.05492	0.05611	0.06015	0.06528	0.07027
		I	0.05634	0.05753	0.06042	0.06567	0.0707
		II	0.05642	0.05761	0.06043	0.06569	0.07073
16	$\bar{m}_{\Sigma bl}(j)$. $K_{bl} = 13$	0	0.04254	0.03868	0.03589	0.03375	0.03203
		I	0.04067	0.03698	0.03432	0.03227	0.03063
		II	0.04058	0.03689	0.03424	0.03219	0.03055
17	$\bar{m}_{hub}(j)$. $K_{hub} = 0.04$	0	0.02945	0.02829	0.02794	0.02835	0.02883
		I	0.0289	0.02776	0.02742	0.02782	0.02829
		II	0.02887	0.02773	0.02739	0.02779	0.02827
18	$\bar{m}_{\Sigma_{t.r.bl}}(j)$ $\sigma_{a-t.r} = 1.7\sigma_{ser}; R_{a-t.r} = 1.16R;$	0	0.00111	0.00101	0.00093	0.00088	0.00083
		I	0.00106	0.00096	0.00089	0.00084	0.0008
		II	0.00106	0.00096	0.00089	0.00084	0.0008
19	$\bar{m}_{a-t.r.hub}(j)$	0	0.00081	0.00078	0.00075	0.00074	0.00072
		I	0.0008	0.00076	0.00074	0.00072	0.00071
		II	0.0008	0.00076	0.00074	0.00072	0.00071
20	$\bar{m}_{prop} = \sum \bar{m}_j$	0	0.07391	0.06875	0.06552	0.06371	0.06242
		I	0.07143	0.06647	0.06337	0.06165	0.06042
		II	0.0713	0.06635	0.06326	0.06155	0.06032
21	$\bar{m}_{m.gb}(ij)$ $K_{m.gb} = 0.0721; \xi = 0.9$	0	0.05596	0.05115	0.04992	0.04998	0.05018
		I	0.05577	0.05095	0.04883	0.04895	0.04915
		II	0.05576	0.05095	0.04878	0.04889	0.0491
22	$\bar{m}_{int.gb}(ij)$	0	0	0	0	0	0
		I	0	0	0	0	0
		II	0	0	0	0	0
23	$\bar{m}_{t.r.g}(ij)$ $K_{t.r.g.} = 0.127; \xi = 0.9$	0	0.00392	0.00359	0.0035	0.0035	0.00352
		I	0.00391	0.00357	0.00342	0.00343	0.00345
		II	0.00391	0.00357	0.00342	0.00343	0.00344
24	$\bar{m}_{tr.sh}(ij)$ $K_{tr.sh} = 0.103; \xi = 0.9$	0	0.00874	0.00775	0.00732	0.00712	0.00697
		I	0.00871	0.00772	0.00719	0.007	0.00685
		I	0.00871	0.00772	0.00718	0.00699	0.00684
25	$\bar{m}_{tr} = \sum \bar{m}_{ij}$	0	0.06862	0.06248	0.06074	0.0606	0.06066
		I	0.06839	0.06225	0.05945	0.05937	0.05944
		II	0.06838	0.06224	0.05938	0.05931	0.05938
26	m_{eq} $K_{eq} = 1,6; K_{wiring} = 17;$ $K_{el.eq} = 14$	0	317.348	301.897	291.152	283.125	276.835
		I	293.22	278.727	268.649	261.12	255.22
		II	292.013	277.569	267.525	260.021	254.141
27	m_o	0	1793.72	1724.5	1712.81	1726.91	1747.37
		I	1793.47	1724.7	1701.37	1718.13	1740.38
		II	1793.7	1724.94	1701	1717.87	1740.22

Error $\xi = \left| \frac{\boldsymbol{m}_0' - \boldsymbol{m}_0''}{\boldsymbol{m}_0'} \right| \cdot 100 = \left| \frac{1701.37 - 1701}{1701.37} \right| \cdot 100 \approx 0$.

4.2.3. Conclusions

Takeoff mass of the helicopter under design $m_0 = 1701$ *kg* with specific load **p** = 260 N/m^2. Our helicopter is close to SW-4 (Fig. 4.3).

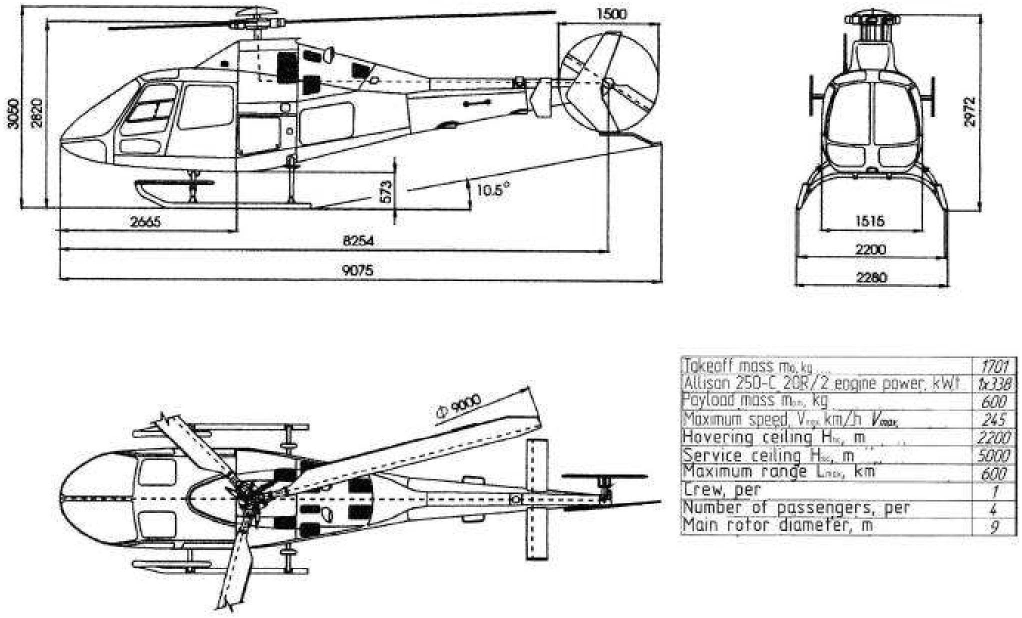


Fig. 4.3. General View of SW-4

The characteristics of the helicopter under design and prototype-helicopter SW-4 are presented in Table 4.4 for comparison.

Table 4.4

Parameters of a Helicopter under Design

Parameters	Helicopters	
	SW-4	Helicopter under design
Takeoff mass m_0, *kg*	1700	1701
Payload mass m_{pl} *kg*	600	600
Number of seats **(**n_{cr}+n_{pas}**)**	1+4	1+4
Main rotor radius *R*, *m*	4.5	4.52
Main rotor tip sped ωR, *m/s*	205	205
Specific load *p*, *H/м²*	262.3	260
Maximum speed V_{max}, *km/h*	245	245
Maximum range L_{max}, *km*	600	600
Hovering ceiling H_{hov}, *m*	2200	2200
Service ceiling H_{ser}, *m*	5200	5000

4.3. Selection of Light Multipurpose Aircraft Parameters with Single Rotor Diagram

4.3.1. Initial Data

1. Crew mass m_{cr} = 90 *kg.*
2. Payload mass m_{pl} = 900 *kg* (8 pass.).
3. Maximal speed V_{max} = 210 *km/h.*
4. Maximal range of flight with normal takeoff mass L_{max} = 365 *km.*
5. Hovering ceiling H_{hov} = 1000 *m.*
6. Dynamic ceiling H_{dyn} = 4000 *m*.

4.3.2. Parameter Selection Completing Initial Data

1. Statistical table includes Augusta A129, Bell 222A, MVV/Kawasaki VK-117, Sikorsky S-55, Westland Whirlwind sr.3.HAR.Mk.1,.
2. Airfoil portion NACA 23012 is taken for rotor blades.
3. Tip speed ωR is taken equal to 190.7 *m/s*.
4. Takeoff mass m_o^o = 3410 *kg* at zero approximation.
5. Variability range of the specific load **p** is taken as:

$$\mathbf{p_1} = 154\ N/m^2;\ \mathbf{p_2} = 204\ N/m^2;\ \mathbf{p_3} = 254\ N/m^2;$$

$$\mathbf{p_4} = 304\ N/m^2;\ \mathbf{p_5} = 354\ N/m^2.$$

Calculations are summarized in Table 4.5. Dependence diagram $m_o(p)$ is based on Table 4.5 (Fig.4.4).

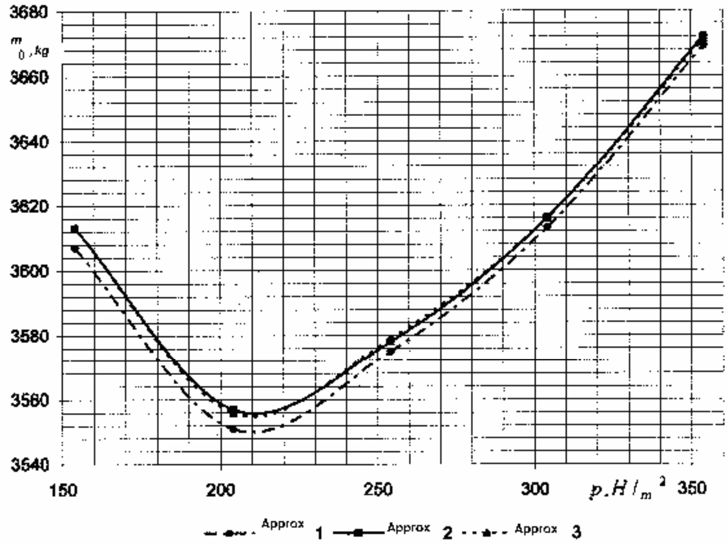


Fig. 4.4. Influence Diagram of Specific Load p on m_o

Table 4.5

Parameter Calculation for Helicopter under Design

Sr. No.	Parameters, mass ratio, coefficients for $\omega R = 190.7\ m/s$	Dimension	Specific load p, N/m^2				
			154	204	254	304	354
			Values of parameters, mass ratio, coefficients				
1	$\sigma_{V_{max}}$	—	0.0371	0.0491	0.0611	0.0731	0.0851
	σ_{min}^{allow}	—	0.0482	0.0638	0.0794	0.095	0.1105
	$\bar{V}_{Hdyn}^{cr} = 0.2$, $\sigma_{H_{dyn}}$	—	0.0482	0.0638	0.0794	0.095	0.1105
	$\sigma_{ser.} = 1.03 \cdot \sigma_{\min}^{allow}$	—	0.0496	0.0657	0.0817	0.0978	0.1139
2	$R = \sqrt{\frac{m_0 \cdot g}{\pi \cdot p}}$	0	8.304	7.218	6.47	5.914	5.481
		I	8.474	7.365	6.602	6.035	5.594
		II	8.482	7. 372	6.608	6.041	5.598
3	$Z = 57.1 \cdot \sigma_{ser}$	0, I, II	3	4	5	6	7
4	$\bar{m}_{fus.}(i)$	0	0.1324	0.1324	0.1324	0.1324	0.1324
		I	0.1285	0.1285	0.1285	0.1285	0.1285
	$S_{fus} = 30$, $K_{fus} = 1.5$	II	0.1283	0.1283	0.1283	0.1283	0.1283
5	$\bar{m}_{LG}(i)$	0, I, II	0,02	0.02	0.02	0.02	0.02
6	$\bar{m}_{HE}$ $\bar{S}_{HE} = 0.0044, K_{HE} = 131$	0, I, II	0.0244	0.0184	0.0148	0.0124	0.0106
7	$\bar{m}_{man.con}(i)$	0	0.056	0.0487	0.0436	0.0399	0.037
		I	0.0549	0.0477	0.0428	0.0391	0.0362
	$K_{man.con} = 7\ kg/m$	II	0.0549	0.0477	0.0428	0.0391	0.0362
8	$\bar{m}_{pow.con}(i)$	0	0.0179	0.0155	0.0138	0.0126	0.0116
	$a_{pow.con} = 30.8 \cdot R; K_{pow.con} = 13.2$	I	0.0183	0.0158	0.0141	0.0129	0.0119
		II	0.0183	0.0158	0.0141	0.0129	0.0119
9	$\bar{m}_{fr} = \sum \bar{m}_i$	0	0.2507	0.235	0.2247	0.2173	0.2117
		I	0.246	0.2304	0.2201	0.2128	0.2072
		II	0.2458	0.2302	0.2199	0.2126	0.207
10	$\tilde{N}_{H_{dyn\,0}}$	0	13.865	15.798	17.612	19.336	20.986
	$S_e = \frac{0.0107}{m_0^{0.4354} \cdot g}; \xi_{Vcr} = 0.865;$	I	13.839	15.767	17.575	19.293	20.939
	$\bar{N}_{Hdyn} 0.6525; \bar{N}_{ceil} = 0.92$	II	13.838	15.766	17.574	19.292	20.937
11	$\tilde{N}_{takeoff\ \ cont\,0}$	0	15.831	17.839	19.723	21.515	23.232
		I	15.802	17.803	19.681	21.466	23.177
		II	15.801	17.801	19.679	21.464	23.175
12	$\tilde{N}_{V_{\max}\,0}$	0	14.811	15. 318	15.825	16.333	16.84
	$\bar{N}_{HV\,max} = 0.965; l_e = 1.118;$	I	14.684	15.191	15.698	16.206	16.713
	$\xi_{V\,max}\, 0.875$	II	14.678	15.186	15.693	16.201	16.708
13	$\tilde{N}_{Hhov_o}$	0	15.954	18.357	20.479	22.401	24.171
	$\bar{N}_{Hhov} = 0.8471; \eta_0 = 0.7;$	I	15.954	18.357	20.479	22.401	24.171
	$\xi_{Hhov}\, 0.96; T = 1$	II	15.954	18.357	20.479	22.401	24.171

Table 4.5 (Continued)

Sr. No.	Parameters, mass ratio, coefficients for $\omega R = 190.7\ m/s$	Dimension	Specific load p, N/m^2				
			154	204	254	304	354
			Values of parameters, mass ratio, coefficients				
14	$\tilde{N}_{0_{max}}$	0	15.954	18.357	20.479	22.401	24.171
		I	15.954	18.357	20.479	22.401	24.171
		II	15.954	18.357	20.479	22.401	24.171
15	$\bar{m}_{fuel}$ $\bar{C}_{eH} = 0.995; \bar{C}_{ev} = 0.987; K_{Ce} = 0.71$	0	0.0577	0.0655	0.0723	0.0783	0.0839
		I	0.0575	0.0652	0.072	0.078	0.0836
		II	0.0575	0.0652	0.072	0.078	0.0835
16	$\bar{m}_{eng.s}$ $K_{fuel.s} = 0.017; \bar{m}_{APU} = 0.005;$ $\gamma_{eng} = 0.25; K_s = 0.02$	0	0.0956	0.1087	0.1203	0.1307	0.1404
		I	0.0956	0.1087	0.1202	0.1307	0.1404
		II	0.0955	0.1087	0.1202	0.1307	0.1404
17	$\bar{m}_{\Sigma_{bl}}(j)$. $K_{bl} = 13$	0	0.0581	0.0524	0.0484	0.0454	0.043
		I	0.059	0.0532	0.0491	0.046	0.0436
		II	0.059	0.0532	0.0491	0.0461	0.0436
18	$\bar{m}_{hub}(j)$. $K_{hub} = 0.04$	0	0.0381	0.0362	0.0366	0.0372	0.038
		I	0.0384	0.0365	0.0369	0.0375	0.0382
		II	0.0384	0.0365	0.0369	0.0375	0.0382
19	$\bar{m}_{\Sigma_{a-t.r.bl}}(j)$ $\sigma_{a-t.r} = 1.7\sigma_{ser}; R_{a-t.r} = 1.16R;$	0	0.0026	0.0022	0.0024	0.0021	0.0019
		I	0.0027	0.0022	0.0024	0.0021	0.0019
		II	0.0027	0.0022	0.0024	0.0021	0.0019
20	$\bar{m}_{a-t.r.hub}(j)$	0	0.00202	0.00165	0.00212	0.00179	0.00168
		I	0.00203	0.00166	0.00213	0.0018	0.00169
		II	0.00203	0.00166	0.00213	0.0018	0.00169
21	$\bar{m}_{m.gb}(ij)$ $K_{m.gb} = 0.0721; \xi = 0.9$	0	0.0612	0.0612	0.0612	0.0612	0.0612
		I	0.0617	0.0617	0.0617	0.0617	0.0617
		II	0.0617	0.0617	0.0617	0.0617	0.0617
22	$\bar{m}_{int.gb}(ij)$	0	0.003	0.0033	0.0036	0.0039	0.0042
		I	0.003	0.0033	0.0036	0.0039	0.0042
		II	0.003	0.0033	0.0036	0.0039	0.0042
23	$\bar{m}_{t.gb}(ij)$ $K_{t.gb} = 0.127; \xi = 0.9$	0	0.0048	0.0048	0.0048	0.0048	0.0048
		I	0.00484	0.00484	0.00484	0.00484	0.00484
		II	0.00484	0.00484	0.00484	0.00484	0.00484
24	$\bar{m}_{tr.sh}(ij)$ $K_{tr.sh} = 0.103; \xi = 0.9$	0	0.00338	0.00324	0.00313	0.00304	0.00297
		I	0.0034	0.00326	0.00315	0.00306	0.00299
		I	0.0034	0.00326	0.00315	0.00306	0.00299
25	$\bar{m}_{tr} = \sum \bar{m}_{ij}$	0	0.07232	0.07253	0.07273	0.07291	0.07308
		I	0.07286	0.07306	0.07325	0.07343	0.07361
		II	0.07288	0.07308	0.07327	0.07346	0.07361
26	m_{eq} $K_{eq} = 1.6; K_{wiring} = 17;$ $K_{el.eq} = 14$, kg	0	535.05	522.17	513.31	506.73	501.6
		I	549.31	536.17	527.12	520.41	515.17
		II	549.91	536.76	527.71	520.99	515.75
27	m_o, kg	0	3607	3551	3575	3613	3669
		I	3613	3557	3578	3616	3672
		II	3613	3556	3579	3616	3671

Choose the engine by the starting power:

$$N_{0\,max\,eng.} = \frac{\tilde{N}_{0\,max} \cdot m_{0\,min} \cdot g}{n_{eng}} = \frac{0.018357 \cdot 3556 \cdot 9.8}{2} = 305\,kWt,$$

where n_{eng} is a number of engines (n_{eng} = 2).

Parameters of the helicopter under design are presented in Table 4.6.

Table 4.6

Parameters of Helicopter Under Design

Sr. No.	Parameters and characteristics of helicopter and its assemblies	Unit	MI-2	Helicopter under design
1	Normal takeoff mass	kg	3550	3550
2	Airframe mass: - fuselage, - landing gear, - horizontal tail unit, - manual control, - powered control	kg kg kg kg kg kg		807 445 71 65 170 56
3	Power plant mass: - engines, - service systems, - main rotor blades, - main rotor hub, - anti-torque rotor blades, - anti-torque rotor hub, - gearbox, - intermediate gearbox, - tail gearbox, - transmission shaft	kg kg kg kg kg kg kg kg kg kg kg	 135(1eng.without elect.) 171.3 130 22.15 300 17	1183 140 386 167 130 7 17 284 12 17 23
4	Fuel mass	kg	465-1205	465
5	Main rotor diameter	m	14.5	14.5
6	Rotor solidity ratio	—	0.0433	0.0527
7	Number of main rotor blades	pcs.	3	3
8	Main rotor tip speed	m/s	247	200
9	Main rotor angular speed	radps		27.58
10	Anti-torque rotor diameter	m	2.7	2.7
11	Number of anti-torque blades	pcs.	2	2
12	Equipment mass	kg	25	487
13	Engine	turb. eng.	2 x Turb.eng.-350	2
14	Engine takeoff power	kW	2 x 350 hp	2 x 294.4
15	Specific load	N/m^2	20.3 kgc/m^2	211
16	Full-to-takeoff load ratio	—	0.38	0.318

4.3.3. Conclusions

Takeoff mass of the helicopter under design is $m_0 = 3550$ *kg* with the specific load $p = 211$ N/m^2. We choose the engine 250-MTU-020V which produces the necessary power N_{0max} = 309 *kW.*

Our helicopter is close to MI-2 (Fig. 4.5) in its tactical-technical requirements, performance, mass characteristics, and dimensions.

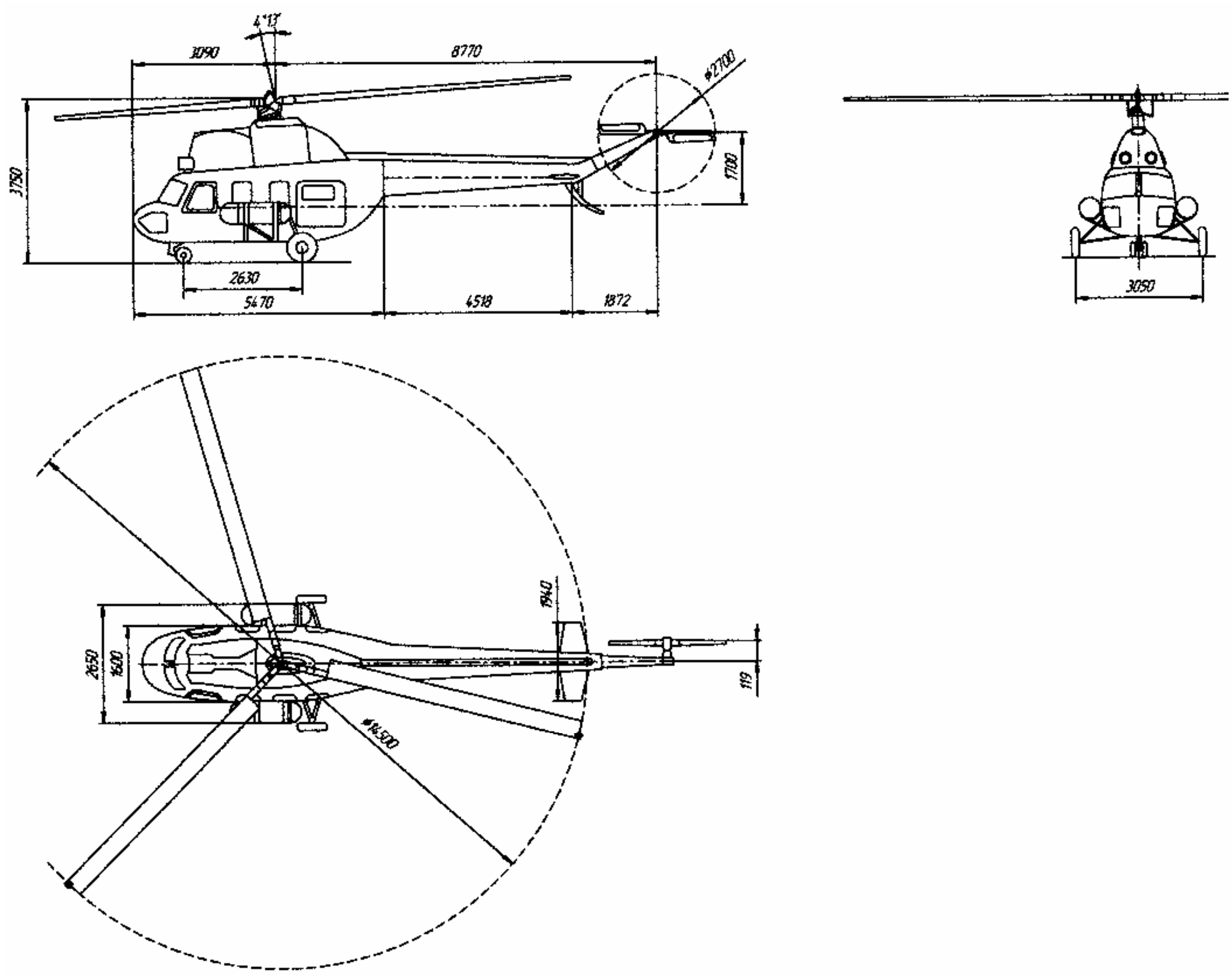


Fig.4.5. General View of MI-2

4.4. Selecting Parameters for Medium Single-Rotor Helicopter

4.4.1. Initial data

1. Crew mass m_{cr} = 270 kg;
2. Payload mass m_{pl} = 4000 kg;
3. Maximal speed V_{max} = 250 km/h;
4. Maximal range of flight with normal takeoff mass L_{max} = 365 km;
5. Hovering ceiling H_{hov} = 700 m;
6. Dynamic ceiling H_{dyn} = 4500 m.

4.4.2. Parameters Selection Completing Initial Data

1. Takeoff mass is at zero approximation

$$m_0^0 = \frac{m_{pl} + m_{cr}}{\overline{m}_{full\ l} - \overline{m}_{fuel}}; \quad \overline{m}_{full\ l} = 1 - \overline{m}_{emp} = 1 - 0.6158 = 0.3842;$$

$$\overline{m}_{fuel} = \overline{q}_{fuel} \cdot L + 0.33 \cdot \overline{Q}_V = 0.22 \cdot 10^{-3} \cdot 365 + 0.057 \cdot 0.33 = 0.0991,$$

where $\overline{q}_{fuel} = 0.22 \cdot 10^{-3} ... 0.24 \cdot 10^{-3}$, $\overline{q}_{fuel} = 0.22 \cdot 10^{-3}$;

$$\overline{Q}_V = 0.057...0.059, \overline{Q}_V = 0.057;$$

$$m_0^0 = \frac{4000 + 270}{0.3842 - 0.0991} = 14977\ kg.$$

Full-to-takeoff load ratio $\overline{m}_{full\ l}$ = 0.384 is taken based on a prototype.

2. Airfoil portion NACA-230M with C_y = 0.67 M_{torq} = 0.72 and with $C_{y\ high}$ = 0.6 M_{torq} = 0.64 is taken for rotor blades.

3. Tip speed of ωR = 214 m/s is taken based on initial data by V_{max}.

Calculations are summarized in Table 4.7. Dependence diagram $m_o(p)$ is based on Table 4.7.

Select engine

$$N_{0\,max\,eng} = \frac{\tilde{N}_{0\,max} \cdot m_{0\,min} \cdot g}{n_{eng}} = \frac{0.02116 \cdot 11511.6 \cdot 9.8}{2} = 1194\ kW.$$

4.4.3. Conclusions

Takeoff mass of the helicopter under design is taken for the specific load of p = 290 N/m^2. We choose the engine TV-117Ag which produces the necessary power of 1100 kW.

Our helicopter is close to MI-8 (refer to Appendices) in its tactical-technical requirements, performance, mass characteristics, and dimensions.

Table 4.7

Selection of Parameters of Medium Helicopter

Sr. No.	Formulas for calculation of parameters, mass ratio and coefficients under ωR =214m/s	Dimension	Number of approximations	Specific load p N/m 2				
				200	250	300	350	400
				Values of parameters, mass ratio, coefficients				
1	$\sigma_{V\,max} = \frac{C_{T0\,V\,max}}{(C_T/\sigma)^{доп}_{V\,max}}$; $C_{T0\,V\,max} = \frac{1,63p}{(\omega R)^2}$; $(C_T/\sigma)^{allow}_{V\,max} = 0,297 - 0,36\bar{V}_{max} = 0,1802$; $\bar{V}_{max} = \frac{V_{max}}{3,6\omega R} = 0,32451$	—	$\sigma_{V\,max}$	0,0395	0,0494	0,0593	0,0691	0,0790
	$\sigma_{H\,dyn} = \frac{C_{T\,Hdyn}}{(C_T/\sigma)_H}$; $C_{T\,H_{dyn}} = \frac{1,63p}{(\omega R)^2 \Delta_{Hдин}}$; $(C_T/\sigma)^{allow}_{Hdyn} = 0,297 - 0,36\bar{V}^{ec}_{Hdyn} = 0,216$; $\bar{V}^{ec\,1)}_{Hdyn} = 0,2...0,25$; $\bar{V}^{ec}_{Hdyn} = 0,225$; $\Delta_{Hdyn} = 0,634316$	—	$\sigma_{H_{dyn}}$	0,0520	0,0649	0,0779	0,0909	0,1039
	$\sigma^{allow}_{min} = max\{\sigma_{V\,max}, \sigma_{H_{dyn}}\}$	—	σ^{allow}_{min}	0,0520	0,0649	0,0779	0,0909	0,1039
	$\sigma^{2)}_{ser} = 1,03\sigma^{allow}_{min}$	—	σ_{ser}	0,0536	0,0668	0,0802	0,0936	0,1070
2	$R = \sqrt{\frac{m_0 g}{\pi p}}$; $m^0_0 = 14977$ kg;	M	$R^{0\,5)}$	15,29	13,68	12,49	11,56	10,81
	$m^I_0 = 11503,5$ kg;	M	$R^{I\,5)}$	13,40	11,99	10,94	10,13	9,48
	$m^{II}_0 = 11510,4$ kg	M	$R^{II\,5)}$	13,41	11,99	10,95	10,13	9,48

Table 4.7 (Continued)

Sr. No.	Formulas for calculation of parameters, mass ratio and coefficients under R=214 m/s	Dimension	Number of approxi-mations	Specific load p N/m²				
				200	250	300	350	400
				Values of parameters, mass ratio, coefficients				
3	$z = \pi \cdot \lambda_{aver}^{3)} \cdot \sigma = \pi \cdot \frac{R}{b_{0,7}} \cdot \sigma \cong \pi \cdot \frac{R}{0,055R} \cdot \sigma$; $z \cong 57,1 \cdot \sigma$; $\lambda_{aver} = 18,182$		z^0, z^I, z^{II}	3,06	3,81	4,58	5,34	6,11
4	$\overline{m}_{fus} = K \cdot S^{0,88} / m_0^{0,75}; K = 2,3\,kg^{0,75} / m^{1,76}$; $S_{fus} \approx 105 m^2$	—	$\overline{m}^{0}_{fus}$	0,102	0,102	0,102	0,102	0,102
		—	$\overline{m}^{I}_{fus}$	0,12438	0,12438	0,12438	0,12438	0,12438
		—	$\overline{m}^{II}_{fus}$	0,12432	0,12432	0,12432	0,12432	0,12432
5	$\overline{m}_{lg} = K_{lg}$	—	$\overline{m}^{0}_{lg}$, $\overline{m}^{I}_{lg}$, $\overline{m}^{II}_{lg}$	0,02	0,02	0,02	0,02	0,02
6	$\overline{m}_{he} = K_{he} \cdot \frac{\overline{S}_{he}}{p}; K_{he} = 131,4\,kg/\,m \cdot s^2$; $\overline{S}_{he} = 0,0056$	—	$\overline{m}^{0}_{he}$, $\overline{m}^{I}_{he}$, $\overline{m}^{II}_{he}$	0,00368	0,00294	0,00245	0,00210	0,00184
7	$\overline{m}_{man.\,con} = K_{man.\,con} \cdot \frac{R}{m_0}; K^{1)}_{man.\,con} = 18...25\,kg/m;$ $K_{man.\,con} = 19\,kg/m;$	—	$\overline{m}^{0}_{man.\,con}$	0,0194	0,01735	0,01584	0,01467	0,01371
		—	$\overline{m}^{I}_{man.\,con}$	0,02213	0,01980	0,01807	0,01673	0,01566
		—	$\overline{m}^{II}_{man.\,con}$	0,02214	0,01979	0,01807	0,01672	0,01565

Table 4.7 (Continued)

Sr. No.	Formulas for calculation of parameters, mass ratio and coefficients under R=214 m/s	Dimension	Number of approxi-mations	Specific load p N/m²				
				200	250	300	350	400
				Values of parameters, mass ratio, coefficients				
8	$\overline{m}_{pow.\,con}= a_{pow.\,con}\cdot K_{pow.\,con}\cdot\sigma^2/zp;$ $a_{pow.\,con}=30{,}8R\ m^2/s^2;\ K_{pow.\,con}=13{,}2\ kg/m^3$	—	$\overline{m}^{0}_{pow.\,con}$	0,02918	0,02606	0,02377	0,02203	0,02059
		—	$\overline{m}^{I}_{pow.\,con}$	0,02557	0,02284	0,02082	0,01931	0,01806
		—	$\overline{m}^{II}_{pow.\,con}$	0,02559	0,02284	0,02084	0,01931	0,01806
9	$\overline{m}_{fr}(p)=\overline{m}_{fus}(p)+\overline{m}_{lg}(p)+\overline{m}_{he}(p)+$ $+\overline{m}_{man.\,con}(p)+\overline{m}_{pow.\,con}(p)$	—	$\overline{m}^{0}_{fr}$	0,1743	0,1684	0,1641	0,1608	0,1581
		—	$\overline{m}^{I}_{fr}$	0,1958	0,1900	0,1857	0,1825	0,1799
		—	$\overline{m}^{II}_{fr}$	0,1957	0,1899	0,1857	0,1825	0,1799
10	$\tilde{N}_{Hdyn0}=\frac{1}{\overline{N}_{nom}\cdot\overline{N}_{Hdyn}\cdot\overline{N}_{Vcr}\cdot\xi_{cr}}x$ $x\left\{16{,}4\cdot10^{-3}\cdot\omega R\left[1+7{,}08\cdot10^{-8}\left(V^{cr}_{dyn}\right)^3\right]+\right.$ $+1{,}82\cdot\frac{p}{V^{cr}_{dyn}\cdot\Delta_{dyn}}+13{,}2\cdot10^{-3}\cdot\overline{S}_{op}\cdot$ $\left.\cdot\left(V^{cr}_{dyn}\right)^3\cdot\Delta_{dyn}\right\};$ $\overline{S}_{op}=\frac{\chi\left(\sum C_xS\right)}{m_0g}=\frac{0{,}018m_0^{0{,}5646}}{m_0g}=\frac{0{,}018}{m_0^{0{,}4354}g}$	W / N	$\tilde{N}^{0}_{Hdyn0}$	16,63	18,46	20,19	21,84	23,43
			$\tilde{N}^{I}_{Hdyn0}$	16,79	18,64	20,39	22,07	23,69
			$\tilde{N}^{II}_{Hdyn0}$	16,79	18,64	20,39	22,07	23,69
		—	$\overline{N}^{0}_{Vcr}$	1,0134	0,0150	1,0164	1,0177	1,0189
№		—	$\overline{N}^{I}_{Vcr}$	1,0130	1,0145	1,0159	1,0172	1,0184
		—	$\overline{N}^{II}_{Vcr}$	1,0130	1,0145	1,0159	1,0172	1,0184
		km / h	$V^{cr\,0}_{dyn}$	156,05	165	172,69	179,48	185,57

Table 4.7 (Continued)

Sr. No.	Formulas for calculation of parameters, mass ratio and coefficients under R=214 m/s	Dimension	Number of approxi-mations	Specific load p N/m² 200	250	300	350	400
				Values of parameters, mass ratio, coefficients				
10	$\chi = 1; \Sigma C_x S = 0{,}018 m_0^{0{,}5646} m^2;$ $\overline{S}^{0}_{op} = 0{,}279 \cdot 10^{-4} m^2/N;$ $\overline{S}^{I}_{op} = 0{,}313 \cdot 10^{-4} m^2/N;$ $\overline{S}^{II}_{op} = 0{,}313 \cdot 10^{-4} m^2/N$ $\overline{N}_{nom} = 0{,}9; \xi_{Vcr} = 0{,}865; \Delta_{dyn} = 0{,}634316;$ $\overline{N}_{Hdyn} = 1 - 0{,}0695 H_{dyn} = 0{,}68725;$ $\overline{N}_{Vcr} = 1 + 5{,}5 \cdot 10^{-7} \left(V^{cr}_{dyn}\right)^2;$ $V^{cr}_{dyn} = 164 \sqrt[4]{\frac{1{,}09 \cdot p}{\left(\omega R + 11{,}6 \cdot 10^6 \overline{S}_{op} \Delta_{dyn}\right)\Delta_{dyn}}}$		$V^{cr\,I}_{dyn}$	153,80	162,63	170,21	176,90	182,90
			$V^{cr\,II}_{dyn}$	153,80	162,63	170,21	176,90	182,90
11	$\tilde{N}_{takeoff\,cont.0} = \frac{\tilde{N}_{takeoff\,cont}}{0{,}865 \cdot \overline{N}_{Vcr}} \cdot \frac{n_{eng}}{n_{eng}-1} = \frac{1{,}156}{\cdot \overline{N}_{Vcr}} \cdot \frac{n_{eng}}{n_{eng}-1} \times$ $\times \left\{ 16{,}4 \cdot 10^{-3} \cdot \omega R \left[1 + 7{,}08 \cdot 10^{-8} \left(V^{cr}_{dyn}\right)^3 \right] + \right.$ $\left. + 1{,}82 \cdot \frac{p}{V_0^{cr}} + 13{,}2 \cdot 10^{-3} \cdot \overline{S} \cdot \left(V_0^{cr}\right)^3 \right\};$ $\overline{S}^{0}_{op} = 0{,}279 \cdot 10^{-4} m^2/N;$ $\overline{S}^{I}_{op} = 0{,}313 \cdot 10^{-4} m^2/N;$ $\overline{S}^{II}_{op} = 0{,}313 \cdot 10^{-4} m^2/N$		$\tilde{N}^{0}_{takeoff\,cont.0}$	17,58	19,29	20,92	22,48	23,99
			$\tilde{N}^{I}_{takeoff\,cont.0}$	17,75	19,50	21,16	22,75	24,28
			$\tilde{N}^{II}_{takeoff\,cont.0}$	17,75	19,50	21,16	22,75	24,28
		—	$\overline{N}^{0}_{Vcr}$	1,0094	1,0105	1,0115	1,0125	1,0133
		—	$\overline{N}^{I}_{Vcr}$	1,0091	1,0102	1,0111	1,0120	1,0129
		—	$\overline{N}^{II}_{Vcr}$	1,0091	1,0102	1,0111	1,0120	1,0129
			$V_0^{cr\,0}$	130,86	138,38	144,83	150,52	155,63
			$V_0^{cr\,I}$	128,57	135,95	142,29	147,88	152,90

Table 4.7 (Continued)

Sr. No.	Formulas for calculation of parameters, mass ratio and coefficients under R=214 m/s	Dimension	Number of approxi-mations	Specific load p N/m² 200	250	300	350	400
				Values of parameters, mass ratio, coefficients				
11	$V_{0\,cr} = 164\sqrt[4]{\dfrac{1,09 \cdot p}{\omega R + 11,6 \cdot 10^{6}\bar{S}_{op}}}$		V_{0cr}^{II}	128,57	135,95	142,29	147,88	152,90
12	$\tilde{N}_{V\,max\,0} = \dfrac{1}{\bar{N}_{HV\,max} \cdot \bar{N}_{V\,max} \cdot \xi_{V\,max}} \cdot \left\{16,4 \cdot 10^{-3} \cdot \omega R \times \left[1 + 7,08 \cdot 10^{-8} V_{max}^{3}\right] + 1,67\dfrac{p\,l_{op}}{V_{max}} + 13,2 \cdot 10^{-3} \cdot \bar{S}_{op} \times V_{max}^{3}\right\}$ $l_{op} = 1,02 + 0,0004\,V_{max} = 1,12;$ $\bar{N}_{HV\,max} = 1 - 0,0695\,H_{V\,max} = 0,96525;$ $V_{max}, km/h; \quad H_{V\,max} = 0,5 km;$ $\bar{N}_{V\,max} = 1 + 5,5 \cdot 10^{-7} \cdot V_{max}^{2} = 1,0344; \quad \xi = 0,875$		$\tilde{N}_{V\,max\,0}^{0}$	16,76	17,19	17,62	18,02	18,47
			$\tilde{N}_{V\,max\,0}^{I}$	17,59	17,99	18,42	18,85	19,28
			$\tilde{N}_{V\,max\,0}^{II}$	17,59	17,99	18,42	18,85	19,28
13	$\tilde{N}_{Hhov\,0} = \dfrac{\tilde{N}_{Hhov}}{\bar{N}_{Hhov} \cdot \xi} = \dfrac{0,6385 \cdot \bar{T}^{3/2} \cdot \sqrt{p}}{\bar{N}_{Hhov} \cdot \xi \cdot \eta_0 \sqrt{\Delta_{Hhov}}}; \quad \xi = 0,9;$ $\eta_0 = 0,7; \quad \bar{T} \cong 1,04; \quad \Delta_{Hhov} = 0,934514;$ $\bar{N}_{Hhov} = 1 - 0,0695\,H_{hov} = 0,95135$		$\tilde{N}_{Hhov}^{0}$	16,53	18,48	20,24	21,87	23,38
			$\tilde{N}_{Hhov}^{I}$	16,53	18,48	20,24	21,87	23,38
			$\tilde{N}_{Hhov}^{II}$	16,53	18,48	20,24	21,87	23,38

Table 4.7 (Continued)

Sr. No.	Formulas for calculation of parameters, mass ratio and coefficients under R=214 m/s	Dimension	Number of approxi-mations	Specific load p N/m^2				
				200	250	300	350	400
				Values of parameters, mass ratio, coefficients				
14	$\tilde{N}_{0\,max} = max\{\tilde{N}_{H\,dyn0}, \tilde{N}_{takeoff\,cont.0}, \tilde{N}_{Vmax0}, \tilde{N}_{Hhov\,0}\}$	W/N	$\tilde{N}^{0}_{0\,max}$	17,58	19,29	20,92	22,48	23,99
			$\tilde{N}^{I}_{0\,max}$	17,75	19,50	21,16	22,75	24,28
			$\tilde{N}^{II}_{0\,max}$	17,75	19,50	21,16	22,75	24,28
15	$\overline{m}_{fuel} = 1{,}07\dfrac{C_{e\,cruis} \cdot L}{V_{max}} \cdot \tilde{N}_{0\,max} \cdot g;$ $C_{ecruis} = C_{etakeoff} \cdot \overline{C}_{eH} \cdot \overline{C}_{eV} \cdot \overline{C}_{et} \cdot \overline{C}_{eN};$ $\overline{C}_{eH} = 0{,}995; \quad \overline{C}_{eV} = 1 - 3 \cdot 10^{-7} \cdot V_{cruis} = 0{,}98613;$ $V_{cruis} \cong 0{,}86 \cdot V_{max}; \quad \overline{C}^{4)}_{et} = 1{,}0; \quad \overline{C}_{eN} = 1{,}075;$ $C_{etakeoff} = \dfrac{K_{Ce}}{(\tilde{N}_{0\,max} \cdot m_0 \cdot g)^{0,1}};$ $K^{1)}_{Ce} = 0{,}64 \ldots 0{,}71\, {kg}/{kW^{0,9} \cdot h};$ $K_{Ce} = 0{,}69\, {kg}/{kW^{0,9} \cdot h}$	—	$\overline{m}^{0}_{fuel}$	0,0894	0,0972	0,1045	0,1115	0,1182
			$\overline{m}^{I}_{fuel}$	0,0926	0,1007	0,1084	0,1157	0,1227
			$\overline{m}^{II}_{fuel}$	0,0926	0,1007	0,1084	0,1157	0,1227
		$\dfrac{kg}{kW \cdot h}$	C^{0}_{ecruis}	0,3317	0,3287	0,3260	0,3237	0,3216
			C^{I}_{ecruis}	0,3403	0,3371	0,3344	0,3319	0,3298
			C^{II}_{ecruis}	0,3403	0,3371	0,3344	0,3319	0,3297
		$\dfrac{kg}{kW \cdot h}$	$C^{0}_{etakeoff}$	0,3145	0,3116	0,3091	0,3069	0,3049
			$C^{I}_{etakeoff}$	0,3226	0,3196	0,3170	0,3147	0,3127
			$C^{II}_{etakeoff}$	0,3226	0,3196	0,3170	0,3147	0,3126

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Table 4.7 (Continued)

<table>
<tr><th rowspan="3">Sr. No.</th><th rowspan="3">Formulas for calculation of parameters, mass ratio and coefficients under R=214 m/s</th><th rowspan="3">Dimension</th><th rowspan="3">Number of approxi-mations</th><th colspan="5">Specific load p N/m²</th></tr>
<tr><th>200</th><th>250</th><th>300</th><th>350</th><th>400</th></tr>
<tr><th colspan="5">Values of parameters, mass ratio, coefficients</th></tr>
<tr><td rowspan="3">16</td><td rowspan="3">$\overline{m}_{eng.s} = (\gamma_{eng} + K_s)\cdot \tilde{N}_{0max}\cdot g + K_{fuel\,s}\cdot m_{fuel} + m_{apu}$;
$\gamma_{eng}^{1)} = 0,13...0,18\,kg/kW$; $\gamma_{eng} = 0,16\,kg/kW$;
$K_s^{1)} = 0,04...0,05\,kg/kW$; $K_s = 0,04\,kg/kW$;
$K_{fuel\,s}^{1)} = 0,06...0,07$; $K_{fuel\,s} = 0,06$;
$\overline{m}_{apu}^{1)} = 0,005...0,008$; $\overline{m}_{apu} = 0,006$</td><td rowspan="3">—</td><td>$\overline{m}^{0}_{eng.s}$</td><td>0,0459</td><td>0,0497</td><td>0,0533</td><td>0,0568</td><td>0,0602</td></tr>
<tr><td>$\overline{m}^{I}_{eng.s}$</td><td>0,0464</td><td>0,0503</td><td>0,0540</td><td>0,0576</td><td>0,0610</td></tr>
<tr><td>$\overline{m}^{II}_{eng.s}$</td><td>0,0464</td><td>0,0503</td><td>0,0540</td><td>0,0576</td><td>0,0610</td></tr>
<tr><td rowspan="3">17</td><td rowspan="3">$\overline{m}_{\Sigma bl} = a_{bl}\cdot K_{bl}\frac{\sigma}{\lambda^{0,7}\cdot p}$; $a_{bl} = 23,62\cdot R^{0,7}\, m^{1,7}/s^{2}$;
$K_{bl}^{1)} = 12,6...13,8\,kg/m^{2,7}$;
$K_{bl} \approx 13,8\,kg/m^{2,7}$; $\lambda_{aver} \approx z/\pi\sigma$</td><td rowspan="3">—</td><td>$\overline{m}^{0}_{\Sigma bl}$</td><td>0,0774</td><td>0,0714</td><td>0,0670</td><td>0,0635</td><td>0,0606</td></tr>
<tr><td>$\overline{m}^{I}_{\Sigma bl}$</td><td>0,0706</td><td>0,0651</td><td>0,0611</td><td>0,0579</td><td>0,0553</td></tr>
<tr><td>$\overline{m}^{II}_{\Sigma bl}$</td><td>0,0706</td><td>0,0651</td><td>0,0611</td><td>0,0579</td><td>0,0553</td></tr>
<tr><td rowspan="3">18</td><td rowspan="3">$\overline{m}_{hub} = a_{hub}\, 10^{-5}\cdot K_{hub}\, K_z\cdot z\cdot \overline{m}_{bl}^{1,35}\cdot(\omega R)^{2,7}\cdot p^{0,35}$;
$a_{hub} = \frac{2,34}{R^{0,65}}\,\frac{s^{0,7}}{m}$; $K = 0,0527\,kg/kHz^{1,35}$;
$K_z = 1 + 0,05(z-4)$; $\overline{m}_{bl} = \overline{m}_{\Sigma bl}/z$</td><td rowspan="3">—</td><td>$\overline{m}^{0}_{hub}$</td><td>0,0560</td><td>0,0541</td><td>0,0542</td><td>0,0549</td><td>0,0558</td></tr>
<tr><td>$\overline{m}^{I}_{hub}$</td><td>0,0539</td><td>0,0520</td><td>0,0521</td><td>0,0529</td><td>0,0537</td></tr>
<tr><td>$\overline{m}^{II}_{hub}$</td><td>0,0539</td><td>0,0520</td><td>0,0521</td><td>0,0529</td><td>0,0537</td></tr>
</table>

Table 4.7 (Continued)

Sr. No.	Formulas for calculation of parameters, mass ratio and coefficients under R=214 m/s	Dimension	Number of approxi-mations	Specific load p N/m² 200	250	300	350	400
				Values of parameters, mass ratio, coefficients				
19	$\overline{m}_{\Sigma t.r.bl} = \frac{\sigma_{tr}}{\sigma}\cdot\left(\frac{\lambda}{\lambda_{tr}}\right)^{0,7}\cdot\left(\frac{R_{tr}}{R}\right)^{2,7}\overline{m}_{\Sigma bl};$	—	$\overline{m}^{0}_{\Sigma t.r.bl}$	0,0035	0,0032	0,0030	0,0029	0,0027
	$\sigma_{tr}^{1)} = (1,7...2,3)\sigma;\ \sigma_{tr} \approx 2\sigma;$ $\frac{\lambda}{\lambda_{tr}} = \frac{2z}{z_{tr}};\ z_{tr} = \frac{2}{3}z;$	—	$\overline{m}^{I}_{\Sigma t.r.bl}$	0,0032	0,0030	0,0028	0,0026	0,0025
	$R_{tr}^{1)} = (0,16...0,25)R;\ R_{tr} \cong 0,185R$	—	$\overline{m}^{II}_{\Sigma t.r.bl}$	0,0032	0,0030	0,0028	0,0026	0,0025
20	$\overline{m}_{tr.hub} = \frac{K_{z\,t.r}}{K_z}\cdot\frac{z_{t.r}}{z}\cdot\left(\frac{(\omega R)_{t.r}}{\omega R}\right)^{2,7}\cdot\left(\frac{R}{R_{t.r}}\right)^{0,65} x$	—	$\overline{m}^{0}_{tr.hub}$	0,0030	0,0029	0,0028	0,0027	0,0027
	$x\left(\frac{\overline{m}_{t.r.bl}}{\overline{m}}\right)^{1,35}\cdot\overline{m}_{hub};$	—	$\overline{m}^{I}_{tr.hub}$	0,0029	0,0028	0,0027	0,0026	0,0026
	$(\omega R)_{t.r} \cong \omega R;\ K_{z\,t.r} = 1;\ \overline{m}_{t.r.bl} = \overline{m}_{\Sigma t.r.bl} / z_{t.r}$	—	$\overline{m}^{II}_{tr.hub}$	0,0029	0,0028	0,0027	0,0026	0,0026
21		—	$\overline{m}^{0}_{apu}$	0,1399	0,1316	0,127	0,124	0,1218
	$\overline{m}_{apu} = \overline{m}_{\Sigma bl} + \overline{m}_{hub} + \overline{m}_{\Sigma t.r.bl} + \overline{m}_{tr.hub}$	—	$\overline{m}^{I}_{apu}$	0,1306	0,1229	0,1187	0,116	0,1141
		—	$\overline{m}^{II}_{apu}$	0,1306	0,1229	0,1187	0,116	0,1141

Table 4.7 (Continued)

Sr. No.	Formulas for calculation of parameters, mass ratio and coefficients under R=214 m/s	Dimension	Number of approxi-mations	Specific load p N/m^2				
				200	250	300	350	400
				Values of parameters, mass ratio, coefficients				
22	$\overline{m}_{m.gb} = a_{m.gb}\cdot K_{m.gb}\cdot \xi^{0,8}\left(\frac{\tilde{N}_{0\,max}}{\omega R}\right)^{0,8}\cdot\frac{1}{p^{0,2}};$ $a_{m.gb} = 7,8R^{0,4}\ \ {}^{1,7}\ /s^2;$ $K_{m.gb} = 0,0748\ kg/(N\cdot m)^{0,8};\quad \xi = 0,872$	—	$\overline{m}^{0}_{m.gb}$	0,07306	0,07198	0,07140	0,07110	0,07099
		—	$\overline{m}^{I}_{m.gb}$	0,06983	0,06887	0,06834	0,06809	0,06801
		—	$\overline{m}^{II}_{m.gb}$	0,06986	0,06887	0,06836	0,06809	0,06801
23	$\overline{m}_{int.gb}=a_{int.gb}\cdot K_{int.gb}\cdot(1-\xi)^{0,8}\left(\frac{\tilde{N}_{0\,max}}{\omega_{tr.sh}}\right)^{0,8}\cdot\frac{1}{p^{0,2}};$ $a_{int.gb}= 7,8/R^{0,4}\ m^{0,6}/s^2;$ $K_{int.gb}= 0,137\ kg/(N\cdot m)^{0,8};$ $\xi = 0,872;\ \omega_{tr.sh} \cong 314\ s^{-1}$	—	$\overline{m}^{0}_{int.gb}$	0,00239	0,00258	0,00275	0,00291	0,00307
		—	$\overline{m}^{I}_{int.gb}$	0,00254	0,00274	0,00293	0,00310	0,00327
		—	$\overline{m}^{II}_{int.gb}$	0,00254	0,00274	0,00293	0,00310	0,00327
24	$\overline{m}_{t.gb} = a_{t.gb}\cdot K_{t.gb}\cdot(1-\xi)^{0,8}\left(\frac{\tilde{N}_{0\,max}\cdot R_{t.r}}{\omega R}\right)^{0,8}\cdot\frac{1}{p^{0,2}};$ $a_{t.gb}= 7,8/R^{0,4}\ m^{0,6}/s^2;$ $K_{t.gb} = 0,105\ kg/(N\cdot m)^{0,8};$ $\xi = 0,872$	—	$\overline{m}^{0}_{t.gb}$	0,00573	0,00564	0,00560	0,00558	0,00557
		—	$\overline{m}^{I}_{t.gb}$	0,00548	0,00540	0,00536	0,00534	0,00533
		—	$\overline{m}^{II}_{t.gb}$	0,00548	0,00540	0,00536	0,00534	0,00533

Table 4.7 (Continued)

Sr. No.	Formulas for calculation of parameters, mass ratio and coefficients under R=214 m/s	Dimension	Number of approximations	Specific load p N/m²				
				200	250	300	350	400
				Values of parameters, mass ratio, coefficients				
25	$\overline{m}_{tr.sh} = a_{tr.sh} \cdot K_{tr.sh} \cdot L_{t.r} (1-\xi)^{2/3} \left(\frac{\tilde{N}_{0\,max}}{\omega_{tr.sh}} \right)^{2/3} \cdot \frac{1}{p^{1/3}}$; $a_{tr.sh} = 6{,}7 / R^{2/3}\ m^{2/3}/s^2$; $K_{tr.sh} = 0{,}0318\ kg^{1/3} \cdot s^{1/3}/m^{2/3}$; $\xi = 0{,}872$; $\omega_{tr.sh} \cong 314\ s^{-1}$	—	$\overline{m}^{0}$	0,00403	0,00384	0,00370	0,00360	0,00352
		—	$\overline{m}^{I}$	0,00388	0,00371	0,00358	0,00348	0,00340
		—	$\overline{m}^{II}$	0,00388	0,00371	0,00358	0,00348	0,00340
26	$\overline{m}_{tr} = \overline{m}_{m.gb} + \overline{m}_{int.gb} + \overline{m}_{tr.sh} + \overline{m}_{t.gb}$	—	$\overline{m}^{0}$	0,0852	0,0840	0,0835	0,0832	0,0832
		—	$\overline{m}^{I}$	0,0817	0,0807	0,0802	0,0800	0,0800
		—	$\overline{m}^{II}$	0,0818	0,0807	0,0802	0,0800	0,0800
27	$\overline{m}_{pp} = \overline{m}_{eng.s} + \overline{m}_{apu} + \overline{m}_{tr}$	—	$\overline{m}^{0}$	0,271	0,2653	0,2638	0,264	0,2652
		—	$\overline{m}^{I}$	0,2587	0,2539	0,2529	0,2536	0,2551
		—	$\overline{m}^{II}$	0,2588	0,2539	0,2529	0,2536	0,2551
28	$1 - (\overline{m}_{pp} + \overline{m}_{fr} + \overline{m}_{fuel})$	—	zero	0,4653	0,4691	0,4676	0,4637	0,4585
		—	1st	0,4529	0,4554	0,453	0,4482	0,4423
		—	2d	0,4529	0,4554	0,453	0,4482	0,4423

Table 4.7 (End)

Sr. No.	Formulas for calculation of parameters, mass ratio and coefficients under R=214 m/s	Dimension	Number of approxi-mations	Specific load p N/m^2				
				200	250	300	350	400
				Values of parameters, mass ratio, coefficients				
29	$m_{eq} = K_{wiring} \cdot L_{t.r} + K_{el.eq} F_{\Sigma bl} + K_{an.eq} (m_0)^{0,6}$; $K^{1)}_{wiring} = 22...24\ kg/m^2$; $K_{wiring} = 23\ kg/m$; $K_{an.eq} = 2\ kg^{0,4}$; $K^{1)} = 5...6\ kg/m^2$; $K_{el.eq} = 5,5\ kg/m^2$; $F_{\Sigma bl} = 1,8\sigma R^2$; $L_{t.r} = R + R_{t.r} + \delta = 1,185R + \delta$; $\delta = 0,2\ m$		m^0	1185,54	1141,36	1109,03	1083,65	1067,17
			m^I	1011,52	972,88	944,22	922,20	904,60
			m^{II}	1012,52	973,88	944,86	922,4	904,60
30	$m_{u.l} + m_{cr} + m_{eq}$		$zero^0$	5455,54	5411,36	5379,03	5353,65	5333,17
			1st	5281,52	5242,88	5214,22	5192,20	5174,60
			2d	5282,13	5243,08	5214,86	5192,86	5174,8
31	$m_0 = \dfrac{m_{u.l} + m_{cr} + m_{eq}}{1 - (\overline{m}_{pp} + \overline{m}_{fuel} + \overline{m}_{fr})}$		m_0^0	11724,8	11535,6	11503,5	11545,5	11631,8
			m_0^I	11661,6	11512,7	11510,4	11584,6	11699,3
			m_0^{II}	11662,9	11510,6	11511,8	11585	11699,8

In Table 4.8 parameters of helicopter under design are given.

Table 4.8

Parameters of Helicopter Under Design

Sr. No.	Parameters and characteristics of helicopter and its assemblies	Unit	Helicopter under design	MI-8
1	Normal takeoff mass:	kg	11511.6	11100
2	Airframe mass:	kg	2145.9	
	- fuselage	kg	1431.1	
	- landing gear	kg	230.1	
	- horizontal tail unit	kg	29.2	
	- manual control	kg	211.4	
	- powered control	kg	244.1	
3	Power plant mass:	kg	2913.6	
	- engines	kg	376.5	
	- service systems	kg	237.3	
	- main rotor blades	kg	712.6	700
	- main rotor hub	kg	599.6	610
	- anti-torque rotor blades	kg	32.2	
	- anti-torque rotor hub	kg	31.1	
	- gearbox	kg	787.6	785
	- intermediate gearbox	kg	33.4	24.4
	- tail gearbox	kg	61.7	58.7
	- transmission shafts	kg	41.4	
4	Fuel mass	kg	1230.6	1450
5	Main rotor diameter	m	22.28	21.288
6	Main rotor solidity ratio		0.0776	0.0777
7	Number of main rotor blades	Pcs.	5	
8	Main rotor tip speed	m/s	214	
9	Main rotor blade chord	m	0.612	0.520
10	Main rotor blade aspect ratio		18.19	
11	Main rotor speed per minute	rpm	183.6	
12	Anti-torque rotor diameter	m	4.12	3.908
13	Anti-torque rotor solidity ratio		0.1552	0.135
14	Number of anti-torque rotor blades	pcs.	3	
15	Anti-torque rotor tip speed	m/s		
16	Anti-torque rotor blade chord	m	0.340	
17	Anti-torque rotor blade aspect ratio		6.06	
18	Anti-torque rotor speed per minute	rpm	1006.7	
19	Equipment mass	kg	949.8	
20	Auxiliary power unit mass	kg	69.1	
21	Thrust-weight ratio $\tilde{N}_{0\,max}$	kW/N	0.0208	
22	Engine	turboprop	turboprop2-	
23	Takeoff capacity	kW	2 x 1100	
24	Specific load	N/m^2	290	
25	Full-to-takeoff load ratio		0.478	

Table 4.1

Parameter Calculation for Helicopter under Design

Sr. No.	Parameters, specific masses, coefficients for $\omega\cdot R$ =190 m/s		Name	Unit	Specific load **p**, N/m^2				
					110	130	150	170	190
1	*Rotor solidity ratio* $\sigma V_{max} = \frac{C_{TOhov.V_{max}}}{(C_T/\sigma)^{allow}_{V_{max}}}$; $C_{TOhov.V_{max}} = \frac{1.63\cdot p}{(\omega\cdot R)^2}$; $\bar{V}_{max} = \frac{V_{max}}{3.6\cdot\omega\cdot R} = 0.278$; $(C_T/\sigma)^{allow}_{V_{max}} = 0.297 - 0.36\cdot\bar{V}_{max} = 0.197$; $\sigma H_{dyn} = \frac{C_{TH_{dyn}}}{(C_T/\sigma)^{allow}_{H_{dyn}}}$; $C_{T\ H\ dyn} = \frac{1.63\cdot p}{(\omega\cdot R)^{2\sim}_{H\ dyn}}$; $(C_T/\sigma)^{allow}_{H_{dyn}} = 0.297 - 0.36\cdot\bar{V}^{cr}_{H_{dyn}}$; $\sigma^{allow}_{min} = max\{\sigma_{V\,max}, \sigma_{Hdyn}\}$; $\sigma_{ser} = 1.03\cdot\sigma^{allow}_{min}$		$\sigma_{V\,max}$	–	*0.0252*	*0.0298*	*0.0344*	*0.0390*	*0.0435*
			$\sigma_{H\ dyn}$	–	0.0319	0.0381	0.0444	0.0507	0.0572
			σ^{allow}_{min}	–	0.0319	0.0381	0.0444	0.0507	0.0572
			σ_{ser}	–	0.0329	0.0392	0.0457	0.0523	0.0589
2	Rotor radius $R = \sqrt{\frac{m_0\cdot g}{\pi\cdot p}}$	$m_{00} = 590.9$ kg	R	m	4.096	3.767	3.507	3.295	3.116
		$m_0^{I} = 626.131$ kg	R	m	4.216	3.878	3.610	3.391	3.208
		$m_0^{\sim} = 613.000$ kg	R	m	4.172	3.837	3.572	3.356	3.174
3	Minimal necessary number of blades $Z = \frac{\pi}{0.055}\cdot\sigma_{ser}$		z	–	1.877	2.241	2.610	2.984	3.361
			z	–	1.879	2.244	2.614	2.988	3.367
			z	–	1.878	2.243	2.613	2.987	3.365
4	Number of blades $z = \mathrm{ceil}(z)$		z	pc	2	3	3	3	4
5	Rotor solidity ratio $\sigma = \frac{z}{57.1}$		σ^0	–	0.035	0.053	0.053	0.053	0.070
			σ^{I}	–	0.035	0.053	0.053	0.053	0.070
			σ^{II}	–	0.035	0.053	0.053	0.053	0.070
6	Fuselage mass ratio $\bar{m}_{fus} = K_{fus}\cdot\frac{S_{fus}^{0.88}}{m_0^{0.75}}$; $K_{fus} = 2.3$ kg$^{0.75}$/m$^{1.76}$; $S_{fus} \approx 13.425$ m^2		$\bar{m}^0_{fus}$	–	0131	0.131	0.131	0.131	0.131
			$\bar{m}^{I}_{fus}$	–	0.126	0.126	0.126	0.126	0.126
			$\bar{m}^{\sim}_{fus}$	–	0.128	0.128	0.128	0.128	0.128

Table 4.1 (Continued)

Sr. No.	Parameters, mass ratio, coefficients for $\omega \cdot R$ =190 m/s	Name	Unit	Specific load **p**, N/m^2				
				110	130	150	170	190
7	Landing gear mass ratio $\overline{m}_{LG} = K_{LG}$	$\bar{m}^{0}_{LG}$ $\bar{m}^{I}_{LG}$ $\bar{m}^{\sim}_{LG}$	–	0.020	0.020	0.020	0.020	0.020
8	Tail unit mass ratio $\overline{m}_{HE} = K_{HE} \cdot \dfrac{S_{HE}}{p}$; $\bar{S}_{HE} = 0.00725$	$\bar{m}^{0}_{HE}$ $\bar{m}^{I}_{HE}$ $\bar{m}^{\sim}_{HE}$	–	2.59E-03	2.59E-03	2.59E-03	2.59E-03	2.59E-03
9	Manual control mass ratio $\overline{m}_{man.con} = K_{man.con} \cdot \dfrac{S_{HE}}{m_0}$; $K_{man.con} = 7.2$ kg/m	$\bar{m}^{0}_{man.c}$	–	0.050	0.046	0.043	0.040	0.038
		$\bar{m}^{I}_{man.c}$	–	0.048	0.045	0.042	0.039	0.037
		$\bar{m}^{\sim}_{man.c}$	–	0.049	0.045	0.042	0.039	0.037
10	Airframe structural mass ratio $\overline{m}_{fr} = \overline{m}_{fus} + \overline{m}_{LG} + \overline{m}_{HE} + \overline{m}_{man.con.}$	$\bar{m}^{0}_{fr}$	–	0.204	0.199	0.196	0.193	0.191
		$\bar{m}^{I}_{fr}$	–	0.197	0.192	0.189	0.186	0.184
		$\bar{m}^{\sim}_{fr}$	–	0.199	0.195	0.192	0.189	0.186
11	Specific thrust-weight ratio required for flight under the service ceiling $\tilde{N}_{H_{dyn0}} = \dfrac{1}{\bar{N}_{nom} \cdot \bar{N}_{H_{dyn}} \cdot \bar{N}_{V_{cr}} \cdot \xi_{cr}} \cdot (16.4 \cdot 10^{-3} \cdot \omega R \cdot \left[1 + 7.08 \cdot 10^{-8} \left(V^{cr}_{dyn}\right)\right] +$ $+1.82 \cdot \dfrac{p}{V^{cr}_{dyn} \cdot^{\sim}_{dyn}} + 13.2 \cdot 10^{-3} \cdot \bar{S}_{cr} \cdot \left(V^{cr}_{dyn}\right)^3 \cdot^{\sim}_{dyn}$ $V^{cr}_{dyn} = 164 \cdot \sqrt[4]{\dfrac{1.09 \cdot p}{\left(\omega \cdot R + 11.6 \cdot 10^6 \cdot \bar{S}_{t/wr} \cdot^{\sim}_{dyn}\right) \cdot^{\sim}_{dyn}}}$;	$\tilde{N}^{0}_{H_{dyn}}$	$\frac{Vt}{H}$	13.172	14.168	15.125	16.050	16.948
		$\tilde{N}^{I}_{H_{dyn}}$		13.132	14.122	15.074	15.994	16.887
		$\tilde{N}^{\sim}_{H_{dyn}}$		13.147	14.139	15.093	16.015	16.909
		$\bar{N}^{0}_{V_{cr}}$	–	1.0067	1.0073	1.0079	1.0084	1.0089
		$\bar{N}^{I}_{V_{cr}}$	–	1.0068	1.0074	1.0080	1.0085	1.0090
		$\bar{N}^{\sim}_{V_{cr}}$	–	1.0068	1.0074	1.0079	1.0084	1.0089
		V^{crI}_{dyn}	$\frac{km}{hr}$	111.273	116.019	120.245	124.067	127.565

Table 4.1 (Continued)

Sr. No.	Parameters, mass ratio, coefficients for $\omega \cdot R$ =190 m/s	Name	Unit	Specific load **p**, N/m^2				
				110	130	150	170	190
11	$\bar{S}_{cr} = \dfrac{\chi \cdot \Sigma(C_x \cdot S)}{m_0 \cdot g} = 9.16 \cdot 10^{-5};\ \Sigma(C_x \cdot S) = 0.0174 \cdot m_0^{0.5346}$	$V_{dyn}^{cr\,I}$	$\frac{km}{hr}$	111.273	116.019	120.245	124.067	127.565
		$V_{dyn}^{cr\,-}$	$\frac{km}{hr}$	111.060	115.796	120.014	123.829	127.320
		$V_0^{cr\,0}$	$\frac{km}{hr}$	91.128	95.014	98.475	101.605	104.470
		$V_0^{cr\,I}$	$\frac{km}{hr}$	91.648	95.556	99.037	102.184	105.066
		$V_0^{cr\,-}$	$\frac{km}{hr}$	91.457	95.358	98.831	101.972	104.848
12	Thrust-weight ratio required for flight under V_{max} $\tilde{N}_{V_{max\,0}} = \dfrac{1}{\bar{N}_{HV_{max}} \cdot \bar{N}_{V_{max}} \cdot \bar{N}_{V_{cr}} \cdot \xi_{V_{max}}} \cdot (16.4 \cdot 10^{-3} \cdot \omega R \cdot (1 + 7.08 \cdot 10^{-8} \cdot V_{max}^3) +$ $+1.67 \cdot \dfrac{p \cdot I_{cr}}{V_{max}} + 13.2 \cdot 10^{-3} \cdot \bar{S}_{cr} \cdot V_{max}^3\ )$ $I_{cr} = 1.02 + 0.0004 \cdot V_{max} = 1.096;\ \bar{N}_{V_{max\,0}} = 1 - 0.0695 \cdot H_{V_{max}}^{-3} = 0.965;$ $\bar{N}_{V_{max}} = 1 + 5.5 \cdot 10^{-7} \cdot V_{max}^2 = 1.02$	$\tilde{N}^0_{V\,max\,0}$	$\frac{Vt}{H}$	16.280	16.503	16.727	16.951	17.174
		$\tilde{N}^{I}_{V\,max\,0}$		16.023	16.247	16.471	16.694	16.918
		$\tilde{N}^{II}_{V\,max\,0}$		16.116	16.340	16.564	16.787	17.011
13	Thrust-weight required for hovering under H_{hov} $\tilde{N}_{H_{hov}} = \dfrac{0.6385 \cdot \bar{T}^{3/2} \cdot \sqrt{p}}{\bar{N}_{H_{hov}} \cdot \xi_0 \cdot \eta_0 \cdot \sqrt{\tilde{\ }_{H_{hov}}}}$	$\tilde{N}^0_{H_{hov\,0}}$	$\frac{Vt}{H}$	13.407	14.639	15.794	16.888	17.931
		$\tilde{N}^{I}_{H_{hov\,0}}$		13.389	14.616	15.765	16.852	17.890
		$\tilde{N}^{-}_{H_{hov\,0}}$		13.395	14.624	15.775	16.865	17.905
14	Required specific thrust-weight ratio $\tilde{N}_{0_{max}} = max\left\{\tilde{N}_{H_{hov}\ 0}, \tilde{N}_{V_{max}\ 0}, \tilde{N}_{H_{hov}\ 0}\right\}$	$\tilde{N}^0_{0\,max}$	$\frac{Vt}{H}$	16.280	16.503	16.727	16.951	17.931
		$\tilde{N}^{I}_{0\,max}$		16.023	16.247	16.471	16.852	17.890
		$\tilde{N}^{II}_{0\,max}$		16.116	16.340	16,564	16.865	17.905
15	Calculated power of power unit: $N_{0\,max} = \tilde{N}_{0\,max} \cdot m_0 \cdot g$	$N^0_{0\,max}$	kW	94.37	95.66	96.96	98.26	103.94
		$N^{I}_{0\,max}$		98.42	99.79	101.17	103.51	109.89
		$N^{II}_{0\,max}$		96.92	98.26	99.61	101.42	107.67

Table 4.1 (Continued)

Sr. No.	Parameters, mass ratio, coefficients for $\omega \cdot R$ =190 m/s	Name	Unit	Specific load **p**, N/m^2				
				110	130	150	170	190
16	Fuel mass ratio $\overline{m}_{fuel} = 1.19 \cdot \frac{C_{e\ cruis} \cdot L}{V_{cruis}} \cdot 0.765 \cdot \tilde{N}_{0\ max} \cdot g$; $C_{e\ cruis} = C_{e\ takeoff} \cdot \overline{C}_{eH} \cdot \overline{C}_{eV} \cdot \overline{C}_{et} \cdot \overline{C}_{eN}$; $C_{e\ V} = 1 - 3 \cdot 10^{-7} \cdot V_{cruis}^{2} = 0.993$; $C_{eH} = 0.995$; $C_{et} = 1.0$; $\overline{C}_{eN} = 1.075$; $\overline{C}_{e\ takeoff} = 1.14 \cdot \left(N_{takeoff} \cdot 10^{-3}\right)^{-0.236}$; $V_{max} = 082 \cdot V_{cruis} = 155.8$ km/h	$\overline{m}^{0}_{fuel}$	–	0.124	0.125	0.126	0.127	0.133
		$\overline{m}^{I}_{fuel}$	–	0.120	0.122	0.123	0.125	0.131
		$\overline{m}^{-}_{fuel}$	–	0.122	0.123	0.124	0.126	0.132
		$C^{0}_{e\ cruis}$		0.414	0.413	0.411	0.410	0.405
		$C^{I}_{e\ cruis}$	$\frac{kg}{kWt \cdot hr}$	0.410	0.408	0.407	0.405	0.399
		$C^{-}_{e\ cruis}$		0.411	0.410	0.409	0.407	0.401
		$C^{0}_{e\ takeoff}$		0.390	0.389	0.387	0.386	0.381
		$C^{I}_{e\ takeoff}$	$\frac{kg}{kWt \cdot hr}$	0.386	0.385	0.383	0.381	0.376
		$C^{-}_{e\ takeoff}$		0.387	0.386	0.385	0.383	0.378
17	Engine with systems mass ratio $\overline{m}_{eng.s} = \left(\gamma_{eng} + K_{s}\right) \cdot \tilde{N}_{0\,max} + K_{fuel.s} \cdot \overline{m}_{fuel}$; $\gamma_{eng} = 0.757$; $K_{s} = 0.033$; $K_{fuel.s} = 0.017$	$\overline{m}^{0}_{eng.s}$	–	0.128	0.130	0.132	0.134	0.141
		$\overline{m}^{I}_{eng.s}$	–	0.126	0.128	0.130	0.133	0.141
		$\overline{m}^{-}_{eng.s}$	–	0.127	0.129	0.130	0.133	0.141
18	Total mass ratio of rotor blades $\overline{m}_{\Sigma bl} = a_{bl} \cdot K_{bl} \cdot \frac{\sigma}{\lambda^{0.7} \cdot p}$; $a_{bl} = 23.62 \cdot R^{0.7}$; $K_{bl} = 17$	$\overline{m}^{0}_{\Sigma bl}$	–	0.045	0.054	0.044	0.038	0.043
		$\overline{m}^{I}_{\Sigma bl}$	–	0.046	0.055	0.045	0.038	0.044
		$\overline{m}^{-}_{\Sigma bl}$	–	0.046	0.055	0.045	0.038	0.044
19	Main rotor hub mass ratio $\overline{m}_{hub} = a_{hub} \cdot 10^{-5} \cdot K_{hub} \cdot K_{z} \cdot z \cdot \overline{m}_{bl}^{1.35} \cdot (\omega \cdot R)^{2.7} \cdot p^{0.35}$; $a_{hub} = 2.34 \cdot R^{-0.65}$	$\overline{m}^{0}_{hub}$	–	0.036	0.044	0.038	0.033	0.038
		$\overline{m}^{I}_{hub}$	–	0.036	0.045	0.038	0.033	0.039
		$\overline{m}^{-}_{hub}$	–	0.036	0.045	0.038	0.033	0.038
20	Total mass ratio of anti-torque rotor blades $\overline{m}_{\Sigma a-t.r.bl.} = \frac{\sigma_{a-t.r}}{\sigma} \cdot \left(\frac{\lambda}{\lambda_{a-t.r}}\right)^{0.7} \cdot \left(\frac{R_{a-t.r}}{R}\right)^{2.7} \cdot \overline{m}_{\Sigma bl}$	$\overline{m}^{0}_{\Sigma a-t.r.bl.}$	–	1.22E-03	1.95E-03	1.60E-03	1.35E-03	1.90E-03
		$\overline{m}^{I}_{\Sigma a-t.r.bl}$	–	1.25E-03	1.99E-03	1.64E-03	1.38E-03	1.94E-03
		$\overline{m}^{-}_{\Sigma a-t.r.bl}$	–	1.24E-03	1.97E-03	1.62E-03	1.37E-03	1.93E-03

Table 4.1 (Continued)

Sr. No.	Parameters, mass ratio, coefficients for $\omega \cdot R$ =190 m/s	Name	Unit	Specific load **p**, N/m^2				
				110	130	150	170	190
21	Anti-torque rotor mass ratio $\overline{m}_{a-t.r.hub} = \frac{K_{z\ a-t.r}}{K_z} \cdot \frac{Z_{a-t.r}}{z} \cdot \left(\frac{(\omega \cdot R)_{a-t.r}}{\omega \cdot R}\right)^{2.7} \cdot \left(\frac{R}{R_{a-t.r}}\right)^{0.65} \cdot \left(\frac{\overline{m}_{a-t.r.bl.}}{\overline{m}_{bl}}\right)^{1.35} \cdot \overline{m}_{hub}$	$\bar{m}^{0}_{a-t.r.hub}$	–	8.70E-05	1.05E-04	8.94E-05	7.75E-05	8.95E-05
		$\bar{m}^{I}_{a-t.r.hub}$	–	8.78E-05	1.06E-04	9.02E-05	7.81E-05	9.02E-05
		$\bar{m}^{I}_{a-t.r.hub}$	–	8.75E-05	1.06E-04	8.99E-05	7.79E-05	9.00E-05
22	Rotors mass ratio $\overline{m}_r = \bar{m}_{\Sigma bl} + \bar{m}_{hub} + \bar{m}_{\Sigma a-t.r.bl} + \bar{m}_{a-t.r.hub}$	$\bar{m}^{0}_{r}$	–	0.082	0.100	0.084	0.072	0.083
		$\bar{m}^{I}_{r}$	–	0.083	0.102	0.085	0.073	0.085
		$\bar{m}^{-}_{r}$	–	0.083	0.101	0.085	0.072	0.084
23	Gearbox mass ratio $\overline{m}_{m.gb} = a_{m.gb} \cdot K_{m.gb} \cdot \xi^{0.8} \cdot (1-\xi)^{0.8} \cdot \left(\frac{\tilde{N}_{0\,max}}{\omega \cdot R}\right)^{0.8} \cdot p^{-0.2}$; $a_{m.gb} = 7.8 \cdot R^{-0.4}$; $K_{m.gb} = 0.121$	$\bar{m}^{0}_{m.gb}$	–	0.081	0.076	0.073	0.070	0.070
		$\bar{m}^{I}_{m.gb}$	–	0.081	0.076	0.073	0.071	0.071
		$\bar{m}^{-}_{m.gb}$	–	0.081	0.076	0.073	0.070	0.071
24	Tail rotor gearbox mass ratio $\overline{m}_{t.r.gb} = a_{t.r.gb} \cdot K_{t.r.gb} \cdot L_{t.r.} \cdot (1-\xi)^{0.8} \cdot \left(\frac{\tilde{N}_{0\,max} \cdot R_{t.r}}{\omega \cdot R}\right)^{0.8} \cdot p$; $a_{t.r.gb} = 7.8 \cdot R^{-0.4}$; $K_{t.r.gb} = 0.127$; $L_{t.r.} = 4$ m	$\bar{m}^{0}_{t.r.gb}$	–	0.019	0.018	0.017	0.016	0.016
		$\bar{m}^{I}_{t.r.gb}$	–	0.019	0.018	0.017	0.016	0.016
		$\bar{m}^{-}_{t.r.gb}$	–	0.019	0.018	0.017	0.016	0.016

Table 4.1 (End)

Sr. No.	Parameters, mass ratio, coefficients for $\omega \cdot R$ =190 m/s	Name	Unit	Specific load **p**, N/m^2				
				110	130	150	170	190
25	Transmission shaft mass ratio $\overline{m}_{tr.sh} = a_{tr.sh} \cdot K_{tr.sh} \cdot L_{t.r.} \cdot (1-\xi)^{2/3} \cdot \left(\frac{\tilde{N}_{0max}}{\omega_{tr.sh}}\right)^{2/3} \cdot p^{-1/3}$; $a_{tr.sh} = 6.7 \cdot R^{2/3}$; $K_{tr.sh} = 0.103$	$\bar{m}^{0}_{tr.sh}$	–	0.0082	0.0083	0.0083	0.0084	0.0087
		$\bar{m}^{I}_{tr.sh}$	–	0.0079	0.0080	0.0081	0.0082	0.0086
		$\bar{m}^{\sim}_{tr.sh}$	–	0.0080	0.0081	0.0082	0.0083	0.0086
26	Transmission mass ratio $\overline{m}_{tr} = \overline{m}_{m.gb} + \overline{m}_{t.r.gb} + \overline{m}_{tr.sh}$	$\bar{m}^{0}_{tr}$	–	0.108	0.102	0.098	0.095	0.095
		$\bar{m}^{I}_{tr}$	–	0.107	0.102	0.098	0.095	0.096
		$\bar{m}^{\sim}_{tr}$	–	0.107	0.102	0.098	0.095	0.096
27	Power unit mass ratio $\overline{m}_{pp} = \overline{m}_{eng.s} + \overline{m}_{v} + \overline{m}_{cargo}$	$\bar{m}^{0}_{pp}$	–	0.318	0.333	0.314	0.300	0.320
		$\bar{m}^{I}_{pp}$	–	0.317	0.332	0.313	0.301	0.321
		$\bar{m}^{\sim}_{pp}$	–	0.317	0.332	0.313	0.300	0.321
28	$1-(\overline{m}_{pp} + \overline{m}_{fr} + \overline{m}_{fuel})$	va^{0}	–	0.355	0.343	0.364	0.380	0.357
		va^{I}	–	0.366	0.354	0.375	0.388	0.364
		va^{II}	–	0.362	0.350	0.371	0.385	0.361
29	Helicopter takeoff mass $\overline{m}_0 = \frac{m_{u.l.} + m_{crew} + m_{eq}}{1-\left(\overline{m}_{pp} + \overline{m}_{fr} + \overline{m}_{fuel}\right)}$	m^{0}_{0}	kg	671.17	693.63	653.35	626.98	667.44
		m^{I}_{0}	kg	650.47	672.29	634.10	613.43	654.32
		m^{II}_{0}	kg	657.80	679.84	640.92	617.56	658.98

5. METHODOLOGY OF INTEGRATED HELICOPTER DESIGN AND MODELLING

No company can be considered commercially viable if it cannot provide high quality of the production samples, their easy modernization and replacement. The number of available modifications is usually considerable.

High quality of production and manufacturing is mostly provided by computer integrated systems CAD/CAM/CAE which allow helicopter designers integrate helicopter design, engineering analysis, and preproduction and solve all the problems occurring at different stages of helicopter life cycle.

As we already know, modern helicopters:

– provide both engineering sophistication and high operating quality due to many scientific and technical decisions and inventions in aerodynamics, helicopter design, engineering, strength, weight perfection, power plants, helicopter systems, equipment, materials, production technology, operating convenience, reliability, and safety;

– comply with up-to-date quality standards, advanced environmental standards, aircraft airworthiness regulations (Aviation Rules), AR 27 and AR 29, harmonized in structure and requirements with FAR (JAR);

– demonstrate high level of structural, technological, and operating unification and portability;

– are economically efficient;

– implement integrated design technologies as well as technologies of preproduction, engineering analysis, tests, certification, information support of the aviation complexes, namely CAD/CAM/CAE/PLM and ERP systems (Fig. 5.1).

Formerly, design methods applied for assembled helicopter structures were based on two-dimensional models and loft alignment though didn't take all structural and technological peculiarities into account. This led to the invention of new integrated design methods.

The concept of integrated design in the assembled helicopter structures forms the methodological basis for the development of bindings for airframe power plants, predetermined static strength, service life, tightness, and quality of the outer surface under minimal coupling mass.

Integrated design methods combine helicopter design with computer parametric three-dimensional modelling (3D models) of helicopter structures. As helicopters consist of many parts, assemblies, units, and components interattached through different types of detachable and nondetachable joints, design and accomplishment quality of these joints influence such helicopter characteristics as weight, service, aerodynamic, etc.

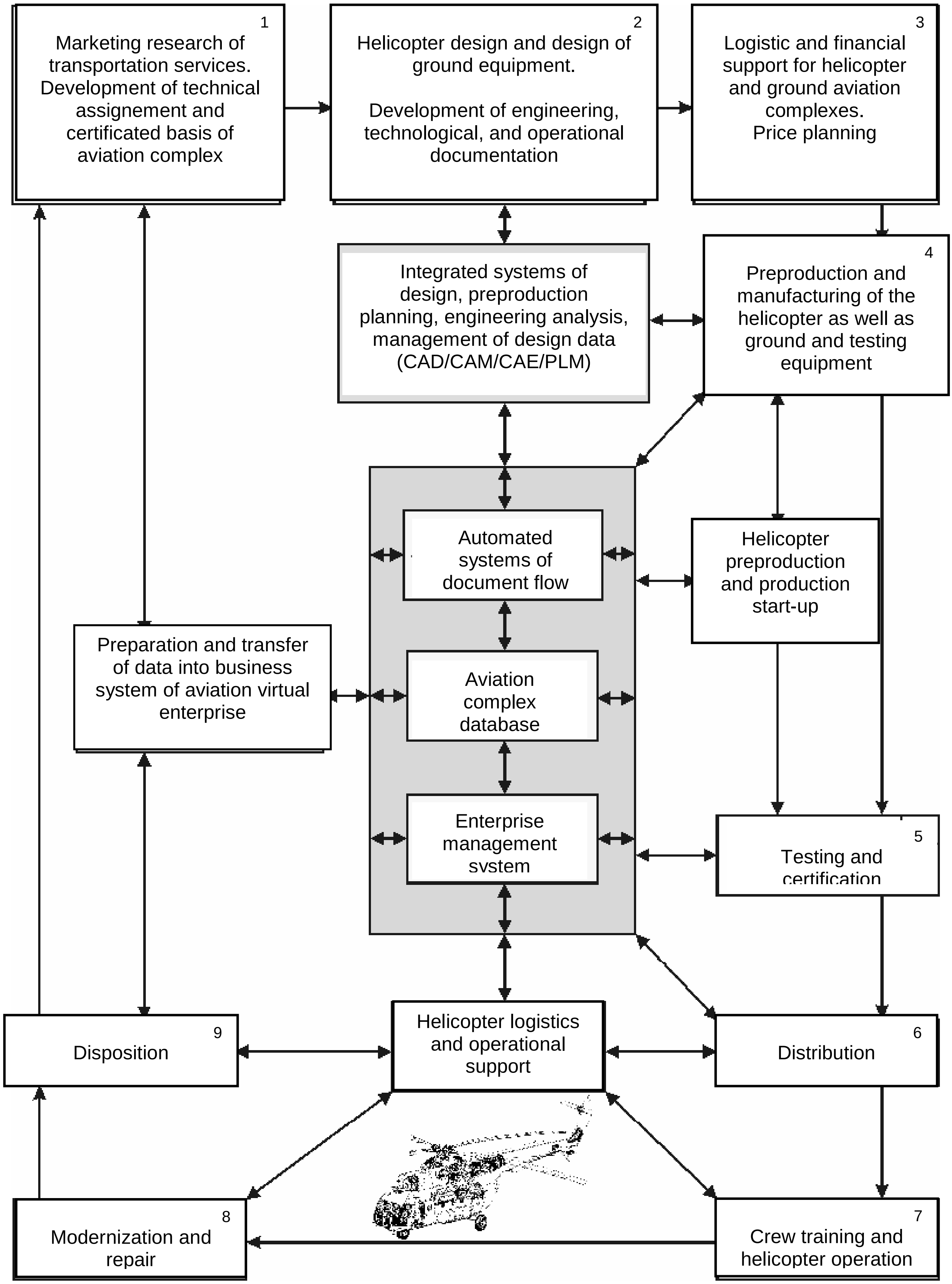


Fig. 5.1. Helicopter Life Cycle and Its Information Support

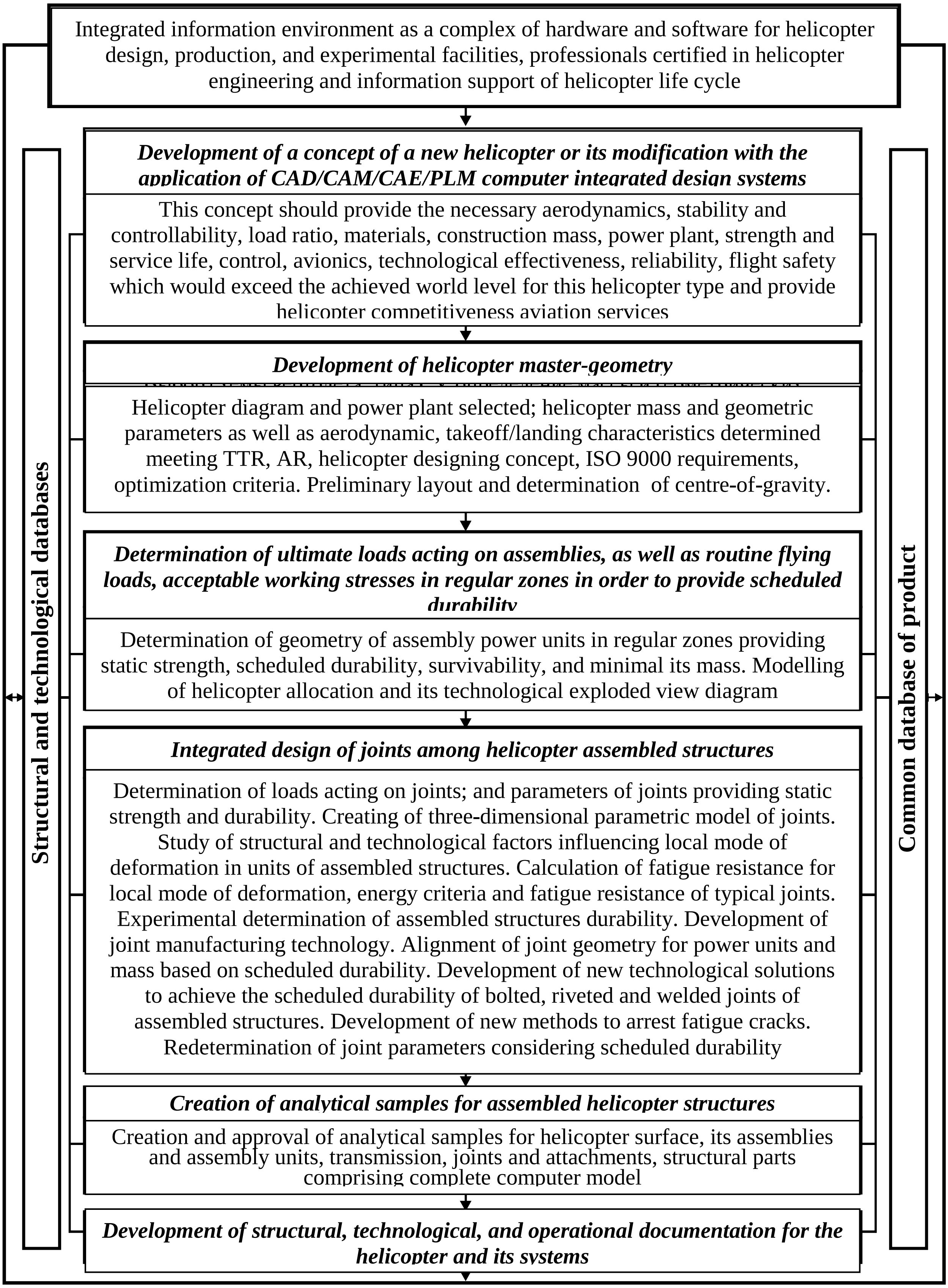


Fig. 5.2. Integrated Concept Design for Assembled Helicopter Structures

Integrated design of helicopter structures can be divided into several stages (Fig. 5.2):

1. Building of integrated environment, complex of technical and software tools for helicopter design, manufacturing, and experimental base.
2. Development of a concept of a new helicopter or its modification with the application of CAD/CAM/CAE/PLM computer integrated design systems.
3. Development of a parametric model of the helicopter master-geometry with the help of CAD/CAM/CAE/PLM system.
4. Load calculation aimed at determining helicopter components, standard flight loads, regular working stresses required for scheduled durability.
5. Integrated design of joints for the assembled helicopter structures.
6. Engineering of the analytical samples of the assembled helicopter structures.
7. Development of the necessary engineering, technological, and operating documentation.

Integrated design of the assembled helicopter structures are carried out in one database with the application of engineering and technological data.

Principles of integrated design of the assembled helicopter structures are based on the suggested concept (Fig. 5.2).

Aviation technology and manufacturing is highly interested in modern and efficient integrated systems which secure high quality, durability, reliability and service life, certifications for designing, engineering, preproduction technologies, serial production, flight tests on the base of the Continuous Acquisition and Life Cycle Support technology (CALS-technologies).

Computer technologies together with advanced design and manufacturing technologies provide gradual increase in labour productivity and helicopter quality together with significant reduction of the delivery time, thus leading to new up-to-date helicopters which would satisfy even the most demanding customers.

To form the information space, engineering, manufacturing, and operational databases are integrated into one database.

Information environment helps to solve the task of providing transportation safety under minimum expenditures for a ton-kilometre or passenger-kilometre as well as reduced costs of helicopter life cycle.

In order to monitor the product during its life cycle, one database must contain data on aviation technology, manufacturing companies, and service centres with the description of all organizational, engineering, and technological processes. All world aviation companies employ modern methods and ideas of aviation product monitoring in its life cycle and integrated information technologies based on them.

Computer technologies intensify the work on technical documentation, engineering, technological preproduction, production control, product monitoring; and provide information support of the product life cycle.

When developing a new product, helicopter designers use integrated CAD/CAM/CAE/PLM systems for the issues of engineering and technological preproduction, incorporating helicopter structure, materials, parts, joints, and assembly parts.

Integrated computer technologies reduce the expenditures on engineering, production and monitoring of helicopter operation, raises labour productivity, quality and competitive ability of the manufactured product.

Computer technologies of integrated helicopter design apply parametric analytical samples for calculating aerodynamics and strength; service life and survivability; helicopter mass and its centre-of-gravity; structure dynamics and its operating safety as well as for technological preproduction and quality control, operation and repair, etc.

As we know, helicopter design aims at developing reliable, fail-safe helicopter of minimal mass with the predetermined service life. It is specified in airworthiness regulations and aviation rules and poses certain economic problems.

Nowadays, low transportation costs and high flight safety are the criteria used to evaluate quality of civil helicopters. Helicopter and airframe designers reach conceptually predetermined quantitative criteria by:

– reducing structure mass that reduces direct operational costs due to the growth of payload;

– increasing service life and life cycle by increasing flight reliability and safety and reducing expenditures on depreciation, service, and repair.

Methodology of integrated helicopter design is based on 5 principles:

1. Principle of using analytical samples of helicopter structures

3D-models of master-geometry, space allocation, and analytical samples of helicopter structures are created by the methods of analytical geometry with the help of integrated CAD/CAM/CAE/PLM systems in one information space to support helicopter life cycle.

2. Principle of using master-geometry for helicopter images

Included in the image, helicopter parameters such as minimal mass and scheduled durability must meet TTR, AR, new helicopter concept. They are determined by the algorithm:

Terms of reference $\rightarrow$ ***Helicopter scheme*** $\rightarrow$ ***TTR***, ***AR*** $\rightarrow \mathbf{m}_{\boldsymbol{0}} \rightarrow$ $\rightarrow \mathbf{m}_{\boldsymbol{0}\,\mathbf{min}} \rightarrow \mathbf{p}_{\mathbf{opt}} \rightarrow \bar{\mathbf{N}}_{\mathbf{opt}} \rightarrow \mathbf{N}_{\boldsymbol{0}} \rightarrow \sigma_{\mathbf{i}} \rightarrow$ ***blade airfoil portion*** $\rightarrow$ $\rightarrow \boldsymbol{(R,\ \lambda_i,\ \eta_i,\ D_{fus.},\ L_{t.rot.},\ L_{V.E.})} \rightarrow \varphi_{CG} \rightarrow$ ***analytical sample of helicopter surface***,

$$\boldsymbol{m}_0 = \frac{\boldsymbol{m}_{cr} + \boldsymbol{m}_{equip.\ cont.} + \boldsymbol{m}_{pl}}{1 - \left[\bar{\boldsymbol{m}}_{mast}\left(\boldsymbol{p},\ \boldsymbol{n}_p,\ \boldsymbol{N}_{sched},\ \boldsymbol{O}_{main}\right) + \right]\bar{\boldsymbol{m}}_{pp}\left(\boldsymbol{p},\ \bar{\boldsymbol{N}}_0,\ \gamma_{eng},\ \boldsymbol{R},\ \boldsymbol{N}_{eng}\right) + \bar{\boldsymbol{m}}_{fuel}\left(\boldsymbol{p},\ \boldsymbol{C}_{fuel},\ \boldsymbol{k},\ \boldsymbol{L}\right)}.$$

Stages of helicopter parametric modelling and model variants of the master-geometry for its assemblies are shown in Fig. 5.3. Each stage is subjected to multiple iterations which follow each other thereby providing sufficient level of design pro-development.

3. Principle of regular zones in helicopter structures

Design parameters and implementation technology of regular zones in helicopter structures must provide the perception of calculated breaking loads, scheduled durability under loads equivalent to standard flight loads in the operational environment, specified coefficient of the fatigue quality $\boldsymbol{k_y}$, specified quality of the outer surface, and tightness degree; and meet the following conditions:

$$P_{DEV} \geq P_{CAL}\left(DP_{RZ}, \sigma_{DRZ}\left(N_{SCHED\ RZ}\right)\right);$$

$$N_{SCHED\ RZ} \geq N_{CAL\ RZ}\left(DP_{RZ},\ \sigma_{0EQU},\ \sigma_K, IT\right);$$

$$\Delta_3 < 0 \text{ under } P \geq P_{OPER};\ \Delta h \leq 0.05\ mm;\ k_y \leq 3.$$

4. Principle of irregular areas in helicopter structures

Design parameters and implementation technology of irregular zones in helicopter structures must provide the perception of rated forces zone under static loading, scheduled durability, quality of outer surface and tightness equal to characteristics of regular zones or exceeding them, and meet the following conditions:

$$P_{DEV} \geq P_{CAL}\left(DP_{IRZ}, \sigma_{DIRZ}\left(N_{SCHED\ IRZ}\right)\right);\ \Delta h_{IRZ} \leq \Delta h_{RZ};\ \Delta h_{3\ RZ} \leq \Delta h_{3\ RZ};$$

$$N_{SCHED} \geq min\left[N_{CAL\ IRZ}\left(DP_{IRZ},\ \left(\sigma_{0EQU} \cdot \varepsilon_{EQU}\right),\ \sigma_K, IT\right)\right];$$

$$N_{OPER}\left(DP_{IRZ},\ \sigma_0,\ \sigma_K, IT\right).$$

5. Principle of providing survivability of structures with fatigue cracks

Design parameters of fail-safe assembled helicopter structures must provide means to control critical locations and detect fatigue cracks. This goal can be achieved only through employing advanced methods of delaying crack growth, recovering load-carrying capability, and providing tightness of damaged structures; these methods based on the following inequalities:

$$\left(N_{OST\ SZRTU}/N_{OST\ STR}\right) \geq 1;\ \Delta_{3\ SZRTU} < 0.$$

To realize these principles, scientific methods must appear of integrated design.

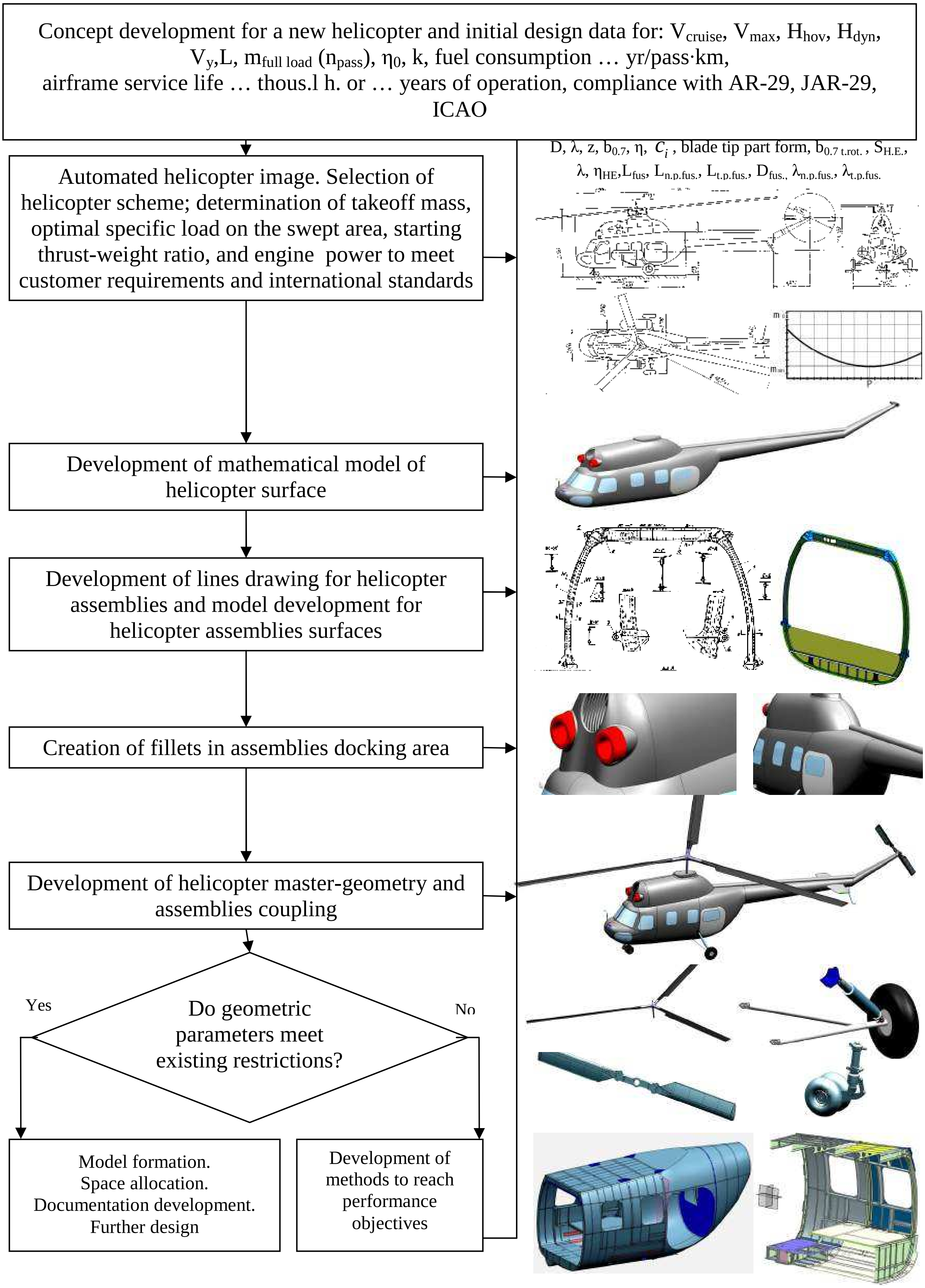


Fig. 5.3. Technology for developing master-geometry of some helicopter assemblies

Helicopter performance will improve only with the progress in aerodynamic quality.

Main rotor can be improved by employing new optimized aerodynamic airfoils, optimizing blade tip shapes, and bending down angles of blade tips. Aerodynamic efficiency of the main rotor will grow e.g. with specified geometric twists of the blade, extended and non-rectangular blade shapes, etc. These improvements will increase the relative efficiency coefficient of hovering rotor by ***3…5%*** and raise its maximum aerodynamic efficiency in forward flight by ***7…10%***.

To grow, helicopter aerodynamic efficiency needs reduced resistance of non-load carrying helicopter elements, namely the resistance caused by interference between individual structure elements. In particular, it is advisable to use retractable landing gear; bring down resistances of sleeves, main and tail rotors; improve flow between the main rotor and the fuselage; apply boundary layer suction or blowing air for additional drag reduction.

Computer technology and computer systems such as CATIA and SIEMENS NX made integrated design possible for assembled helicopter structures.

Modern computer project is a system of engineering, technological, and calculation models as well as helicopter lifecycle management data provided for certification, quality control, operating servicing, and recycling.

Geometry model (or analytical standard) is the basic, primary element of computer-aided design of a new engineering unit. Helicopter computer project includes 4 models:

– helicopter master geometry (helicopter surface model revealing all points lying on helicopter surface);

– model of helicopter inner space;

– model of connections and joints of helicopter technological breaks;

– model of the whole product geometry (analytical standards of all parts, units, assemblies, and helicopter as a whole), i.e. full computer model of the helicopter.

Development of these models involves a number of successive stages.

Helicopter master geometry development comprises the development of:

1) a helicopter mathematical model;
2) lines drawing of assemblies;
3) models for assembly surfaces combined into one model of helicopter surface;
4) a frame (applies tracing of basis surfaces of primary structural members in form of lines drawings and structure-power diagrams).

Model of helicopter inner space needs:

1) design and technology division;
2) panelling of fuselage, tail boom and tail unit;
3) transmission circuit developed;

4) primary structural members established;
5) structural elements of a fuselage, tail boom, and tail unit developed;
6) main rotor structural elements developed;
7) anti-torque rotor structural elements developed;
8) equipment, appliances, finished products, etc. located;
9) systems arranged;
10) cockpit arranged;
11) cargo/passenger compartment arranged;
12) assemblies and systems attached;
13) helicopter design tree provided.

Model of connections and joints of helicopter technological breaks expects:

1) properly determined structures of connections and joints;
2) clearly identified role of interconnected tolerance system for joint elements.

This model development includes the following stages:

1. Zone modelling of:
   – dimensionally accurate structure components with all connections and couplings;
   – structural attachment elements;
   – shapes of assemblies and appliances with accurate binding of attachment elements as well as structure elements check to determine interpenetration, clearances, and assemblability.
2. Modelling and attribute information of:
   – analytical standards of all structure elements;
   – drawings database.
3. Assembled modelling of the systems that pass through the assemblies (without technological exploded view).
4. Collection and control of all design data.

In integrated helicopter design, parametric models a based on the data from general CAD helicopter design, including helicopter diagram, weight and geometric data, performance characteristics, etc.

Model of the whole product geometry is developed by the computer integrated systems such as SIEMENS NX and CATIA.

Fig. 5.4 shows the example of how the model of the whole product geometry is developed.

Mathematic model of helicopter surface is a system of adequate algorithms and numeric data required to determine geometric parameters of the surface arbitrary point. It should meet the following requirements:

– provide full and one-valued description of helicopter surface;
– include slick long and cross sections; designed surfaces blended with many analytical surfaces (cylindrical, tapered, parabolic, plain, etc.);
– visualize the form of the surface under design.

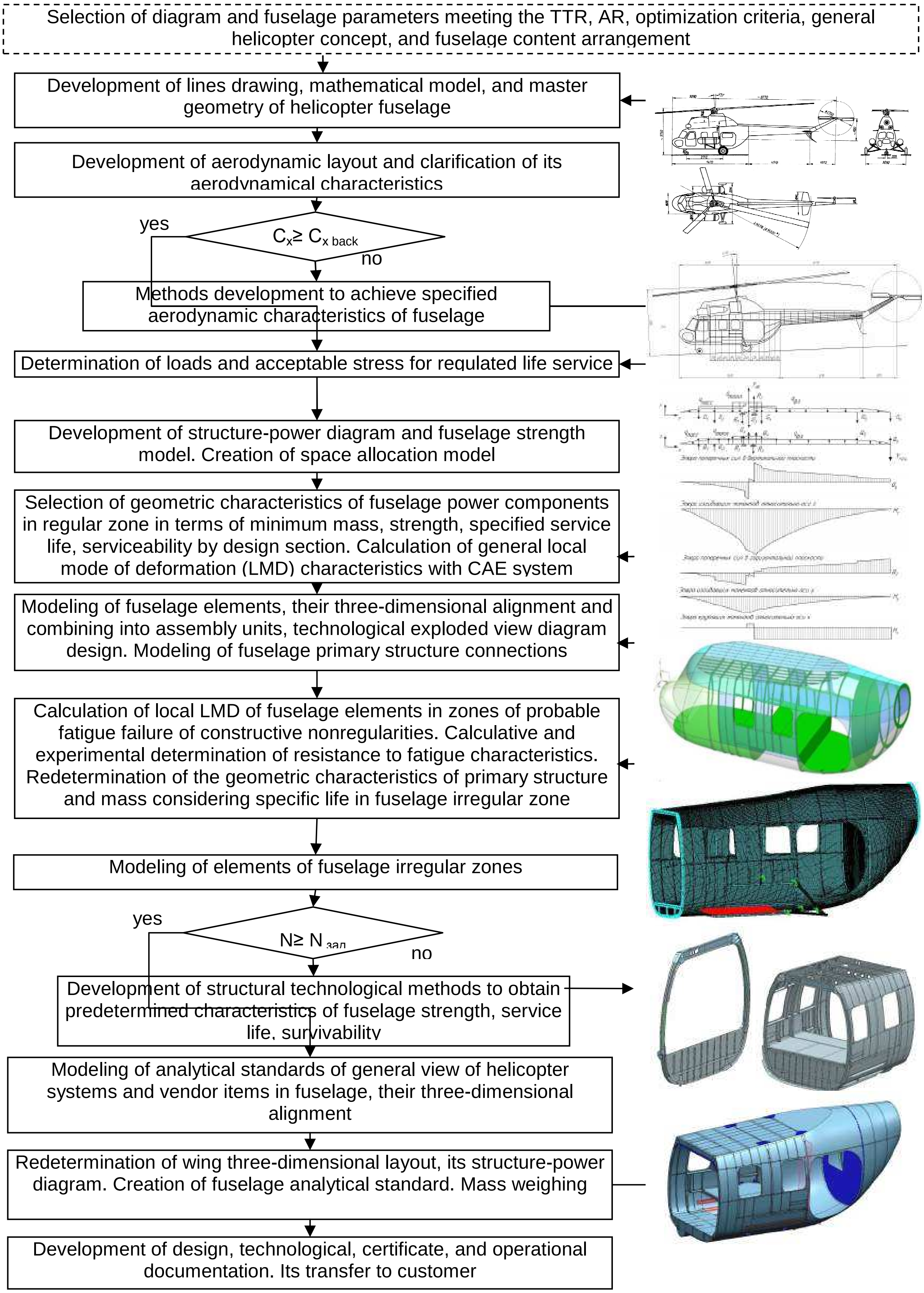


Fig. 5.4. Integrated Fuselage Design with Integrated Computer Systems

Mathematical models and lines drawings are developed for every helicopter assembly simultaneously with the trial and binding of all primary structural members and theoretical contours.

Lines drawings include:

– data about the assembly theoretical surface;
– overall and main assembly dimensions;
– assembly structure-power diagram;
– setting of the airfoil reference section (for the ruled surfaces) necessary for further surface development;
– parameters of reference sections and generators support lines;
– tables with nodes;
– binding of the assembly axes frame to any other helicopter assembly.

Main geometric parameters for a helicopter are selected at the stage of sketch design (including helicopter diagram and main geometric parameters of the helicopter and its components), with the helicopter general view drawing taken as basic. After this, mathematic model is developed (Fig. 5.5).

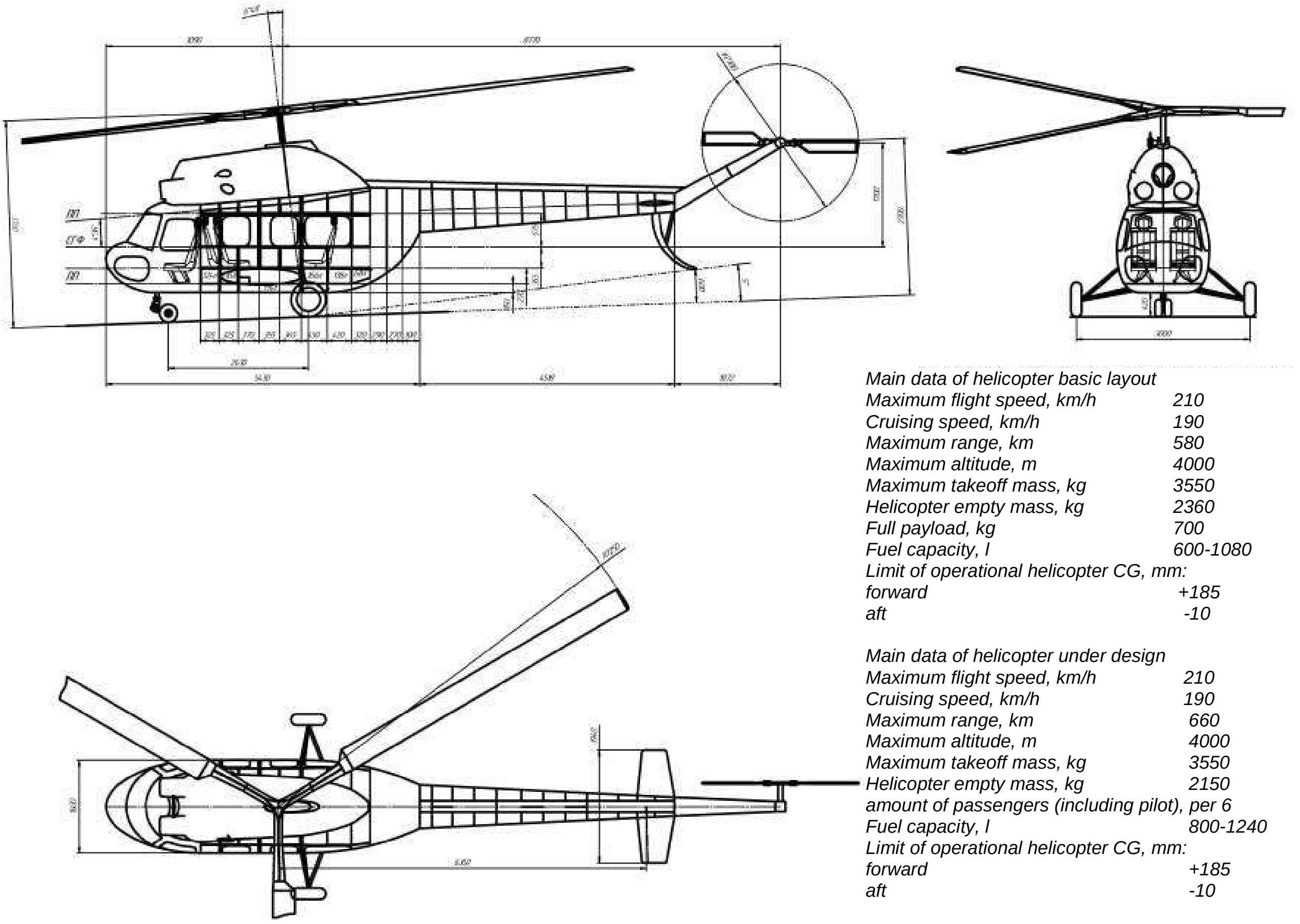


Fig. 5.5. Fragment of Helicopter General View Drawing

Lines drawings are necessary to build the assembly mathematical model in any other graphic package with two main principles of lines drawing for surface forming taken into account. These principles conform to the mathematical model building principles:

1. All arcs are outlined by second-order curves.
2. Surface forms due to the kinematical move of the shaping line along the guide line.

When using CAD/CAM/CAE SIEMENS NX system, fuselage model is developed in the following way.

The frame of support curve lines is created, describing theoretical fuselage lines on the basis of the general view drawing with a range of cuts and sections.

Support lines of the generator parameters are set by the second-order curve method. This method is more widely used for specifying flat lines in aircraft design for the reason that they possess curvature of the same sign, i.e. they don’t have inflexion points.

Mathematical apparatus of SIEMENS NX system allows building up second-order curves set by the engineer’s method:

– in five points;
– in four points and by a tangent line direction in a terminal point;
– in three points and by two tangent lines directions in terminal points;
– by the method of engineer’s triangle;
– in two points and by two tangent lines directions in terminal points and by a discriminant;
– by the method of setting equation coefficients for second-order curves.

The curves are built by choosing **Insert>Curve>*General conic.***

The line of maximum thickness is a result of two projections crossing. It is built by choosing **Insert>Curve from curves>*Combined Curve Projection***.

To create fuselage master-geometry, you use the module of surface building. Helicopter surface can be built by curves through:

– building up with a set of curves in tandem;
– building up in two sets of curves ordered in orthogonally related directions, i.e. surface is built with the net of curves;
– building up by sections set with the help of conic sections.

The third method is most widely used because it allows creating surfaces of necessary curvature and helps to get developed views (with certain departures).

SIEMENS NX system suggests twenty variants to set a surface by sections. The most widely used way is ***section, ends, apexes, discriminant*** (**Section Ends-Apex-Rho**). This method involves building up section surfaces with the help of initial and end curves and discriminant values. Apex curves set slopes at the start and in the end. Fig. 5.8 shows the example of this method.

Surface building is performed as: **Insert>Mesh Surface>*Sections*** (Fig. 5.6).

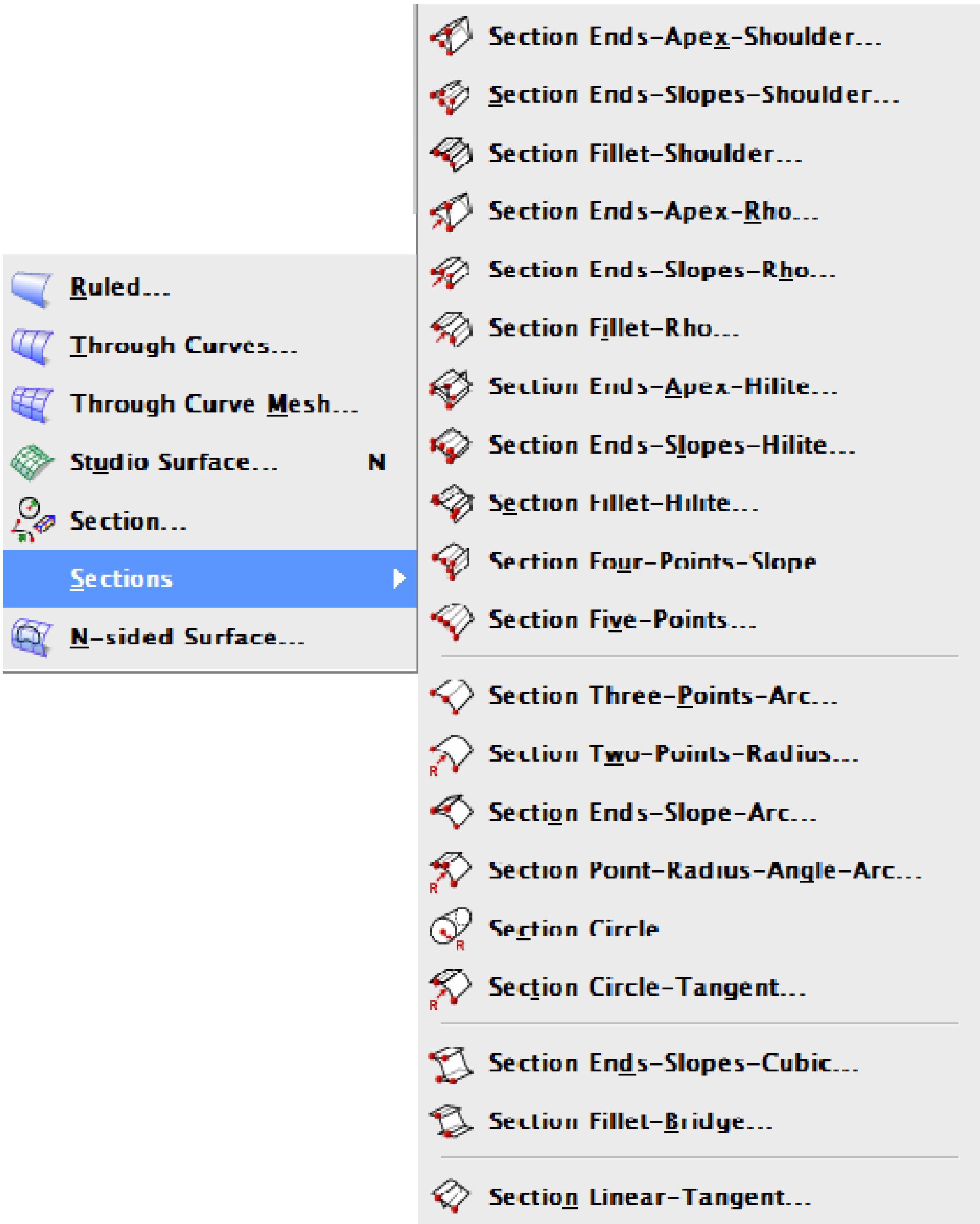


Fig. 5.6. Ways to Set Surfaces by Sections

As the fuselage can be constructively divided in three parts (nose, middle, and tail), it is a composite surface (Fig. 5.7). The surface of every segment is formed by the continuous displacement of the generator's arc according to the directing lines (support lines of the generator parameters).

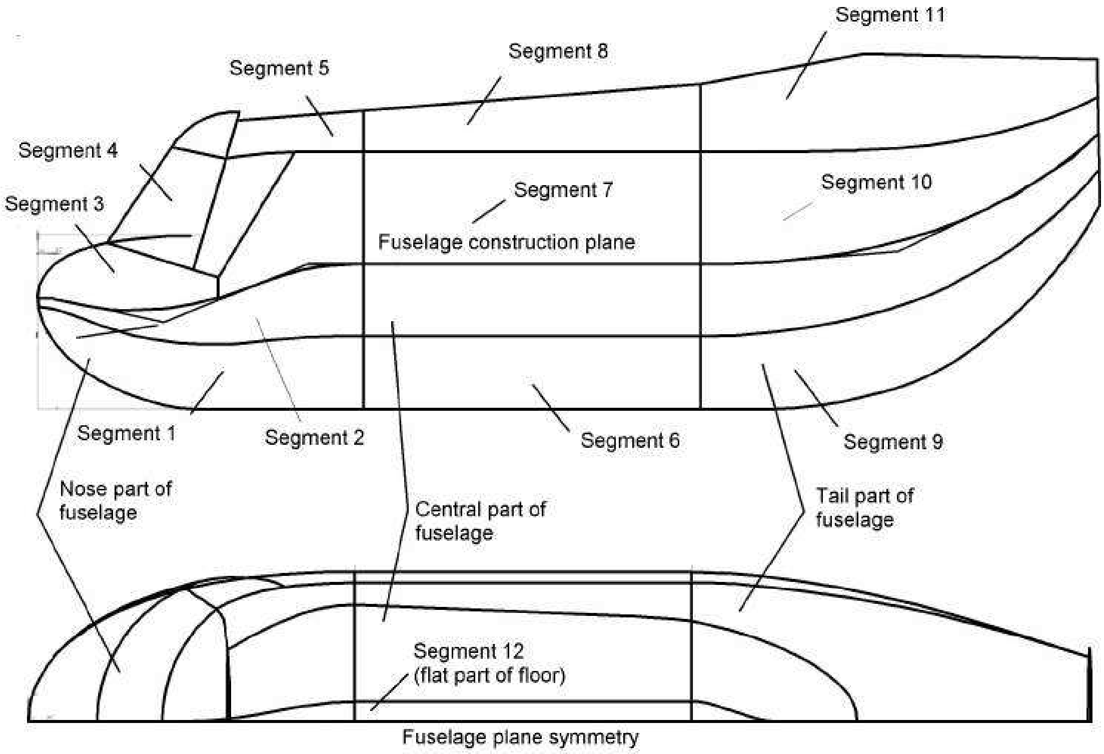


Fig. 5.7. Fragment of Lines Drawing of Helicopter Fuselage

To create a surface segment, you set start guide line (**Start Guide**), end guide line (**End Guide**), intersection line of tangent lines (**Apex Curve**), axial line (**Spine Curve**), and discriminant and type of its changing (Fig. 5.8). All segments of the fuselage surface are built by analogy.

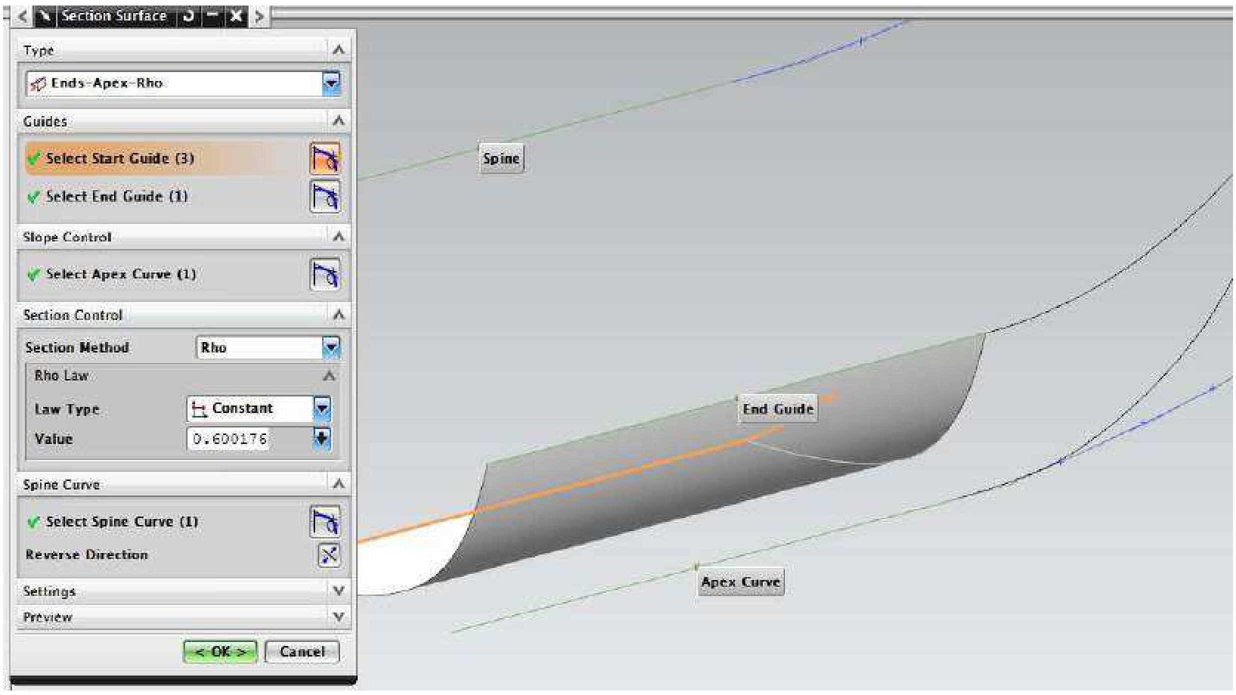


Fig. 5.8. Example of Segment 6 Building up

When using surface building up and modification commands, keep in mind that surfaces can have different matching (G0-G3):

G0 – two surfaces (body's planes) have a common border, i.e. connect to each other;

G1 – two surfaces have a common border, with tangent condition in each point of its border is fulfilled for both surfaces;

G2 – two surfaces have common border, with continuity condition by curvature each point of its border is fulfilled for both surfaces;

G3 – two surfaces have common border, with the curvature incrementing equally for each point of border in both surfaces.

Fig. 5.9 shows the resulting fuselage master geometry.

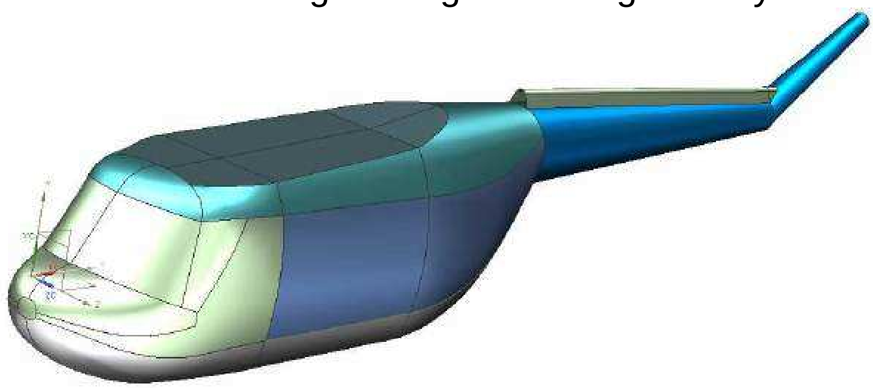


Fig. 5.9. Fuselage Master Geometry

After master-geometry is built up, helicopter designers proceed with fuselage structure-power diagram (SPD) and models of space allocation.

Fuselage SPD determines load accommodation by primary structures. Design procedures involve such primary structural members of the fuselage as:

– longitudinal structural members;
– transverse structural members;
– layout of longitudinal and transverse structural members;
– power units coupling.

SPD development covers parameter selection for the fuselage structure components including their geometric characteristics and materials used.

Ultimate loads affecting the fuselage are determined in accordance with AR depending on the helicopter takeoff mass.

Flight loads should be calculated for:

– all ranges of rated flight altitudes;

– all required altitude-to-mass combinations within operating restrictions prescribed in AFM.

In a general case of a straight flight, helicopter is affected by the following external forces (Fig. 5.10): lift force of the main rotor; helicopter drag force **X**; helicopter weight force $\mathbf{G_0}$; fuselage lift force $\mathbf{Y_{fus}}$; aerodynamic force of the stabilizer $\mathbf{Y_{st}}$; balancing aerodynamic force $\mathbf{Y_{bal}}$; manoeuvre force $\mathbf{Y_{m.t.r.}}$ and torque $\mathbf{M_{yfus}}$ caused by fuselage rotation; torque of the main rotor $\mathbf{M_{m.r}}$; and torque of the tail rotor $\mathbf{M_{t.r.}}$ Manoeuvre force $\mathbf{Y_{m.t.r.}}$ arises on the anti-torque rotor and causes rotational mode of the helicopter due to the changing tail rotor angle of attack at the start of manoeuvre.

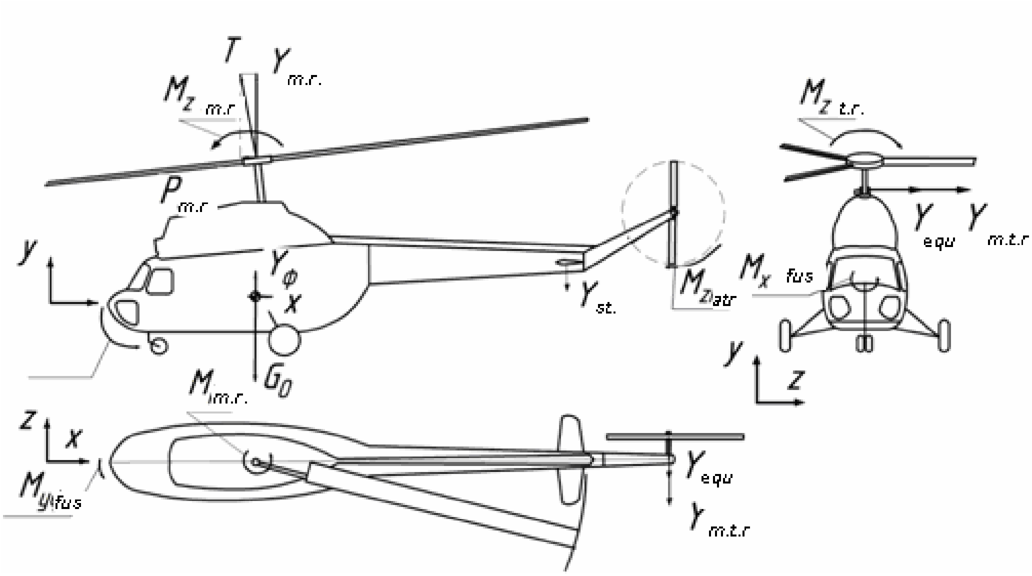


Fig. 5.10. Diagram of Forces affecting the Helicopter in Horizontal Flight

Main loads affecting the helicopter fuselage are the loads of main gearbox, tail boom with anti-torque rotor and transmission, landing gear and engines mounted on the fuselage members. Besides, fuselage can contain fuel tanks, crew, passengers, equipment, and cargo. There can also be local aerodynamic loads that may reach a dynamic pressure value of ram airflow in separate parts (cockpit canopy, nose, etc.).

To define the ultimate loads effecting fuselage, designers study operational loads with respect to the factor of safety **f** regulated by AR.

When calculation diagram of the helicopter fuselage in flight is developed, two strong frames are admitted to serve as supports (Fig. 5.11). In Fig. 5.11,

$\mathbf{q_{fus}}$ is the fuselage empty weight computational load per unit length;

$\mathbf{q_{pas}}$ is the passengers computational load per unit length;

$\mathbf{q_{fuel}}$ is the fuel computational load per unit length;

$\mathbf{G_1}$, $\mathbf{G_4}$ are nose and main LG concentrated load in a vertical plane;

$\mathbf{G_2}$ is the engines concentrated load in a vertical plane;

$\mathbf{G_3}$, $\mathbf{G_5}$, and $\mathbf{G_6}$ are concentrated loads of the main, intermediate, and tail gearboxes correspondingly;

$\mathbf{Q_1}$, $\mathbf{Q_4}$ are lateral forces arising from the LG concentrated masses in a horizontal plane;

$\mathbf{Q_2}$ is the lateral force arising from the engine concentrated mass in a horizontal plane;

$\mathbf{Q_3}$, $\mathbf{Q_5}$, and $\mathbf{Q_6}$ are lateral forces arising from the main, intermediate, and tail gearboxes in a horizontal plane;

$\mathbf{Y_{pr}}$ is the total of the balancing and manoeuvre forces.

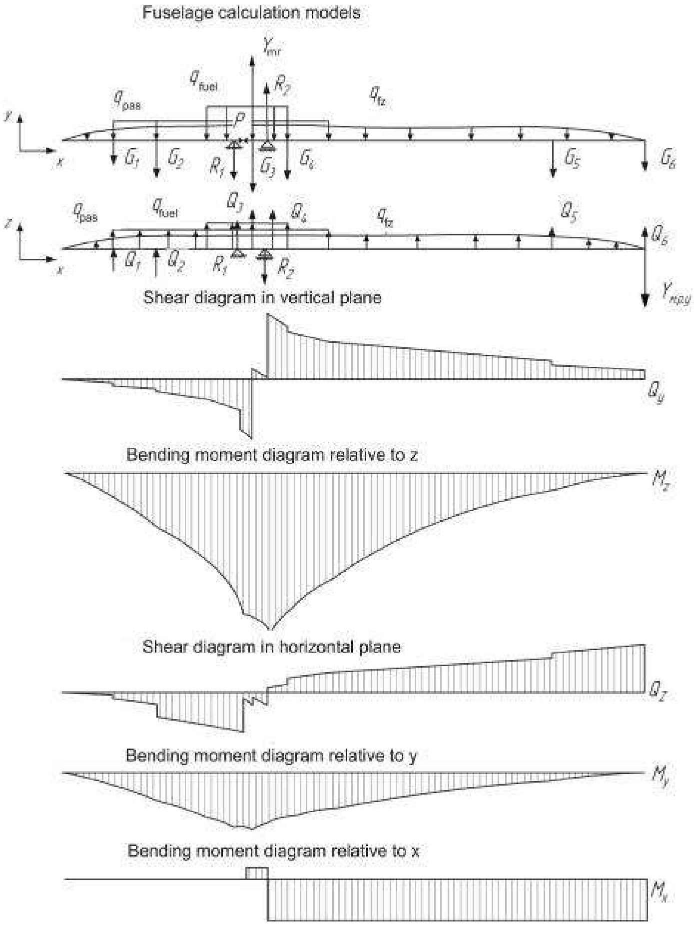


Fig. 5.11. Fuselage Loading and Fastening Diagrams. Shear and Moment Diagrams

After the calculation diagram is developed and ultimate load for all concerned loading cases is determined, an analytical designing calculation of fuselage is carried out. This calculation allows choosing helicopter structure-power diagram, determining geometric parameters of the fuselage structural members, specifying space allocation model (Figs 5.12, 5.13).

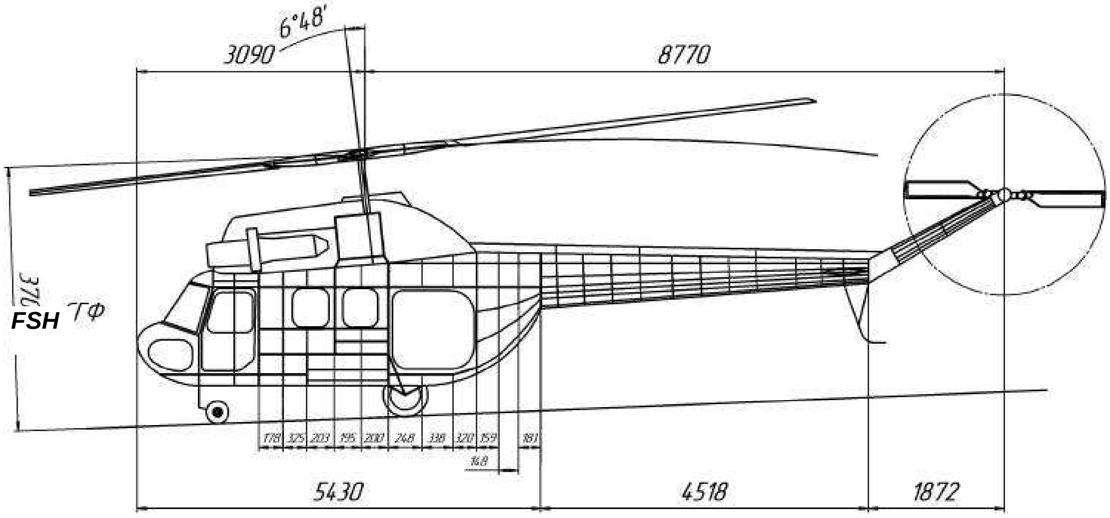


Fig. 5.12. Helicopter Structure-Power Diagram

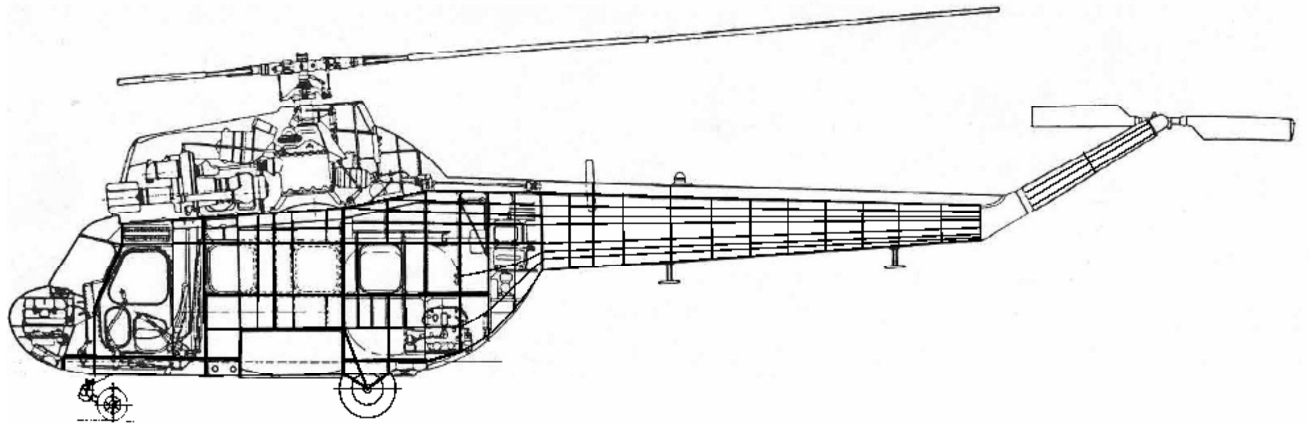


Fig. 5.13. Fuselage Layout Diagram

At this stage, helicopter designers choose a fuselage zone to study, develop its calculation diagram, and specify affecting loads. In the context of geometry and load application, the helicopter fuselage is an asymmetric construction. That is why you must consider the whole section of helicopter fuselage in the calculation.

Fuselage section calculation diagram is shown in Fig. 5.14.

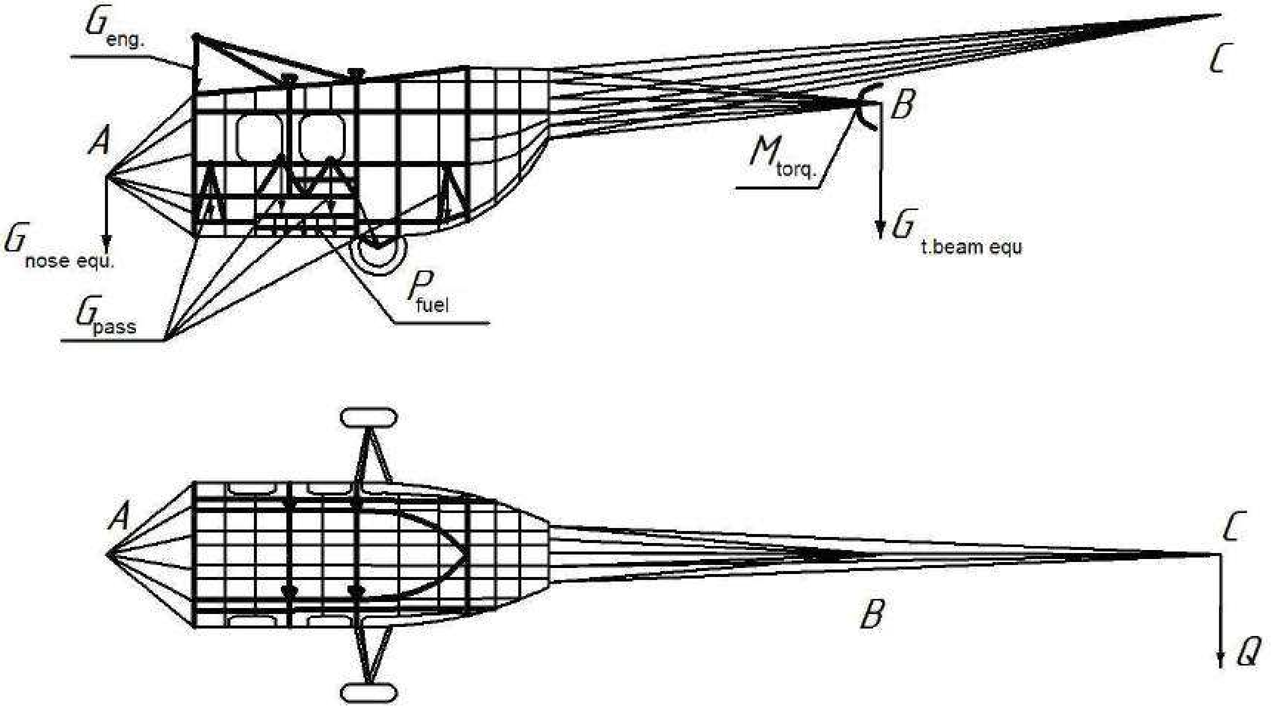


Fig. 5.14. Calculation Diagram of Fuselage Section

The model is fastened with hinge attachment at the main gearbox attachment fittings. In points A and B (Fig. 5.14), lateral forces $\mathbf{G}_{\mathbf{nose\ equ}}$, $\mathbf{G}_{\mathbf{tail\ equ}}$, and $\mathbf{M}_{\mathbf{tor}}$ are applied. These forces replace the weight effect of nose part and tail boom as well as torque. Combination of balancing and manoeuvre forces of the tail rotor is applied in point C.

Figs 5.15, 5.16 show finite-element model of helicopter fuselage section.

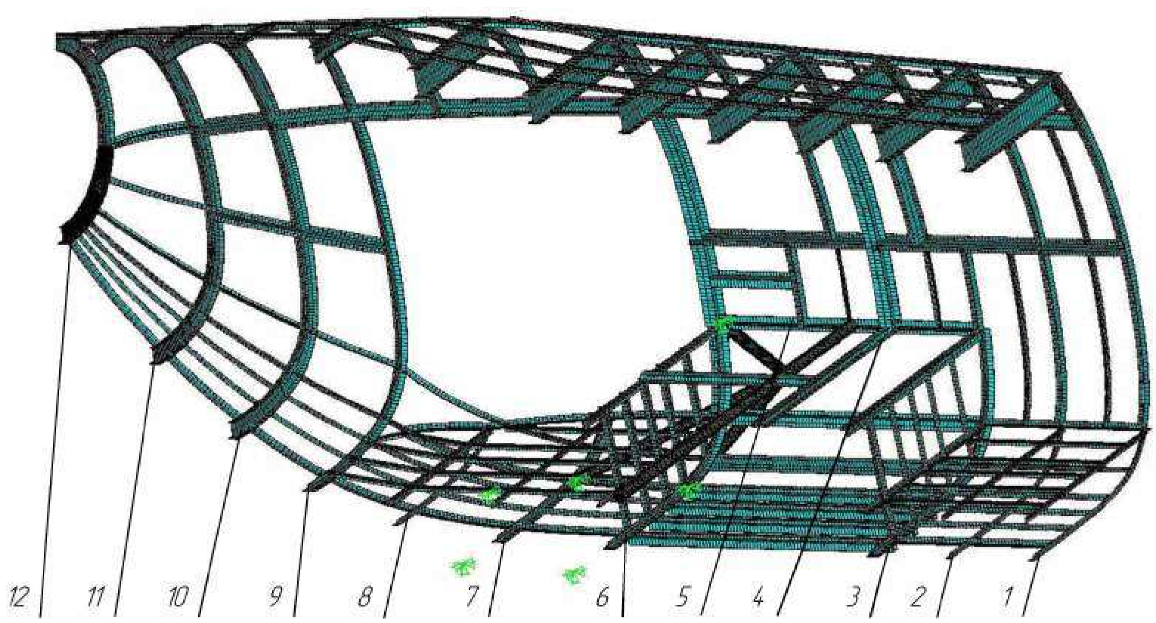


Fig. 5.15. Beam Elements of Finite-element Model of Fuselage Section

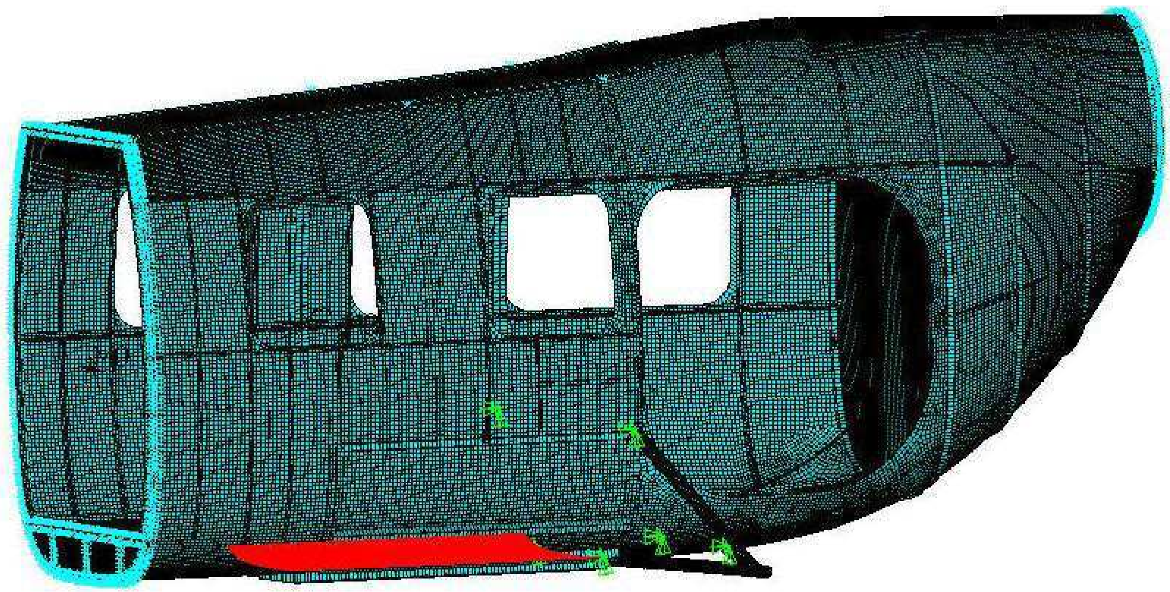


Fig. 5.16. Finite Element Model of Helicopter Fuselage Section

Equivalent stresses von Mises on the nodes of finite element model of helicopter fuselage section is shown in Fig. 5.17.

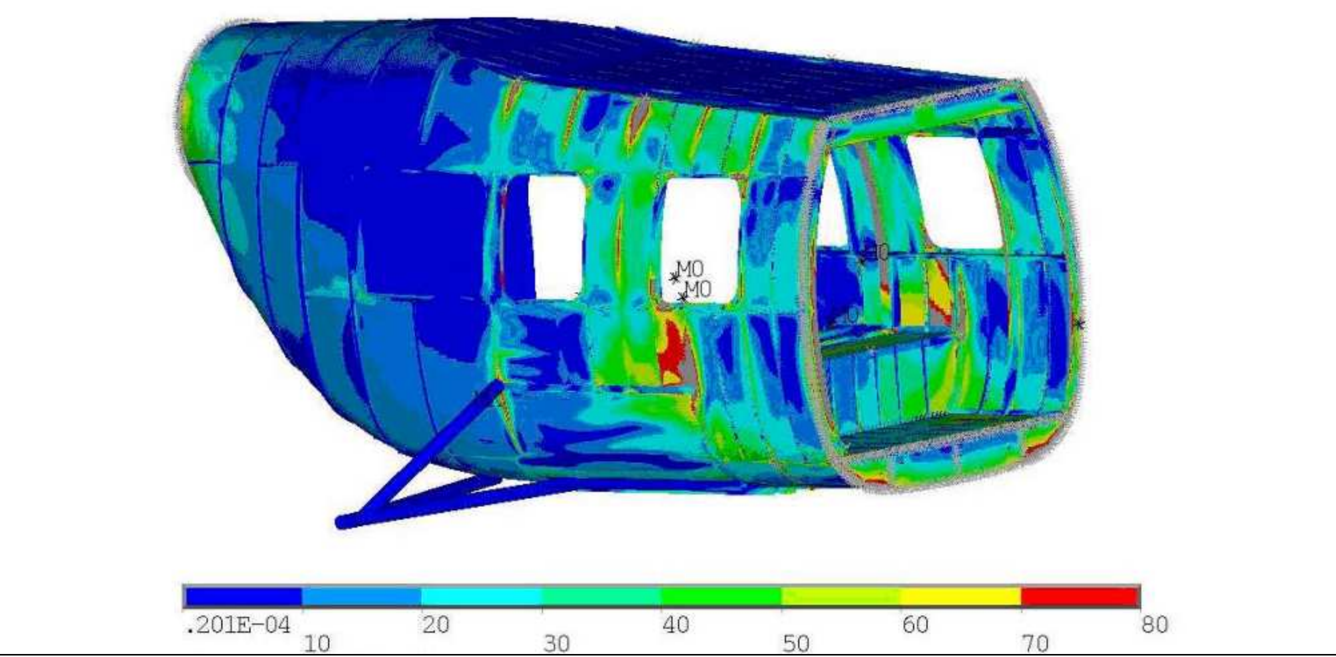


Fig. 5.17. Equivalent Stresses von Mises in Helicopter Fuselage Section

These calculations determine LMD of the helicopter fuselage section for selected loading cases. Obtained LMD parameters for the helicopter fuselage section variant become the initial data for further analysis of the static strength and fatigue life of different structure variants of the designed aircraft [32].

Next design stage covers the development of the fuselage space allocation model (Fig. 5.18). The model is developed on the basis of the structural-technological exploded view, panelling, primary structural members as well as fuselage design tree structure. It includes planes of structural members, their axes, allocation of equipment and appliances, system routes layout, tracing, position of movable members specifying the layout, and tank space.

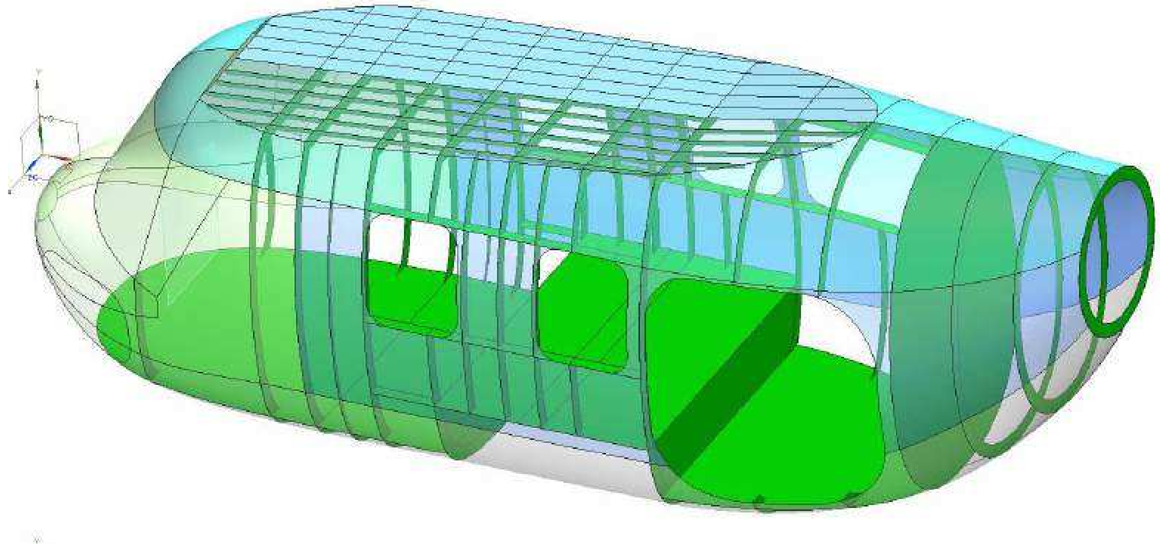


Fig. 5.18. Helicopter Fuselage Space Allocation Models

In the space allocation model, longitudinal and transverse structural members become surfaces. The model is developed based on the intersection curves of master-geometry with planes of the structural members forming a set of objects. To create the intersection curve, use **Insert>Curve from bodies>*Intersection curve.***

The next design stage involves both the selection of geometric and structural technological parameters of the fuselage structural members and their modelling (development of analytical standards).

Analytical standards are modelled together with the fuselage master geometry on the basis of space allocation model.

Typical structural members or produced frames based on two-dimensional contours as well as their combination are used to develop models.

Sketches provide a basis for all frames obtained by rotation or extension along the path. To develop a frame with the help of a sketch, you use one of the commands for sketch extrude in the intended direction or for the rotation about an axis. The most frequently used commands are **Extrude** and **Revolve** from the **Feature** toolbar or **Insert>Feature** from menu.

Obtained analytical standards are combined into assemblies. Fig. 5.19 shows analytical standards of the LG attachment fitting and an assembly model of a strong frame.

Fig. 5.19. Analytical Standards of Landing Gear Attachment Fitting and Assembly Model of Strong Frame

All elements are linked together by an associative relation.

Analytical standards of structural members are obtained from the parametric condition, i.e. mathematical models of objects with parameters are developed which, if changed, can cause configuration changes in parts or their reciprocal displacements, etc.

Today modelling is often performed in context, i.e. considering the surrounding of a part or joint under development. Many complex items with tight layout needs must be modelled in context, with developing 3D models of the joints based on geometry and parameters of already-existing parts and assemblies building up intermodelling links and controlling changes.

When modelling in context or using references to the geometry of other parts out of context, helicopter designers must correctly determine parameters and objects geometry that can be used for referencing. NX system provides the **WAVE**, a powerful and flexible device orientated to intermodel links. When building geometric and parametric links, bind to predetermined elements and variables, i.e. to objects that are stable, easily controlled, and don't deteriorate.

As the model is presented to be an assembly, it is created as **File>New>*Assembly*** with its name saved (use only Latin letters and numbers for names and locations). Then you can add the components to any assembly level, position them, and set specific assembly links.

To add assembly components, you use **Add component…** from the **Assemblies** toolbar or **Assemblies> Components>*Add component*** from main menu; choose a downloaded component or existing model in the command dialogue; and select the positioning mode. In the **Placement** tab, you start the absolute coordinate system **Absolute original** to place the component at zero of the absolute coordinate system.

Fig. 5.20 shows the resulting assembly model of analytical elements of the fuselage section primary structural members.

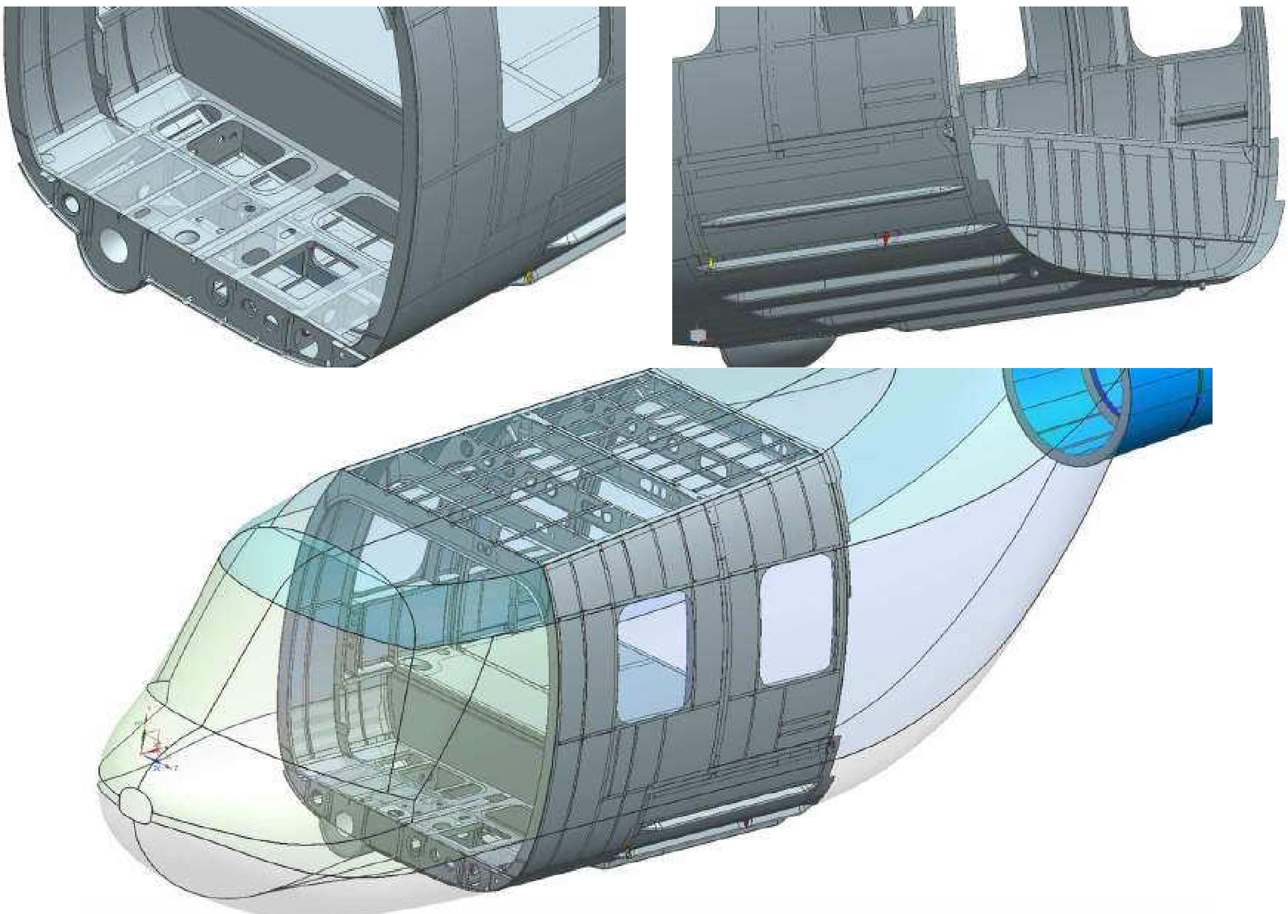


Fig. 5.20. Assembly Model of Analytical Elements of the Fuselage Section Primary Structural Members

At last, you model analytical standards of the fuselage systems components including electric power supply system, control system, fuel system, etc. Three-dimensional alignment of these elements is necessary to determine space allocation, re-determine SPD, complete the helicopter fuselage modelling, calculate masses, specify design errors, and forecast other technological characteristics. When fulfilled, fuselage standards allow developing structural-technological exploded view of the fuselage.

At this stage, helicopter designers produce design, technological, certification, and operational documentation concerning the fuselage model, and get control programs for the NC machine. Then they consider economic factors for the fuselage development and operation, its reparability, write recycling documentation, etc.

On the basis of the fuselage systems received components standards, analytical standards of the procuring assembled tools are developed, and helicopter designers proceed to plan operating positions and calculate productive capacity required by manufacturers.

Appendix 1

Table A.1

CORRELATION BETWEEN MKGSS SYSTEM OF UNITS AND OFF-SYSTEM UNITS AND SI UNITS

Value	Unit: Name	Unit: Designation	Conversion into SI units
Mass	kilogram-force-squared second per metre	$kgf{\cdot}s^2/m$	9.80665 kg
Force	kilogram-force	kgf	9.80665 N
Moment	kilogram-force- metre	$kgf{\cdot}m$	$9.80665\ N{\cdot}m$
Moment of Inertia	kilogram-force- metre - squared second	$kgf{\cdot}m{\cdot}s^2$	$9.80665\ kg{\cdot}m^2$
Pressure	kilogram-force per square metre	kgf/m^2	9.80665 Pa
	technical atmosphere:	atm	98066.5 Pa
	kilogram-force per square centimetre	kgf/m^2	98066.5 Pa
	standard atmosphere:	atm	101325 Pa
	millimetre of water column	mm H_2O	9.80665 Pa
	millimetre of mercury column	mm Hg	132.322 Pa
	bar	bar	$1{\cdot}10^5$ Pa
Work and energy	kilogram-force- metre	$kgf{\cdot}m$	9.80665 J
	horse power-hour	hp-hour	$2.64780{\cdot}10^6$ J
	kilowatt-hour	$kW{\cdot}h$	$36{\cdot}10^6$ J
Capacity	kilogram-force- metre per second	$kgf{\cdot}m/s$	9.80665 W
	horse power	hp	735.499 W
Dynamical Viscosity	kilogram-force-second per square metre	$kgf{\cdot}s/m^2$	$9.80665\ Pa{\cdot}s$
Flat Angle	second	″	$4.848177{\cdot}10^{-6}$ rad
	minute	′	$2.908882{\cdot}10^{-4}$ rad
	degree	°	0.01745329 rad
	square corner	L	1.570796 rad
	r (full angle)	r	6.283185 rad
Space Angle	square degree	$\alpha^{\circ 2}$	$3.0462{\cdot}10^{-4}$ avg
Linear Speed	kilometre per hour	km/h	0.277778 m/s
Angular Velocity	degree per second	°/s	0.01745329 rad/s
	turn per second	t/s	6.283185 rad/s
	turn per minute	t/min	0.1047197 rad/s
Stress (normal, shear)	kilogram-force per square millimetre	kgf/mm^2	$9.80665{\cdot}10^6$ Pa
	kilogram-force per square centimetre	kgf/cm^2	$9.80665{\cdot}10^4$ Pa
Heat	calorie (international)	cal	4.1868 J
Specific Heat Capacity	kilocalorie per kilogram-degree Celsius	kcal /(kg·°C)	$4.1868{\cdot}10^3$ J/(kg·K)
Resistivity	Ohm-square millimetre per metre	Ohm mm^2/m	$10^{-6}\ Ohm{\cdot}m$

Appendix 2

STANDARD ATMOSPHERE AND ITS PARAMETERS ACCORDING TO GOST 4401-81

Geometric Altitude	Values in Function of Geometric Altitude							
	Temperature		Pressure		Density	Relative Density Factor	Acceleration of Gravity	Sound Velocity
m	***T***. K	***t***. ^{o}C	Pa	mm Hg	$\boldsymbol{\rho}$. kg/m^3	$\boldsymbol{\Delta = \rho_H/\rho_0}$	***g***. m/s^2	***a***. m/s
0	283.15	15.0	101325	760	1.225	1.0	9.8066	340.294
100	287.5	14.35	100129	751.03	1.213	0.990204	9.8063	339.910
200	286.85	13.7	98945.4	742.15	1.2016	0.980898	9.806	339.526
300	286.2	13.05	97777.27	733.36	1.19011	0.971518	9.8057	339.141
400	285.55	12.4	96614.0	724.45	1.17865	0.96216	9.8054	338.755
500	284.9	11.75	95461.3	716.019	1.16727	0.95287	9.8051	338.37
600	284.25	11.1	94322.3	707.476	1.15558	0.943657	9.8048	337.983
700	283.6	10.451	93194.4	699.016	1.14478	0.934514	9.8045	337.597
800	282.95	9.801	92077.5	690.683	1.13366	0.925436	9.8042	337.210
900	282.301	9.151	90971.5	682.312	1.12261	0.916416	9.8039	336.822
1000	281.651	8.501	89876.3	674.128	1.11166	0.907477	9.8036	336.435
1100	281.001	7.851	88791.8	665.993	1.10079	0.898604	9.8033	336.046
1200	280.351	7.201	87718.0	657.939	1.0899	0.889714	9.8029	335.658
1300	279.702	6.552	86654.8	649.914	1.07928	0.881045	9.8026	335.463
1400	279.052	5.902	85602.0	642.068	1.06865	0.872367	9.8023	334.879
1500	278.401	5.252	84597.0	634.25	1.0581	0.863755	9.802	334.489
1600	277.753	4.603	83527.7	626.509	1.04764	0.855216	9.8017	334.098
1700	277.103	3.953	82505.9	618.845	1.03725	0.846735	9.8014	333.707

Appendix 2 (Continued)

Geometric Altitude	Values in Function of Geometric Altitude							
	Temperature		Pressure		Density	Relative Density Factor	Acceleration of Gravity	Sound Velocity
m	***T***. K	***t***. °C	Pa	mm Hg	***ρ***. kg/m³	$\boldsymbol{\Delta = \rho_H / \rho_0}$	***g***. m/s²	***a***. m/s
1800	276.453	3.303	81494.3	611.258	1.02694	0.838318	9.8011	333.316
1900	275.804	2.654	80492.9	603.746	1.01671	0.829967	9.8008	332.924
2000	275.154	2.004	79501.4	596.31	1.00655	0.821673	9.8005	332.532
2100	274.505	1.355	78519.9	588.948	0.996479	0.813452	9.8002	332.139
2200	273.855	0.705	77548.3	581.66	0.98648	0.805289	9.7999	331.746
2300	273.205	0.055	76584.4	574.445	0.976563	0.797194	9.7996	331.352
2400	272.556	-0.594	75634.2	567.304	0.96672	0.789159	9.7992	330.958
2500	271.906	-1.244	74691.7	560.234	0.956954	0.781187	9.7989	330.563
2600	271.257	-1.893	73758.8	553.236	0.94726	0.77327	9.7986	330.168
2700	270.607	-2.543	72835.3	546.31	0.937649	0.765428	9.7983	329.773
2800	269.958	-3.192	71921.3	539.454	0.92811	0.757641	9.798	329.377
2900	269.309	-3.841	71016.6	532.668	0.918645	0.749914	9.7977	328.98
3000	268.659	-4.491	70121.2	525.952	0.909254	0.742248	9.7974	328.584
3100	268.01	-5.14	69234.9	519.304	0.89938	0.734188	9.7971	328.186
3200	267.36	-5.79	68357.8	512.725	0.89069	0.727094	9.7968	327.788
3300	266.711	-6.439	67489.7	506.214	0.881524	0.719611	9.7965	327.39
3400	266.062	-7.088	66630.6	499.77	0.872427	0.712185	9.7962	326.991
3500	265.413	-7.737	65780.4	493.393	0.863402	0.704818	9.7959	326.592
3600	264.763	-8.387	64939.0	487.083	0.854449	0.697509	9.7956	326.192

Geometric Altitude	Values in Function of Geometric Altitude							
	Temperature		Pressure		Density	Relative Density Factor	Acceleration of Gravity	Sound Velocity
m	***T***. K	***t***. °C	Pa	mm Hg	ρ. kg/m^3	$\Delta = \rho_H / \rho_0$	***g***. m/s^2	***a***. m/s
3700	264.114	-9.036	64106.4	480.837	0.84567	0.690343	9.7952	325.792
3800	263.465	-9.685	63282.5	474.658	0.836756	0.683066	9.7949	325.392
3900	262.816	-10.334	62467.2	468.542	0.828016	0.615931	9.7946	324.990
4000	262.166	-10.984	61660.4	462.491	0.819347	0.668855	9.7943	324.589
4100	261.517	-11.633	60862.2	456.504	0.810747	0.661834	9.7940	324.187
4200	260.868	-12.282	60072.3	450.579	0.802216	0.65487	9.7937	323.784
4300	260.219	-12.931	59290.8	444.718	0.793755	0.64796	9.7934	323.381
4400	259.570	-13.58	58517.6	438.918	0.785363	0.64111	9.7931	322.977
4500	258.921	-14.229	57752.6	433.18	0.777038	0.634316	9.7928	322.573
4600	258.272	-14.878	56995.7	427.503	0.768782	0.627577	9.7925	322.169
4700	257.623	-15.527	56246.9	421.886	0.760593	0.62089	9.7922	321.764
4800	256.974	-16.176	55506.1	416.33	0.752472	0.614263	9.7919	321.358
4900	256.325	-16.825	54773.2	410.833	0.744417	0.607687	9.7915	320.952
5000	255.676	-17.474	54048.3	405.395	0.736429	0.601166	9.7912	320.545
5100	255.027	-18.123	53331.1	400.016	0.728506	0.594699	9.7909	320.138
5200	254.378	-18.772	52621.7	394.695	0.720649	0.588285	9.7906	319.731
5300	253.729	-19.421	51920.0	389.432	0.712858	0.581925	9.7903	319.232
5400	253.080	-20.070	51225.9	384.225	0.705131	0.57561	9.7900	318.914
5500	252.431	-20.719	50539.8	379.076	0.697469	0.56936	9.7897	318.505

Appendix 3

SOME STATISTIC DATA ON BLADE SECTIONS OF MAIN ROTORS

Table A.3.1

Aerodynamic Performance of Blade Section NACA 23012

Characteristics	M	$\alpha°$								
		-2	1.0	3.5	7	9	11	12.5	14.5	15
C_y	0.3	-0.085	0.205	0.46	0.81	1.035	1.21	1.365	1.525	1.525
	0.4	-0.1	0.2	0.445	0.8	1.01	1.2	1.33	1.42	1.42
	0.5	-0.085	0.225	0.485	0.85	1.0	1.185	1.24	1.25	1.245
	0.6	-0.085	0.225	0.485	0.843	0.94	1.0	1.03	1.048	1.05
	0.7	-0.085	0.245	0.505	0.715	0.785	0.837	0.87	0.91	0.915
	0.8	-0.065	0.285	0.43	0.556	0.625	0.675	0.715	0.76	0.77
	0.85	-0.055	0.185	0.3	0.435	0.49	—	—	—	—
	0.9	-0.075	0.09	0.2	—	—	—	—	—	—
C_{xp}	0.3	0.008	0.008	0.01	0.015	0.018	0.022	0.029	0.045	0.05
	0.4	0.008	0.008	0.01	0.015	0.023	0.0355	0.043	0.07	0.074
	0.5	0.008	0.008	0.01	0.019	0.031	0.0575	0.0835	0.121	0.13
	0.6	0.008	0.009	0.0135	0.0365	0.0765	0.128	0.167	0.218	0.23
	0.7	0.009	0.013	0.0275	0.09	0.138	0.181	0.213	0.254	0.252
	0.8	0.0125	0.03	0.067	0.13	0.177	0.121	0.253	0.294	0.304
	0.85	0.028	0.049	0.08	0.145	0.185	—	—	—	—
	0.9	0.069	0.08	0.1075	—	—	—	—	—	—

Table A.3.2

Aerodynamic Performance of Blade Section NACA 0012

Characteristics	M	α°								
		-2	1.0	3.5	7	9	11	12.5	14.5	15
C_y	0.3	-0.185	0.085	0.32	0.645	0.835	1.02	1.155	1.34	1.39
	0.4	-0.18	0.095	0.335	0.665	0.855	1.035	1.175	1.25	1.25
	0.5	-0.215	0.10	0.355	0.71	0.915	1.08	1.1	1.1	1.1
	0.6	-0.215	0.11	0.375	0.75	0.91	0.94	0.95	0.96	0.965
	0.7	-0.235	0.11	0.395	0.735	0.81	0.84	0.86	0.863	0.865
	0.8	-0.245	0.135	0.4	0.57	0.65	0.72	0.765	0.765	0.75
	0.85	-0.19	0.095	0.29	0.5	0.61	0.71	—	—	—
	0.9	-0.08	0.02	0.14	0.4	0.56	0.7	—	—	—
C_{xp}	0.3	0.0095	0.007	0.009	0.0125	0.0165	0.021	0.024	0.029	0.034
	0.4	0.0095	0.007	0.009	0.0125	0.0165	0.021	0.0245	0.061	0.08
	0.5	0.0095	0.007	0.009	0.013	0.0165	0.031	0.051	0.106	0.126
	0.6	0.01	0.007	0.0105	0.021	0.039	0.074	0.1095	0.171	0.186
	0.7	0.01	0.0085	0.0185	0.061	0.0955	0.135	0.1675	0.211	0.221
	0.8	0.0245	0.016	0.046	0.095	0.131	0.1675	0.195	0.2285	0.236
	0.85	0.0415	0.036	0.061	0.1065	0.141	0.18	—	—	—
	0.9	0.069	0.069	0.0795	0.118	0.149	0.187	—	—	—

Table A.3.3

Aerodynamic Performance of High-Speed Airfoil ($\bar{c}$ = 9%. is recommended for use in blade tip) (Fig. A.3.2)

Characteristics	M	α°								
		-2	1.0	3.5	7	9	11	12.5	14.5	15
C_y	0.3	-0.065	0.235	0.485	0.835	1.035	1.18	1.165	1.115	1.1
	0.4	-0.065	0.230	0.230	0.485	0.835	1.035	1.1	1.06	1.05
	0.5	-0.065	0.245	0.5	0.86	1.015	1.015	1.0	0.99	0.99
	0.6	-0.065	0.260	0.53	0.9	0.98	0.96	0.065	0.96	0.96
	0.7	-0.07	0.30	0.6	0.96	0.96	0.935	0.935	0.95	0.95
	0.8	-0.07	0.36	0.63	0.81	0.87	0.87	0.89	0.935	0.945
	0.85	-0.12	0.325	0.55	0.77	0.86	0.86	—	—	—
	0.9	-0.165	0.175	0.46	0.815	—	—	—	—	—
C_{xp}	0.3	0.008	0.007	0.009	0.011	0.012	0.0245	0.065	0.125	0.133
	0.4	0.008	0.007	0.009	0.011	0.012	0.055	0.0975	0.142	0.15
	0.5	0.008	0.007	0.0125	0.0125	0.046	0.093	0.13	0.1765	0.1885
	0.6	0.008	0.007	0.01	0.025	0.06	0.11	0.1475	0.195	0.205
	0.7	0.008	0.0075	0.015	0.061	0.1	0.143	0.175	0.195	0.221
	0.8	0.0125	0.012	0.037	0.092	0.128	0.165	0.194	0.2125	0.2415
	0.85	0.021	0.026	0.053	0.11	0.15	0.19	—	—	—
	0.9	0.044	0.04	0.069	0.131	—	—	—	—	—

Table A.3.4

Aerodynamic Performance of High-Speed Airfoil CBM

Characteristics	M	$\alpha°$									
		15	-2	1.0	3.5	7	9	11	12.5	14.5	15
C_y	0.3	-0.21	0.04	0.16	0.43	0.81	1.03	1.23	1.36	1.37	1.32
	0.4	-0.21	0.04	0.16	0.43	0.81	1.03	1.23	1.29	1.21	1.2
	0.5	-0.15	0.06	0.19	0.47	0.99	1.03	1.18	1.18	1.14	1.13
	0.6	-0.14	0.07	0.21	0.52	0.91	1.03	1.03	1.06	1.065	1.07
	0.7	-0.15	0.1	0.25	0.57	0.79	0.87	0.915	0.94	0.98	1.07
	0.8	-0.16	0.14	0.28	0.48	0.64	0.74	0.83	0.88	0.92	0.93
	0.85	-0.16	0.15	0.16	0.39	0.62	0.7	0.79	0.86	0.92	0.93
	0.9	-0.2	0.13	0.04	0.38	0.57	0.67	0.71	0.825	0.835	0.84
C_{xp}	0.3	0.01	0.01	0.01	0.01	0.01	0.02	0.025	0.03	0.04	0.05
	0.4	0.01	0.01	0.01	0.01	0.01	0.02	0.025	0.03	0.07	0.08
	0.5	0.01	0.01	0.01	0.01	0.01	0.03	0.04	0.07	0.115	0.13
	0.6	0.02	0.02	0.02	0.02	0.02	0.051	0.09	0.12	0.17	0.18
	0.7	0.02	0.02	0.02	0.02	0.065	0.095	0.132	0.165	0.21	0.22
	0.8	0.05	0.04	0.04	0.05	0.095	0.125	0.16	0.19	0.235	0.24
	0.85	0.06	0.05	0.05	0.07	0.112	0.142	0.175	0.205	0.255	0.26
	0.9	0.07	0.06	0.06	0.1	0.142	0.175	0.212	0.245	0.29	0.3

Table A.3.5

Geometrical Characteristics of Airfoil NACA 23012 (in % of chord)

x. %	0	1.25	2.5	5	7.5	10	15	20	25	30	40	50	60	70	80	90	95	100
$y_{в}$. %	—	2.67	3.61	4.91	5.8	6.43	7.19	7.5	7.6	7.55	7.14	6.41	5.47	4.36	3.08	1.68	0.92	0
$y_{н}$. %	—	-1.23	-1.71	-2.26	-2.61	-2.92	-3.5	-3.97	-4.28	-4.46	-4.48	-4.17	-3.67	-3.0	-2.16	-1.23	-0.7	0

Table A.3.6

Geometrical Characteristics of Airfoil NACA 0012 (in % of chord)

x. %	0	1.25	2.5	5	7.5	10	15	20	25	30	40	50	60	70	80	90	95	100
$y_{в}$. %	0	1.89	2.62	3.56	4.2	4.68	5.34	5.74	5.94	6.0	5.8	5.29	4.56	3.66	2.62	1.45	0.81	0
$y_{н}$. %	0	-1.89	-2.62	-3.56	-4.2	-4.68	-5.34	-5.74	-5.94	-6.0	-5.8	-5.29	-4.56	-3.66	-2.62	-1.45	-0.81	0

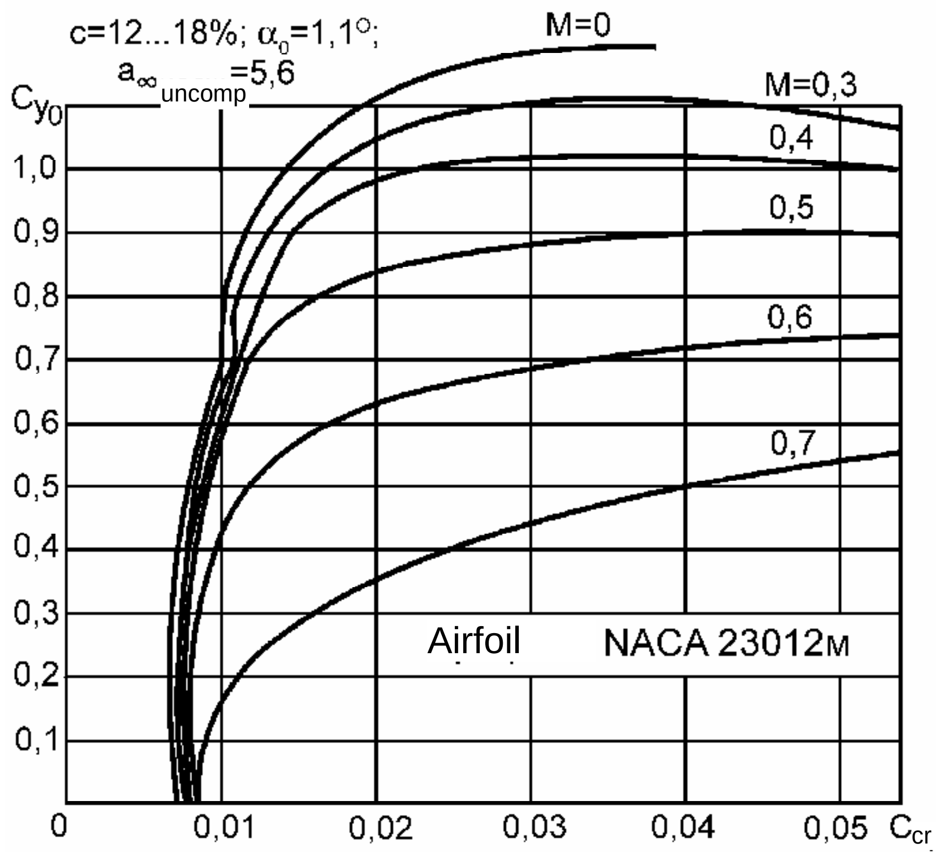


Fig. A.3.1. Airfoil NACA 23012M

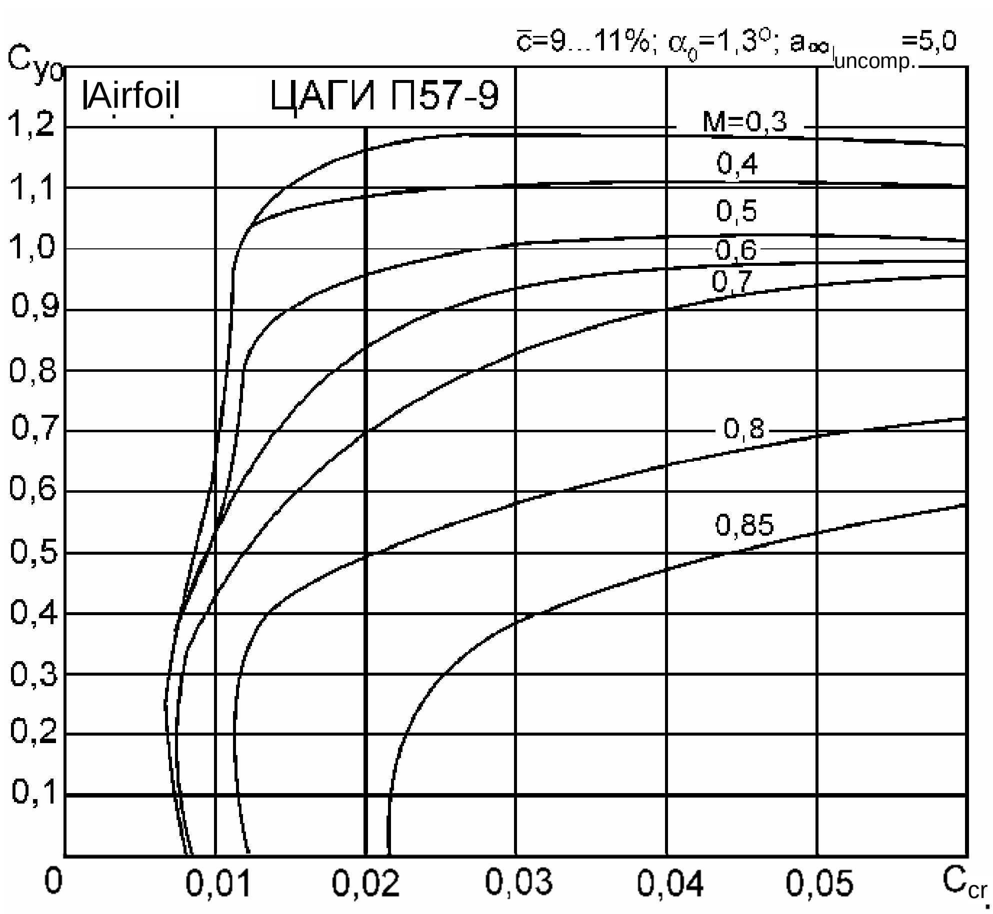


Fig. A.3.2. Airfoil ЦАГИ П57-9

Appendix 4

SOME STATISTICAL DATA ON LIGHT HELICOPTERS

Table A.4.1

Statistical Data on Light Helicopters

Sr. No.	Helicopter Performances	1	2	3	4	5	6	7	8
		Light Helicopters with Piston Engines							
1	Type of Helicopter Performance (designations are given in Tables A.4.1, A.4.2, A.4.3)	A / W 95	Us 254	Us 331	”Angel CH-7”	”Helicycle”	”Mini 500”	”Masquito M-58”	”Angel Kom-press”
2	Designed by	USA	Taiwan	Taiwan	Italy	USA	USA	Belgium	Italy
3	Takeoff mass m_0, kg	222	239	295	360	310	350	390	450
4	Mass of empty helicopter m_{empty}, kg	121	115	151	205	179	155	170	252
5	Full load m_{fload}, kg	101	124	144	155	131	195	220	198
6	Full load ratio $\overline{K}$	0.455	0.5188	0.49	0.43	0.422	0.557	0.564	0.44
7	Q-ty of seats $(n_{cr} + n_{pass})$	1	1	1	1	1	1	1+1p	1+1p
8	Diameter of main rotor D, m	5.94	6.4	6.4	6.5	6.4	6.85	4.6	7.0
9	Q-ty of main rotor rotations n_{mr}, rpm	450	516.5	516.5	520	—	546	750	—
10	Rotor tip velocity ωR, m/s	135… 140	173	173	168	—	167	180.5	—
11	Specific load p, N/m^2	78.63	72.92	90	106.5	113.5	93.2	230.33	114.77
12	Solidity ratio σ	0.0435	0.0358	0.0338	0.035	—	0.0335	0.0498	0.0327

Table A.4.1 (Continued)

Sr. No.	Helicopter Performances	1	2	3	4	5	6	7	8
		Light Helicopters with Piston Engines							
13	Engines	1 piston engine Ro-tax 503	1 piston engine Hirth 2703	1 piston engine Hirth 2706	1 piston engine Ro-tax 582	1 piston engine Ro-tax 618	1 piston engine Ro-tax 582	1 piston engine Jabiru	1 piston engine Ro-tax 914
14	Takeoff rated power N, kW	38.3	40.5	47.84	47.1	54.32	47.1	58.9	84.6
15	Power-to-weight ratio $\tilde{N}_0$, kW/N	0.0176	0.0173	0.0165	0.0133	0.0179	0.0137	0.0154	0.0192
16	Fuel weight m_F, kg	—	—	29.64	31.2	35.1	41.9	40.56	31.2
17	Hover ceiling H_{hc}^{CAI}, m	—	2134	2134	1500	—	2133	3048 СВЗ	—
18	Dynamic ceiling H_{dyn} (service ceiling H_s), m	—	—	—	3500	—	—	—	—
19	Flight range L, km	—	—	—	330	—	—	700	552
20	Maximum speed V_{max}, km/h	222	101	167	140	170	155	180	209
21	Type of transmission (К, З, Ц+Р)	Р+Ц	З	З	К	—	К	К	К

Table A.4.1 (Continued)

Sr. No.	Helicopter Performances	9	10	11	12	13	14	15	16
		Light Helicopters with Piston Engines							
1	Type of Helicopter Performance (designations are given in Tables A.4.1, A.4.2, A.4.3)	Us 496	Us 496	Us 496	Us 496	Us 496	Us 496	Us 496	Us 496
2	Designed by	Taiwan	Taiwan	Taiwan	Taiwan	Taiwan	Taiwan	Taiwan	Taiwan
3	Takeoff mass m_0, kg	514	514	514	514	514	514	514	514
4	Mass of empty helicopter m_{empty}, kg	245	245	245	245	245	245	245	245
5	Full load m_{fload}, kg	269	269	269	269	269	269	269	269
6	Full load ratio $\overline{K}$	0.523	0.523	0.523	0.523	0.523	0.523	0.523	0.523
7	Q-ty of seats $(n_{cr} + n_{pass})$	1+1p	1+1p	1+1p	1+1p	1+1p	1+1p	1+1p	1+1p
8	Diameter of main rotor D, m	7.0	7.0	7.0	7.0	7.0	7.0	7.0	7.0
9	Q-ty of main rotor rotations n_{mr}, rpm	—	—	—	—	—	—	—	—
10	Rotor tip velocity ωR, m/s	—	—	—	—	—	—	—	—
11	Specific load p, N/m^2	131.1	131.1	131.1	131.1	131.1	131.1	131.1	131.1
12	Solidity ratio σ	0.0309	0.0309	0.0309	0.0309	0.0309	0.0309	0.0309	0.0309

Table A.4.1 (Continued)

Sr. No.	Helicopter Performances	9	10	11	12	13	14	15	16
		Light Helicopters with Piston Engines							
13	Engines	1 piston engine Hirth F 30 (A 26)	1 piston engine Hirth F 30	1 piston engine US Air-power	2 piston engines 2xHirth 2706	1 piston engine Lycoming 0-320-B2C	1 piston engine Hirth F 30 (A 26)	1 piston engine Hirth F 30	1 piston engine US Air-power
14	Takeoff rated power N, kW	84.6	95.68	101.57	95.68	119	84.6	95.68	101.57
15	Power-to-weight ratio $\tilde{N}_0$, kW/N	0.0168	0.0217	0.0201	0.0195	0.02205	0.0168	0.0217	0.0201
16	Fuel weight m_F, kg	46.8	46.8	59.28	29.5	—	46.8	46.8	59.28
17	Hover ceiling H_{hc}^{CAI}, m	2134	—	—	—	2400 СВЗ	2134	—	—
18	Dynamic ceiling H_{dyn} (service ceiling H_s), m	3658	—	—	—	—	3658	—	—
19	Flight range L, km	—	—	—	—	1000	—	—	—
20	Maximum speed V_{max}, km/h	158	150	193	160	205	158	150	193
21	Type of transmission (К, З, Ц+Р)	З	К	К	—	К	З	К	К

Table A.4.1 (Continued)

Sr. No.	Helicopter Performances	17	18	19	20	21	22	23	24
		Light Helicopters with Piston Engines							
1	Type of Helicopter Performance (designations are given in Tables A.4.1, A.4.2, A.4.3)	R-22	R-22	R-22	R-22	R-22	R-22	R-22	R-22
2	Designed by	USA	USA	USA	USA	USA	USA	USA	USA
3	Takeoff mass m_0, kg	621	621	621	621	621	621	621	621
4	Mass of empty helicopter m_{empty}, kg	374	374	374	374	374	374	374	374
5	Full load m_{fload}, kg	247	247	247	247	247	247	247	247
6	Full load ratio $\overline{K}$	0.398	0.398	0.398	0.398	0.398	0.398	0.398	0.398
7	Q-ty of seats $(n_{cr} + n_{pass})$	1+1p	1+1p	1+1p	1+1p	1+1p	1+1p	1+1p	1+1p
8	Diameter of main rotor D, m	7.67	7.67	7.67	7.67	7.67	7.67	7.67	7.67
9	Q-ty of main rotor rotations n_{mr}, rpm	540	540	540	540	540	540	540	540
10	Rotor tip velocity ωR, m/s	217	217	217	217	217	217	217	217
11	Specific load p, N/m^2	131.92	131.92	131.92	131.92	131.92	131.92	131.92	131.92
12	Solidity ratio σ	0.03	0.03	0.03	0.03	0.03	0.03	0.03	0.03

Table A.4.1 (Continued)

Sr. No.	Helicopter Performances	17	18	19	20	21	22	23	24
		Light Helicopters with Piston Engines							
13	Engines	1 piston engine Lycoming 0-320-B2C	1 piston engine Lycoming 0-320	1 piston engine Rotorway 162	1 piston engine Lycoming IV0 360-A1A	1 piston engine Lycoming IV0 360-A1A	1 piston engine Textron Lycoming H0 360 C1A	1 piston engine Lycoming 0-320-B2C	1 piston engine Lycom-ing 0-320
14	Takeoff rated power N, kW	119	110.4	110.4	132.48	132.48	132.48	119	110.4
15	Power-to-weight ratio $\tilde{N}_0$, kW/N	0.0193	0.015	0.0165	0.0178	0.0178	0.01703	0.0193	0.015
16	Fuel weight m_F, kg	52 (with oil)	84.24	50.7	91.27	83	—	52 (with oil)	84.24
17	Hover ceiling H_{hc}^{CAI}, m	1586	—	2285	2040 СВЗ	1145	1707	1586	—
18	Dynamic ceiling H_{dyn} (service ceiling H_s), m	4265	—	–	3200	3625	4300	4265	—
19	Flight range L, km	320	—	323	400	328	—	320	—
20	Maximum speed V_{max}, km/h	190	160	185	161	138	—	190	160
21	Type of transmission (К, З, Ц+Р)	К	—	Р+Ц	—	К	К	К	—

Table A.4.1 (Continued)

Sr. No.	Helicopter Performances	25	26	27	28	29	30	31	32
		Light Helicopters with Piston Engines							
1	Type of Helicopter Performance (designations are given in Tables A.4.1, A.4.2, A.4.3)	”Швайцер 300С”	”Швайцер 300С”	”Швайцер 300С”	”Швайцер 300С”	”Швайцер 300С”	”Швайцер 300С”	”Швайцер 300С”	”Швайцер 300С”
2	Designed by	USA	USA	USA	USA	USA	USA	USA	USA
3	Takeoff mass m_0, kg	928.7	928.7	928.7	928.7	928.7	928.7	928.7	928.7
4	Mass of empty helicopter m_{empty}, kg	498.7	498.7	498.7	498.7	498.7	498.7	498.7	498.7
5	Full load m_{fload}, kg	430	430	430	430	430	430	430	430
6	Full load ratio $\overline{K}$	0.463	0.463	0.463	0.463	0.463	0.463	0.463	0.463
7	Q-ty of seats $(n_{cr}+n_{pass})$	1+2p	1+2p	1+2p	1+2p	1+2p	1+2p	1+2p	1+2p
8	Diameter of main rotor D, m	8.17	8.17	8.17	8.17	8.17	8.17	8.17	8.17
9	Q-ty of main rotor rotations n_{mr}, rpm	—	—	—	—	—	—	—	—
10	Rotor tip velocity ωR, m/s	—	—	—	—	—	—	—	—
11	Specific load p, N/m^2	173.87	173.87	173.87	173.87	173.87	173.87	173.87	173.87
12	Solidity ratio σ	—	—	—	—	—	—	—	—

Table A.4.1 (Continued)

Sr. No.	Helicopter Performances	25	26	27	28	29	30	31	32
		Light Helicopters with Piston Engines							
13	Engines	13	1 piston engine Lycoming H10-360-D1A	1 piston engine Lycoming H1D-360-F1AD	—	1 piston engine Lycoming 0-540-F185	1 rotary engine BA3-430	13	1 piston engine Lycoming H10-360-D1A
14	Takeoff rated power N, kW	14	139.84	165.6	—	191.36	176.64	14	139.84
15	Power-to-weight ratio $\tilde{N}_0$, kW/N	15	0.01535	0.0143	0.0158	0.0179	0.0156	15	0.01535
16	Fuel weight m_F, kg	16	100.9	91.26	—	89.7 main 53.8 supp.	226.2	16	100.9
17	Hover ceiling H_{hc}^{CAI}, m	17	—	1250	1300	1524	1600	17	—
18	Dynamic ceiling H_{dyn} (service ceiling H_s), m	18	4400	3658	4700	4267	5000	18	4400
19	Flight range L, km	19	456	483 (without fuel reserve)	100 km (m_{lc}=310 kg) 600 km (m_{lc}=190 kg)	643 (without fuel reserve)	200 km (300 kg) 420 km (250 kg) 800 km (150 kg)	19	456
20	Maximum speed V_{max}, km/h	20	—	188	190	247	215	20	—
21	Type of transmission (К, З, Ц+Р)	21	К	—	—	К	З	21	К

Table A.4.1 (Continued)

Sr. No.	Helicopter Performances	33	34	35	36	37	38	39	40
		Light Helicopters with Piston Engines							
1	Type of Helicopter Performance (designations are given in Tables A.4.1, A.4.2, A.4.3)	Ми 52-1	Ми 52-1	Ми 52-1	Ми 52-1	Ми 52-1	Ми 52-1	Ми 52-1	Ми 52-1
2	Designed by	Russia	Russia	Russia	Russia	Russia	Russia	Russia	Russia
3	Takeoff mass m_0, kg	1200	1200	1200	1200	1200	1200	1200	1200
4	Mass of empty helicopter m_{empty}, kg	694	694	694	694	694	694	694	694
5	Full load m_{fload}, kg	506	506	506	506	506	506	506	506
6	Full load ratio $\bar{K}$	0.422	0.422	0.422	0.422	0.422	0.422	0.422	0.422
7	Q-ty of seats $(n_{cr} + n_{pass})$	1+4p	1+4p	1+4p	1+4p	1+4p	1+4p	1+4p	1+4p
8	Diameter of main rotor D, m	10	10	10	10	10	10	10	10
9	Q-ty of main rotor rotations n_{mr}, rpm	—	—	—	—	—	—	—	—
10	Rotor tip velocity ωR, m/s	—	—	—	—	—	—	—	—
11	Specific load p, N/m^2	149.96	149.96	149.96	149.96	149.96	149.96	149.96	149.96
12	Solidity ratio σ	—	—	—	—	—	—	—	—

Table A.4.1 (End)

Sr. No.	Helicopter Performances	33	34	35	36	37	38	39	40
		Light Helicopters with Piston Engines							
13	Engines	1 rotary engine ВАЗ-430	1 piston engine VB 6.61 auto	1 piston engine M-14B 26B	1 piston engine M-14B 26B	2 rotary engine ВАЗ-430	1 rotary engine ВАЗ-430	1 piston engine VB 6.61 auto	1 piston engine M-14B 26B
14	Takeoff rated power N, kW	198.7	191.36	253.5	253.5	2x161.92	198.7	191.36	253.5
15	Power-to-weight ratio $\tilde{N}_0$, kW/N	0.0169	0.0159	0.0202	0.0178	0.0194	0.0169	0.0159	0.0202
16	Fuel weight m_F, kg	241.8	167.7	—	134.2	—	241.8	167.7	—
17	Hover ceiling H_{hc}^{CAI}, m	1500	—	1500	—	—	1500	—	1500
18	Dynamic ceiling H_{dyn} (service ceiling H_s), m	5000	—	4500	5000	—	5000	—	4500
19	Flight range L, km	200 km (320 kg) 450 km (260 kg) 650 km (210 kg) 800 km (180 kg)	—	360	420	480 300 (400 with fuel reserve)	200 km (320 kg) 450 km (260 kg) 650 km (210 kg) 800 km (180 kg)	—	360
20	Maximum speed V_{max}, km/h	200	176	220	225	210	200	176	220
21	Type of transmission (К, З, Ц+Р)	З	—	З	З	З	З	—	З

Table A.4.2

Statistical Data of Light Helicopters

Sr. No.	Helicopter Performances	1	2	3	4	5	6	7
		Light Helicopters with Gas-Turbine Engines						
1	Type of Helicopter Performance (designations are given in Tables A.4.1, A.4.2, A.4.3)	″Schwei-zer 330″	″Fairchild Hiller FH-1100″	″Hughes 369″ (OH-6A ″Кэйюз″)	″Rogerson Hiller RH-1100″	″Bell 206A″ (″Jet Ranger″)	″Bell OH-58A Kiowa″)	″Hughes 500D″
2	Designed by	USA	USA	USA	USA	USA	USA	USA
3	Takeoff mass m_0, kg	1025	1247	1090	1292	1360	1360	1360
4	Mass of empty helicopter m_{empty}, kg	517	633	557	687	671	718	598
5	Full load m_{fload}, kg	508	614	533	605	689	642	762
6	Full load ratio $\overline{K}$	0.496	0.492	0.489	0.468	0.507	0.472	0.56
7	Q-ty of seats $(n_{cr} + n_{pass})$	1+3p 1+2p	1+3p 1+4p	2+4p	1+6p	1+4p	2+2p	1+4p
8	Diameter of main rotor D, m	8.31	10.8	8.03	10.8	10.16	10.77	8.08
9	Q-ty of main rotor rotations n_{mr}, rpm	471	369	470	369	394	354	478
10	Rotor tip velocity ωR, m/s	205	209	193	209	210	200	202
11	Specific load p, N/m^2	185.5	136	211.2	138.4	164.8	149	260.3
12	Solidity ratio σ	—	0.0318	0.0543	0.0318	0.0414	0.039	0.0674

Table A.4.2 (Continued)

Sr. No.	Helicopter Performances	1	2	3	4	5	6	7
		Light Helicopters with Gas-Turbine Engines						
13	Engines	1 gas-turbine engine Allison Tur-bine 250-C20W	1 gas-turbine engine Allison 250C18	1 gas-turbine engine Allison T-63-A-5A	1 gas-turbine engine Allison 250-C20B	1 gas-turbine engine Allison 250-C18A	1 gas-turbine engine Allison T63-A-700	1 gas-turbine engine Allison 250-C20B
14	Takeoff rated power N, kW	173.7	236 (lim. 204)	233 (lim.173.7)	309	236	236	309.1
15	Power-to-weight ratio $\tilde{N}_0$, kW/N	0.0173	0.0189	0.0193	0.0244	0.0174	0.0174	0.023
16	Fuel weight m_F, kg	224	202	181	202	224	215.3	187.2
17	Hover ceiling H_{hc}^{CAI}, m	1890	2560	2315	3650	1067	1830	2285
18	Dynamic ceiling H_{dyn} (service ceiling H_s), m	—	4235	4740	5275	5200	—	4570
19	Flight range L, km	596	—	—	629	630	490	482
20	Maximum speed V_{max}, km/h	200	204	229	204	241	~220	282
21	Type of transmission (К, З, Ц+Р)	—	З	З	З	З	З	З

Table A.4.2 (Continued)

Sr. No.	Helicopter Performances	8	9	10	11	12	13	14	15
		Light Helicopters with Gas-Turbine Engines							
1	Type of Helicopter Performance (designations are given in Tables A.4.1, A.4.2, A.4.3)	EC 120 XB (fenestron)	EC 120 XB (fenestron)	EC 120 XB (fenestron)	EC 120 XB (fenestron)	EC 120 XB (fenestron)	EC 120 XB (fenestron)	EC 120 XB (fenestron)	EC 120 XB (fenestron)
2	Designed by	International Program	International Program	International Program	International Program	International Program	International Program	International Program	International Program
3	Takeoff mass m_0, kg	1550 max	1550 max	1550 max	1550 max	1550 max	1550 max	1550 max	1550 max
4	Mass of empty helicopter m_{empty}, kg	852	852	852	852	852	852	852	852
5	Full load m_{fload}, kg	698	698	698	698	698	698	698	698
6	Full load ratio $\overline{K}$	0.45	0.45	0.45	0.45	0.45	0.45	0.45	0.45
7	Q-ty of seats $(n_{cr} + n_{pass})$	1+4p	1+4p	1+4p	1+4p	1+4p	1+4p	1+4p	1+4p
8	Diameter of main rotor D, m	10.2	10.2	10.2	10.2	10.2	10.2	10.2	10.2
9	Q-ty of main rotor rotations n_{mr}, rpm	—	—	—	—	—	—	—	—
10	Rotor tip velocity ωR, m/s	—	—	—	—	—	—	—	—
11	Specific load p, N/m^2	186.2	186.2	186.2	186.2	186.2	186.2	186.2	186.2
12	Solidity ratio σ	0.044	0.044	0.044	0.044	0.044	0.044	0.044	0.044

Table A.4.2 (Continued)

Sr. No.	Helicopter Performances	8	9	10	11	12	13	14	15
		Light Helicopters with Gas-Turbine Engines							
13	Engines	1 gas-turbine engine Turboméca TM319 ″Arrius″ 1B1	1 turboprop engine Allison 250-C20B	1 gas-turbine engine A250-C20R/2	1 turboprop engine Turboméca Артуст IIIB	1 turboprop engine Turboméca Астазу IIIA	1 turboprop engine Allison 250-C18A	1 gas-turbine engine Turboméca TM319 ″Arrius″ 1B1	1 turboprop engine Allison 250-C20B
14	Takeoff rated power N, kW	312	309	270	640	436	254	312	309
15	Power-to-weight ratio $\tilde{N}_0$, kW/N	0.0205	0.0230	—	0.0333	0.0245	0.0174	0.0205	0.0230
16	Fuel weight m_F, kg	332	62 supp. 187 main	390	— 447 main	573 supp. 417 main	— 224.6 main	332	62 supp. 187 main
17	Hover ceiling H_{hc}^{CAI}, m	2400	2286	2900	5000	2000	1067	2400	2286
18	Dynamic ceiling H_{dyn} (service ceiling H_s), m	4875	4481	—	6400	5000	5200	4875	4481
19	Flight range L, km	600	515	600	600	785	630	600	515
20	Maximum speed V_{max}, km/h	235	282	245	210	310	241	235	282
21	Type of transmission (К, З, Ц+Р)	—	З	—	З	З	З	—	З

Table A.4.2 (Continued)

Sr. No.	Helicopter Performances	16	17	18	19	20	21	22	23
		Light Helicopters with Gas-Turbine Engines							
1	Type of Helicopter Performance (designations are given in Tables A.4.1, A.4.2, A.4.3)	SA-316C ”Alouette III”	SA-316C ”Alouette III”	SA-316C ”Alouette III”	SA-316C ”Alouette III”	SA-316C ”Alouette III”	SA-316C ”Alouette III”	SA-316C ”Alouette III”	SA-316C ”Alouette III”
2	Designed by	France	France	France	France	France	France	France	France
3	Takeoff mass $\boldsymbol{m_0}$, kg	2250	2250	2250	2250	2250	2250	2250	2250
4	Mass of empty helicopter $\boldsymbol{m_{empty}}$, kg	1134	1134	1134	1134	1134	1134	1134	1134
5	Full load $\boldsymbol{m_{fload}}$, kg	1116	1116	1116	1116	1116	1116	1116	1116
6	Full load ratio $\boldsymbol{\overline{K}}$	0.496	0.496	0.496	0.496	0.496	0.496	0.496	0.496
7	Q-ty of seats $(\boldsymbol{n_{cr}} + \boldsymbol{n_{pass}})$	(1...2)+6p	(1...2)+6p	(1...2)+6p	(1...2)+6p	(1...2)+6p	(1...2)+6p	(1...2)+6p	(1...2)+6p
8	Diameter of main rotor $\boldsymbol{D}$, m	11.02	11.02	11.02	11.02	11.02	11.02	11.02	11.02
9	Q-ty of main rotor rotations $\boldsymbol{n_{mr}}$, rpm	354	354	354	354	354	354	354	354
10	Rotor tip velocity $\boldsymbol{\omega R}$, m/s	205	205	205	205	205	205	205	205
11	Specific load $\boldsymbol{p}$, N/m^2	236	236	236	236	236	236	236	236
12	Solidity ratio $\boldsymbol{\sigma}$	0.0607	0.0607	0.0607	0.0607	0.0607	0.0607	0.0607	0.0607

Table A.4.2 (Continued)

Sr. No.	Helicopter Performances	16	17	18	19	20	21	22	23
		Light Helicopters with Gas-Turbine Engines							
13	Engines	1 turboprop engine Turboméca Artouste IIID	1 turboprop engine General Electric CT58-110-1	1 turboprop engine Lycoming T53-L-13	2 turboprop engines United Aircraft T400-CP-400	2 turboprop engines Allison 250-C20F	2 turboprop engines Allison 250-C20B	1 turboprop engine Turboméca Artouste IIID	1 turboprop engine General Electric CT58-110-1
14	Takeoff rated power N, kW	640	994	1044	671x2	317	313	640	994
15	Power-to-weight ratio $\tilde{N}_0$, kW/N	0.0288	0.0281	0.0242	0.0296	0.0276	0.0257	0.0288	0.0281
16	Fuel weight m_F, kg	— 436,8 main	— 1216,8 main	623	791	569	445	— 436,8 main	— 1216,8 main
17	Hover ceiling H_{hc}^{CAI}, m	1600	1400	1220	2500	2050	1615	1600	1400
18	Dynamic ceiling H_{dyn} (service ceiling H_s), m	4000	3570	3840	3215	4400	5180	4000	3570
19	Flight range L, km	480	743	511	577	740	575	480	743
20	Maximum speed V_{max}, km/h	220	163	204	—	233	270	220	163
21	Type of transmission (К, З, Ц+Р)	З	З	З	З	З	З	З	З

Table A.4.2 (Continued)

Sr. No.	Helicopter Performances	24	25	26	27	28	29
		Light Helicopters with Gas-Turbine Engines					
1	Type of Helicopter Performance (designations are given in Tables A.4.1, A.4.2, A.4.3)	EC-135	A.109A Mк.2 "Hi-rundo"	A.109A	"Ansat"	BK 117	OH-X
2	Designed by	International Program	Italy	Italy	Russia	Germany Japan	Japan
3	Takeoff mass m_0, kg	2500	2450	2360	3300	3000	3500 max
4	Mass of empty helicopter m_{empty}, kg	1300	1418	1298	2000	1695	—
5	Full load m_{fload}, kg	1200	1032	1062	1300	1305	—
6	Full load ratio $\overline{K}$	0.48	0.422	0.45	0.394	0.435	—
7	Q-ty of seats $(n_{cr} + n_{pass})$	1+6p	(1…2)+ +(6…7)p	(1…2)+ +(6…7)p	1+9p	1+ +6…7(10)p	2
8	Diameter of main rotor D, m	10.8	11.0	11.0	11.5	11.0	11.5
9	Q-ty of main rotor rotations n_{mr}, rpm	—	385	385	—	383	—
10	Rotor tip velocity ωR, m/s	—	222	222	—	221	—
11	Specific load p, N/m^2	268	258	248	312	316	—
12	Solidity ratio σ	—	0.0764	0.0764	—	0.0741	—

Table A.4.2 (Continued)

Sr. No.	Helicopter Performances	24	25	26	27	28	29
		Light Helicopters with Gas-Turbine Engines					
13	Engines	2 gas-turbine engines Turboméca ″Arrius″ 2B	2 turboprop engines Allison 250-C20B	2 turboprop engines Allison 250-C20B	2 gas-turbine engines Pratt Whitney/ Klimov PW-206 C	2 turboprop engines Avco Lycoming LTS101-650 B-1	2 gas-turbine engines Mitsubishi TC1-10
14	Takeoff rated power N, kW	470	313	313	464	410	708
15	Power-to-weight ratio $\tilde{N}_0$, kW/N	0.0383	0.0253	0.0265	0.0287	0.0273	—
16	Fuel weight m_F, kg	83 supp. 500 main	429	429	—	474	—
17	Hover ceiling H_{hc}^{CAI}, m	1100	2835	3290	3000	3170	—
18	Dynamic ceiling H_{dyn} (service ceiling H_s), m	610	5485	5300	(5500)	—	—
19	Flight range L, km	750	648	625	595	498	—
20	Maximum speed V_{max}, km/h	274	311	280	280	278	—
21	Type of transmission (К, З, Ц+Р)	—	З	З	—	З	—

Table A.4.2 (Continued)

Sr. No.	Helicopter Performances	30	31	32	33	34
		Light Helicopters with Gas-Turbine Engines				
1	Type of Helicopter Performance (designations are given in Tables A.4.1, A.4.2, A.4.3)	A. 129	Ми-2	”Bell 222A”	AL H	SA 365N (fenestron)
2	Designed by	Italy	Russia MB3	USA	India	France
3	Takeoff mass m_0, kg	3700	3550	3560	4000	3850
4	Mass of empty helicopter m_{empty}, kg	2529	2420	2063	2680	2011
5	Full load m_{fload}, kg	1171	1130	1497	1320	1839
6	Full load ratio $\overline{K}$	0.316	0.318	0.42	0.33	0.476
7	Q-ty of seats $(n_{cr} + n_{pass})$	2	1+8p	1…2+ +(6…7)p	2+10(14)p	(1…2)+ +(12…13)p
8	Diameter of main rotor D, m	11.9	14.5	12.12	13.2	11.93
9	Q-ty of main rotor rotations n_{mr}, rpm	337	—	338	314	349,5
10	Rotor tip velocity ωR, m/s	210	—	215	216,9	218
11	Specific load p, N/m^2	333	211	308	325	344
12	Solidity ratio σ	0.0895	0.0527	0.0767	0.0965	0.0822

Table A.4.2 (Continued)

Sr. No.	Helicopter Performances	30	31	32	33	34
		Light Helicopters with Gas-Turbine Engines				
13	Engines	2 turboprop engines Rolls-Royce GEM-2 MK1004D	2 gas turbine engines-350 of Izotov JV	2 turboprop engines Avco Lycoming LTS-101-650C-3	2 turboprop engines Turboméca TM 332-2B	2 turboprop engines Turboméca Ariel 1C1
14	Takeoff rated power N, kW	708	2x294	2x456	2x788 (TP-1240)	2x641
15	Power-to-weight ratio $\tilde{N}_0$, kW/N	0.0383	0.0172	0.026	0.0316	0.0248
16	Fuel weight m_F, kg	758	393 supp. 500 main	554	1160	889
17	Hover ceiling H_{hc}^{CAI}, m	2390	2000	1400	3000 CB3	1050
18	Dynamic ceiling H_{dyn} (service ceiling H_s), m	—	4000	—	6000	4575
19	Flight range L, km	—	580	527	800	880
20	Maximum speed V_{max}, km/h	259	210	278	290	305
21	Type of transmission (К, З, Ц+Р)	З	З	З	З	З

Table A.4.2 (Continued)

Sr. No.	Helicopter Performances	35	36	37	38	39
		Light Helicopters with Gas-Turbine Engines				
1	Type of Helicopter Performance (designations are given in Tables A.4.1, A.4.2, A.4.3)	SA 366N (fenestron)	SA 365C (fenestron)	”Lynx II” AH Mк 5	S-76B	”Tiger”
2	Designed by	France	France	England	USA	International Program
3	Takeoff mass $\boldsymbol{m_0}$, kg	4050	3000	4535	4536	6000 мах
4	Mass of empty helicopter $\boldsymbol{m_{empty}}$, kg	2718	1876	2658	2390	3300
5	Full load $\boldsymbol{m_{fload}}$, kg	1342	1124	1877	2146	2700
6	Full load ratio $\overline{\boldsymbol{K}}$	0.329	0.375	0.414	0.473	0.45
7	Q-ty of seats $(\boldsymbol{n_{cr}} + \boldsymbol{n_{pass}})$	3	(1…2)+ +(12…13)p	(1…2)+7p	2+12p	2
8	Diameter of main rotor $\boldsymbol{D}$, m	11.93	11.68	12.8	13.41	13
9	Q-ty of main rotor rotations $\boldsymbol{n_{mr}}$, rpm	349	348	313	313	–
10	Rotor tip velocity $\boldsymbol{\omega R}$, m/s	218	213	210	220	–
11	Specific load $\boldsymbol{p}$, N/m^2	361	280	353	321	444
12	Solidity ratio $\boldsymbol{\sigma}$	0.0822	0.0763	0.0786	0.0741	–

Sr. No.	Helicopter Performances	35	36	37	38	39
		Light Helicopters with Gas-Turbine Engines				
13	Engines	2 turboprop engines Avco Lycoming TS101-750A-1	2 turboprop engines Turboméca Ariel	2 turboprop engines Rolls-Royce GEM 41-1 MK 20101	2 turboprop engines Allison 250-C30	2 gas-turbine engines MTU Rolls-Royce
14	Takeoff rated power N, kW	2x618	2x472	2x736	2x478	2x958
15	Power-to-weight ratio $\tilde{N}_0$, kW/N	0.0228	0.0319	0.0329	0.0214	0.0325
16	Fuel weight m_F, kg	889	1037 supp. 499 main	352 supp. 723 main	– 826 main	– 1080 main
17	Hover ceiling H_{hc}^{CAI}, m	0	2840	3230	1890	2000
18	Dynamic ceiling H_{dyn} (service ceiling H_s), m	–	6000	–	4570	–
19	Flight range L, km	760	465	630	748	t=170 min
20	Maximum speed V_{max}, km/h	324	315	259	269	–
21	Type of transmission (К, З, Ц+Р)	З	З	З	З	З

Appendix 5

STRUCTURES OF LIGHT HELICOPTERS

Fig. A.5.1. Helicopter A / W 95

Fig. A.5.2. Helicopter Us 254

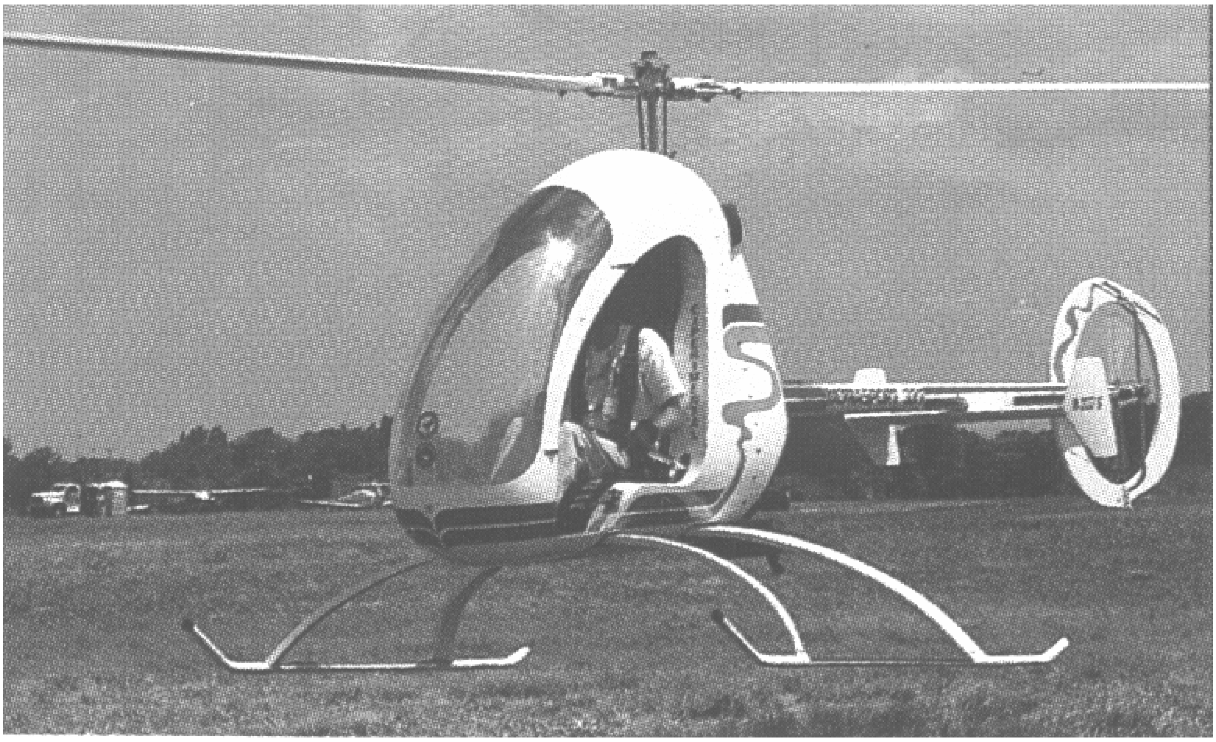


Fig. A.5.3. Helicopter Us 331

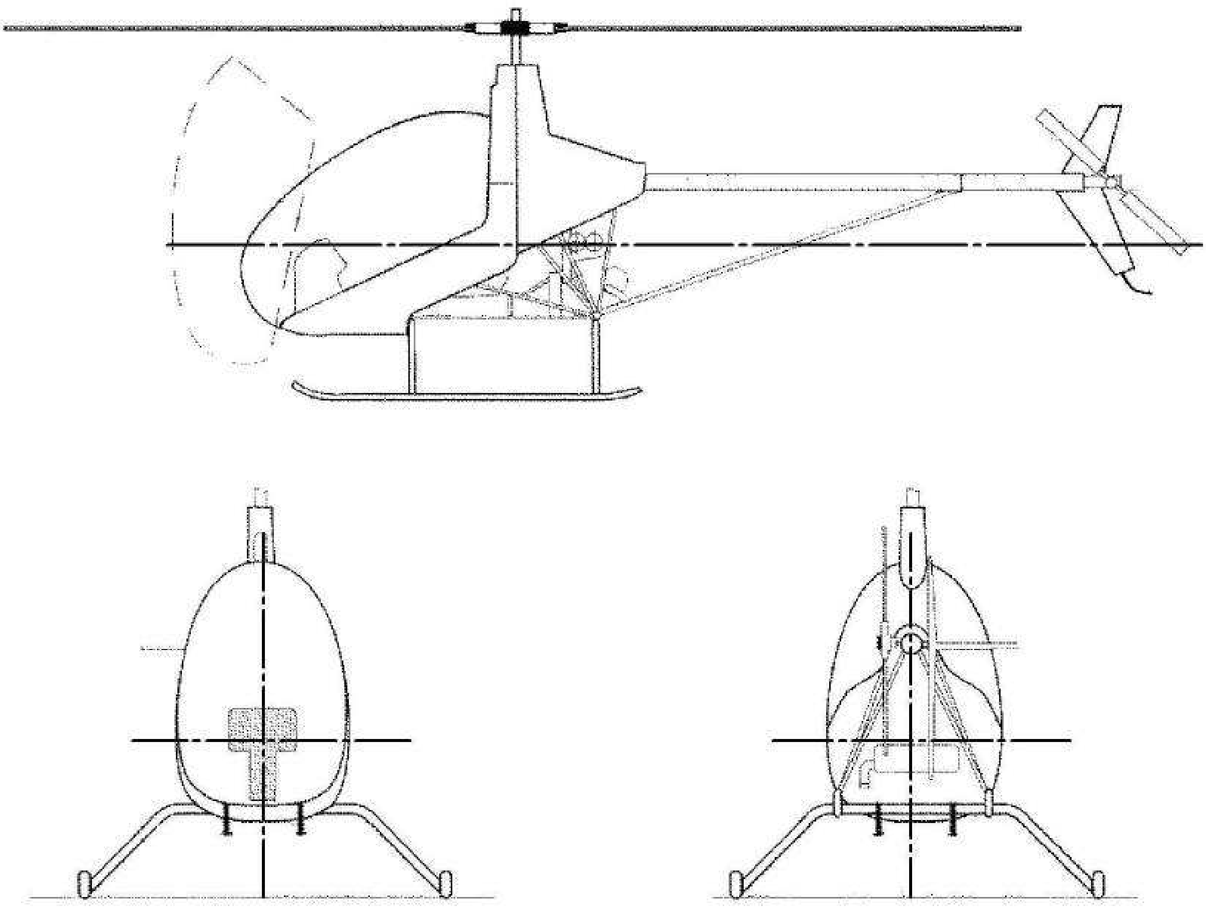


Fig. A.5.4. Structure of Helicopter ″Angel CH-7″

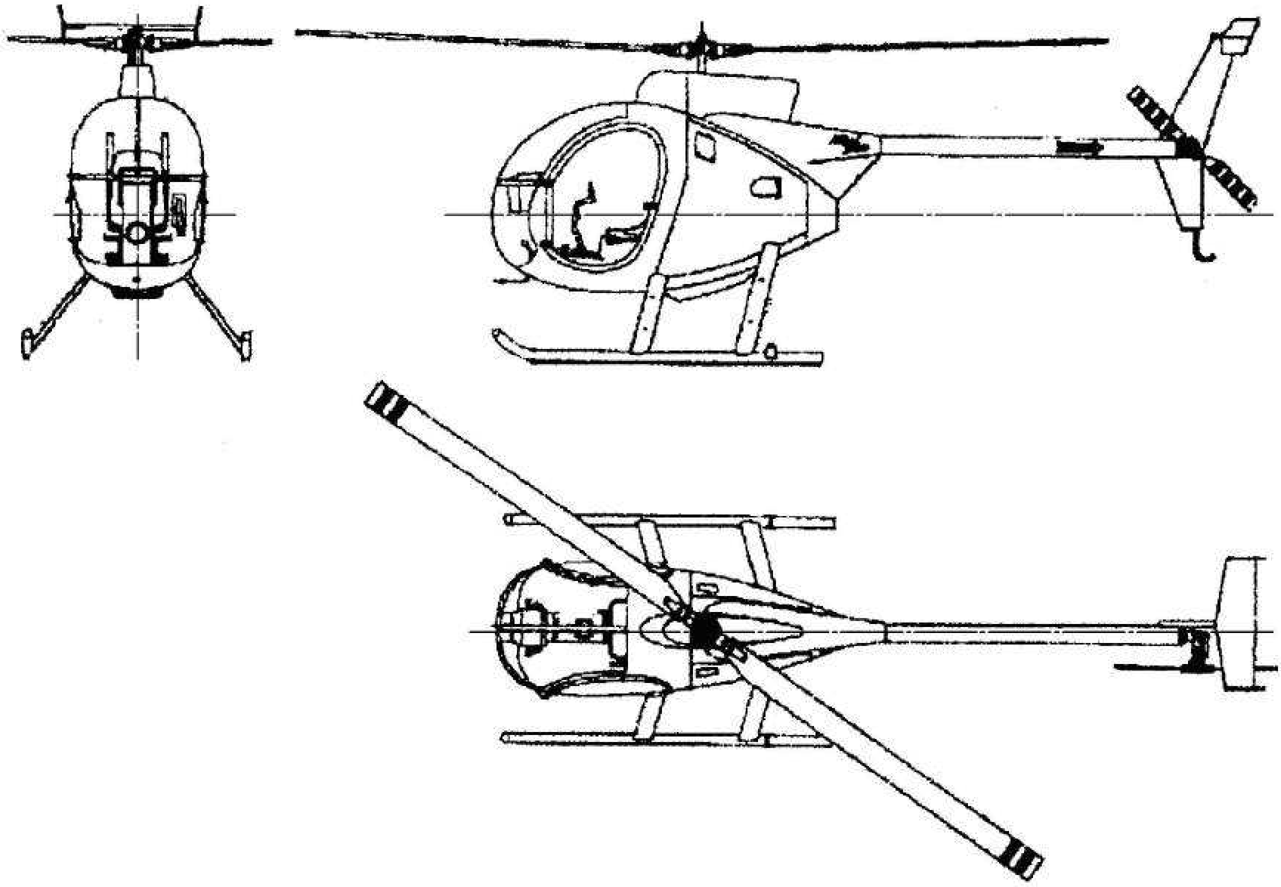


Fig. A.5.5. Structure of Helicopter ″Mini 500″

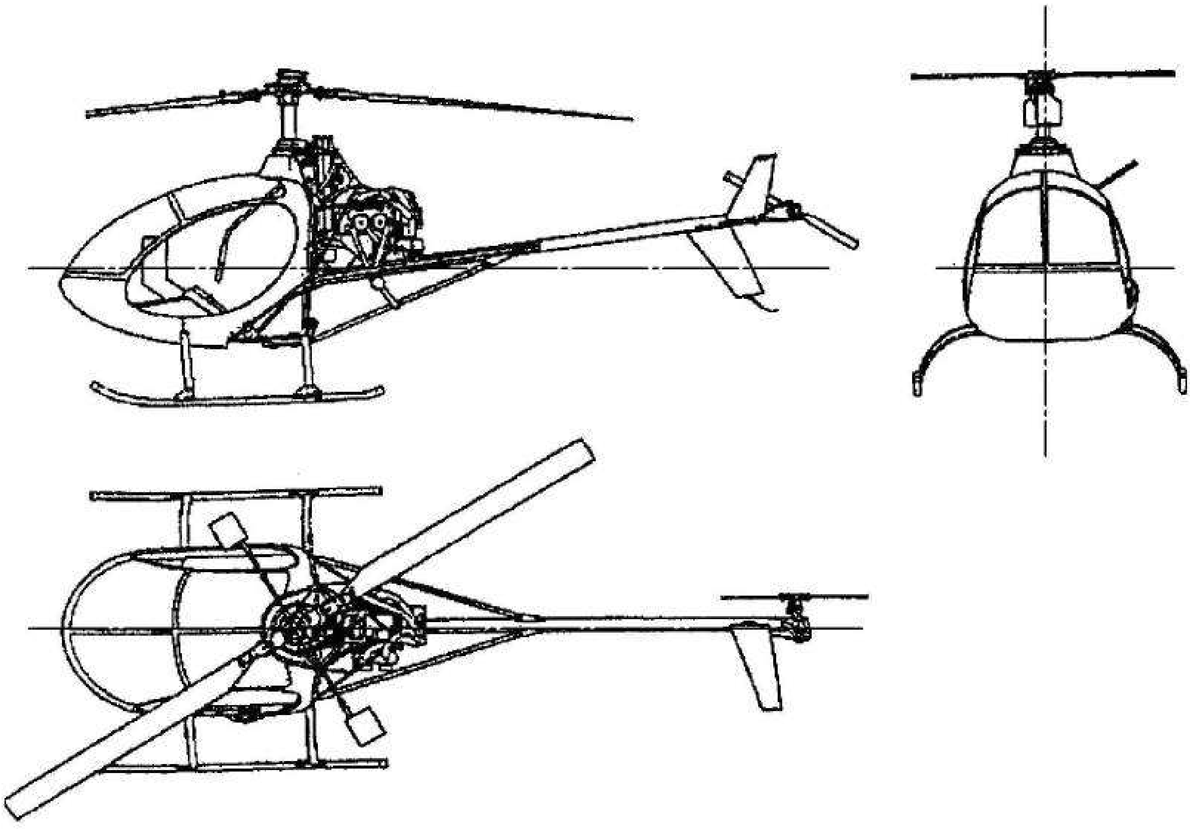


Fig. A.5.6. Structure of Helicopter ″Masguito M-58″

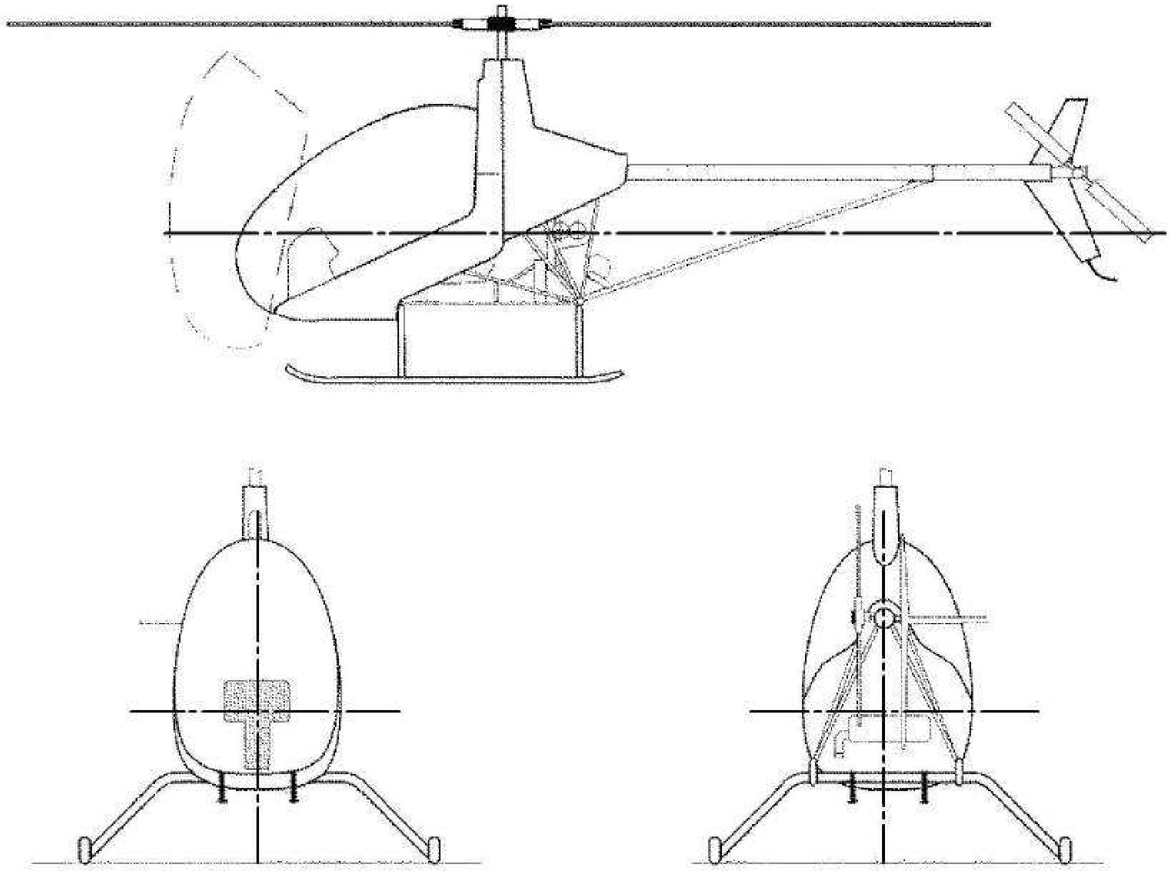


Fig. A.5.7. Structure of Helicopter ″Angel Kompress″

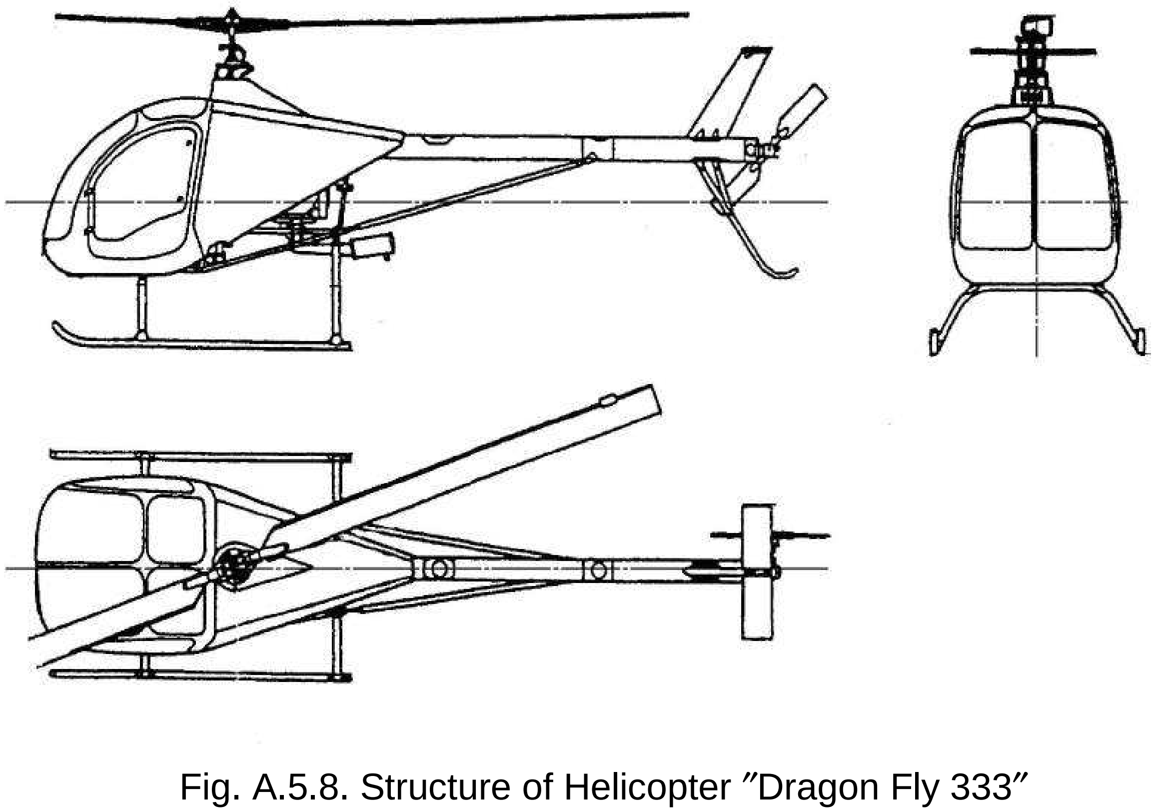


Fig. A.5.8. Structure of Helicopter ″Dragon Fly 333″

Fig. A.5.9. Helicopter ″Babe Bell″

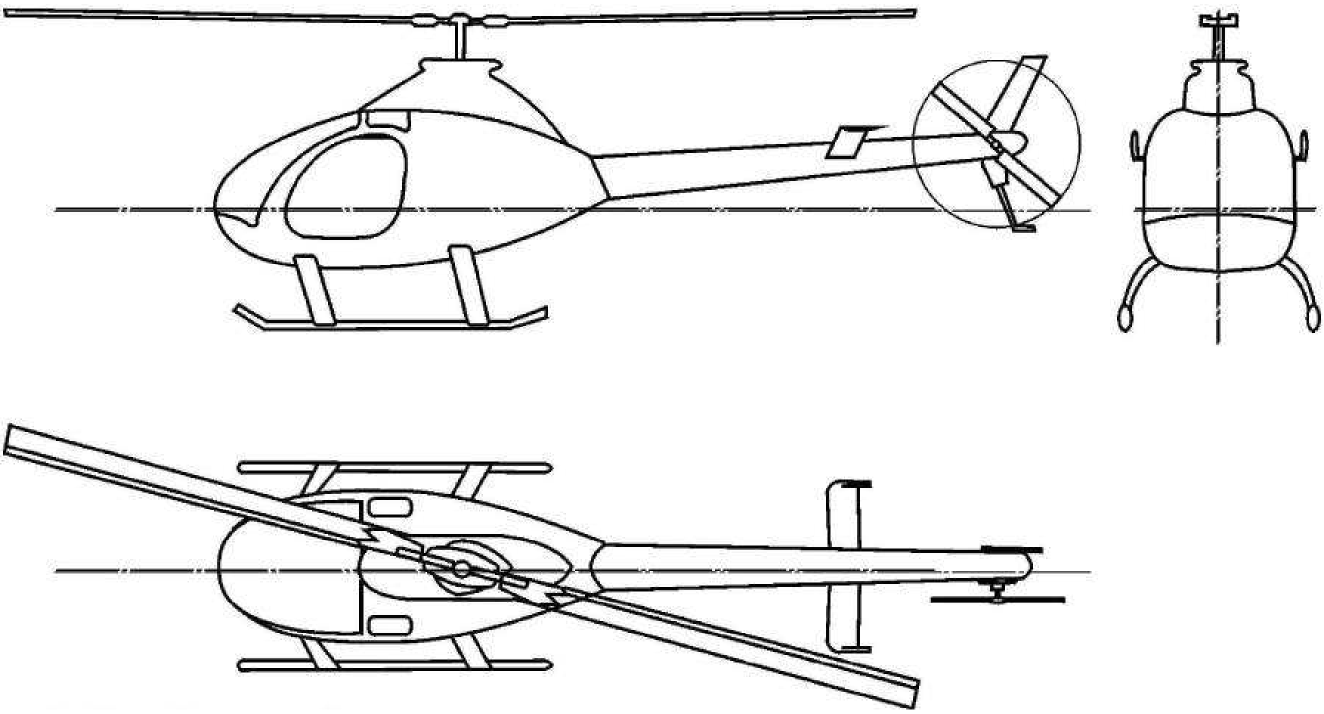


Fig. A.5.10. Structure of Helicopter ″Exec 162F″

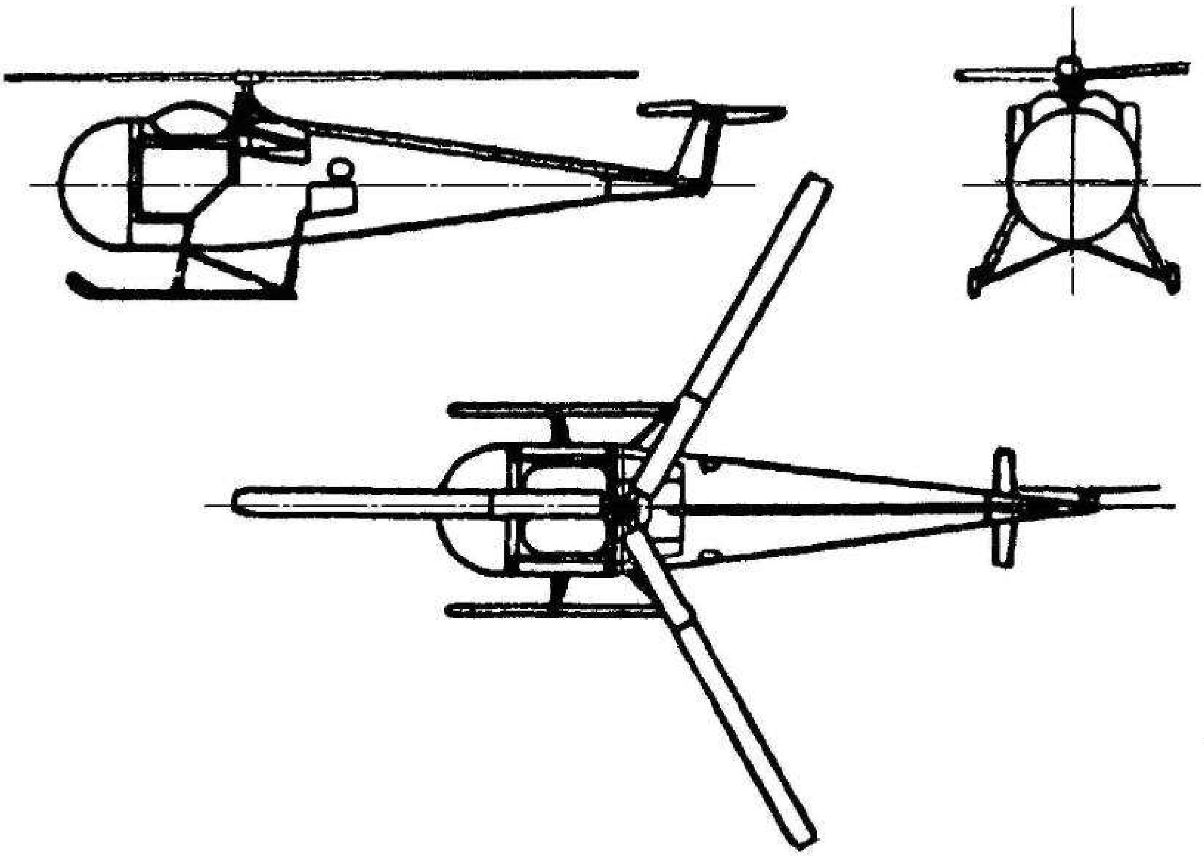


Fig. A.5.11. Structure of Helicopter ″Brantly B-28″

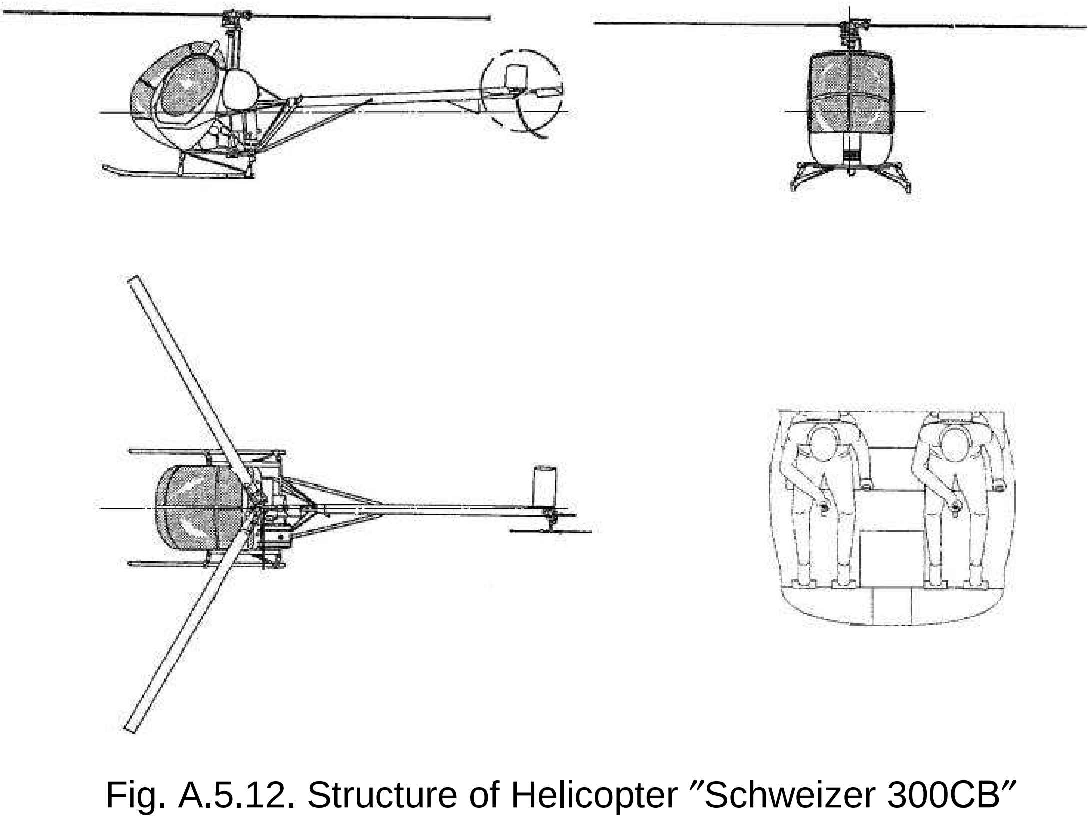


Fig. A.5.12. Structure of Helicopter ″Schweizer 300CB″

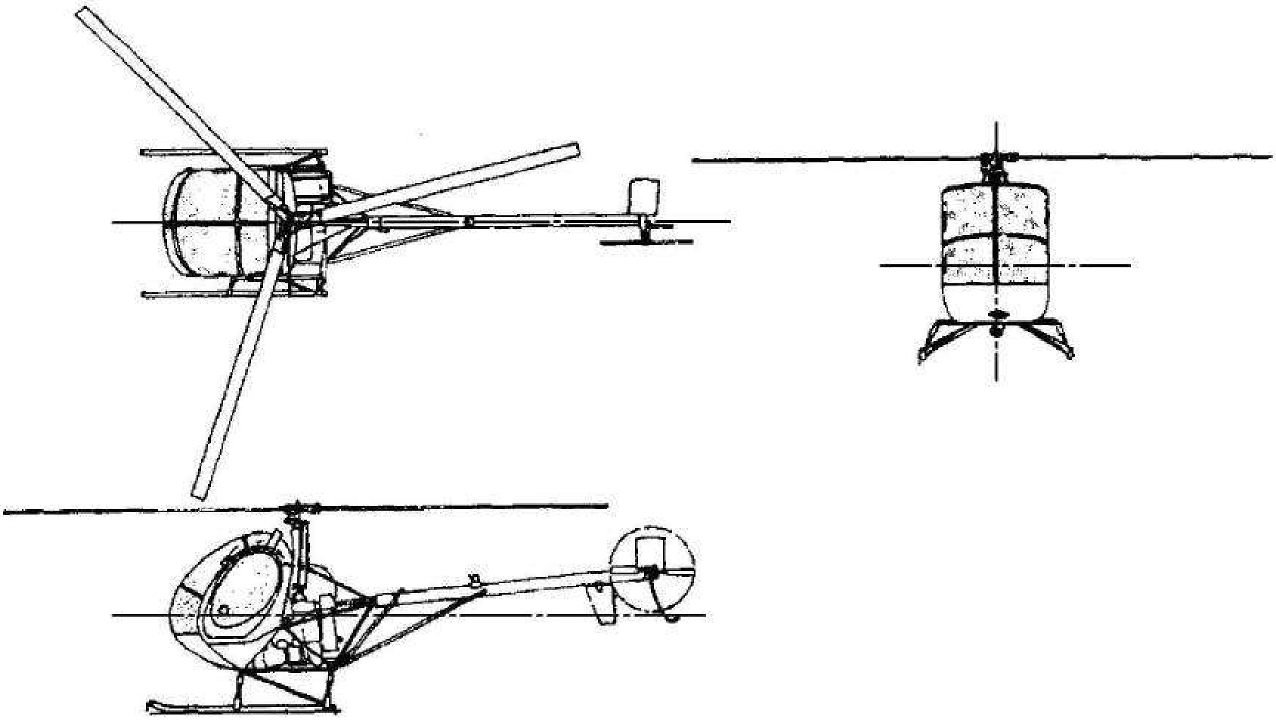


Fig. A.5.13. Structure of Helicopter ″Hughes 269A″

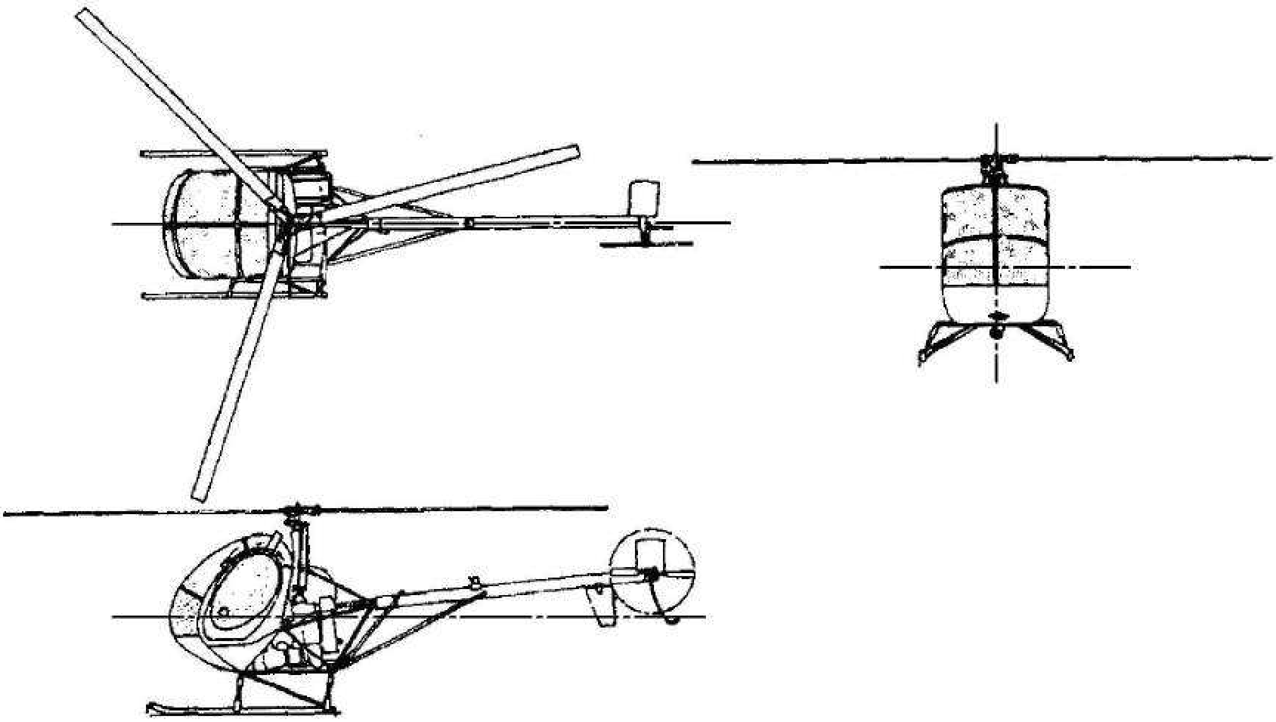


Fig. A.5.14. Structure of Helicopter ″Schweizer 300C″

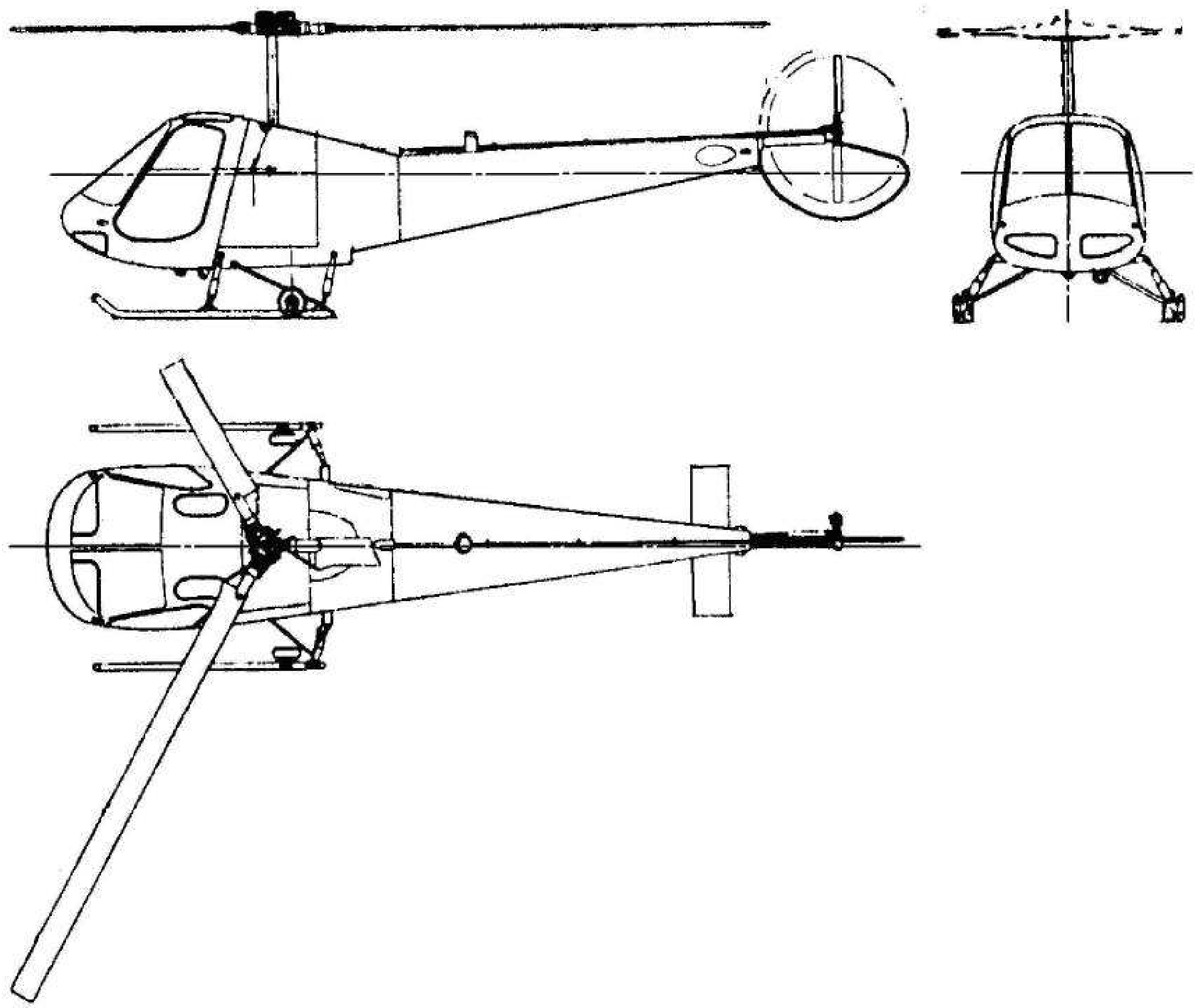


Fig. A.5.15. Structure of Helicopter ″ Enstrom F-28A″

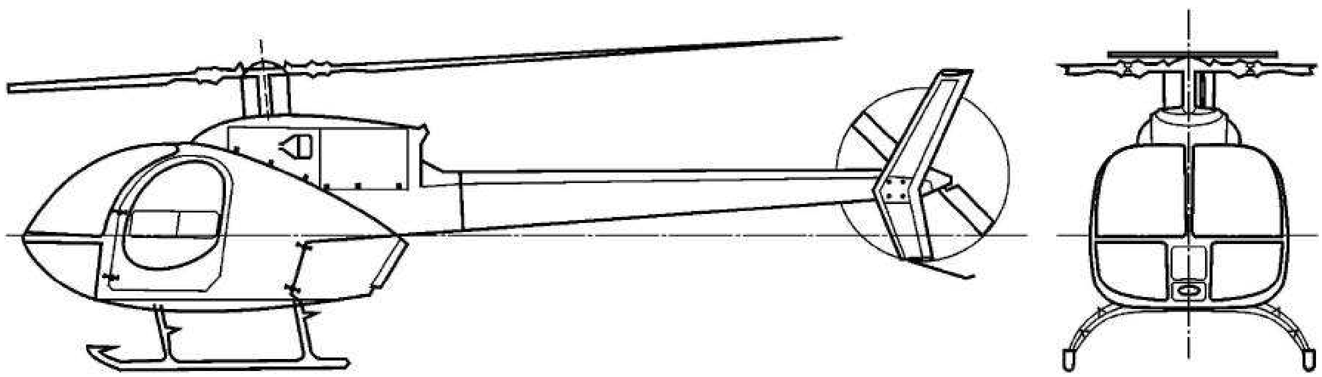


Fig. A.5.16. Structure of Helicopter ″Актай″

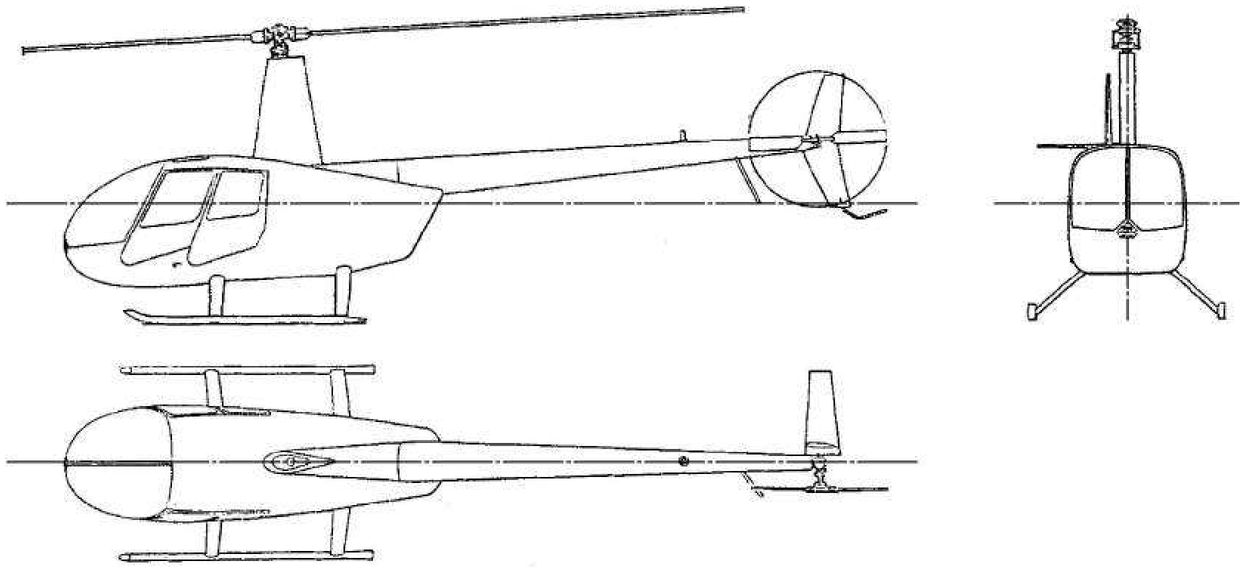


Fig. A.5.17. Structure of Helicopter R-44

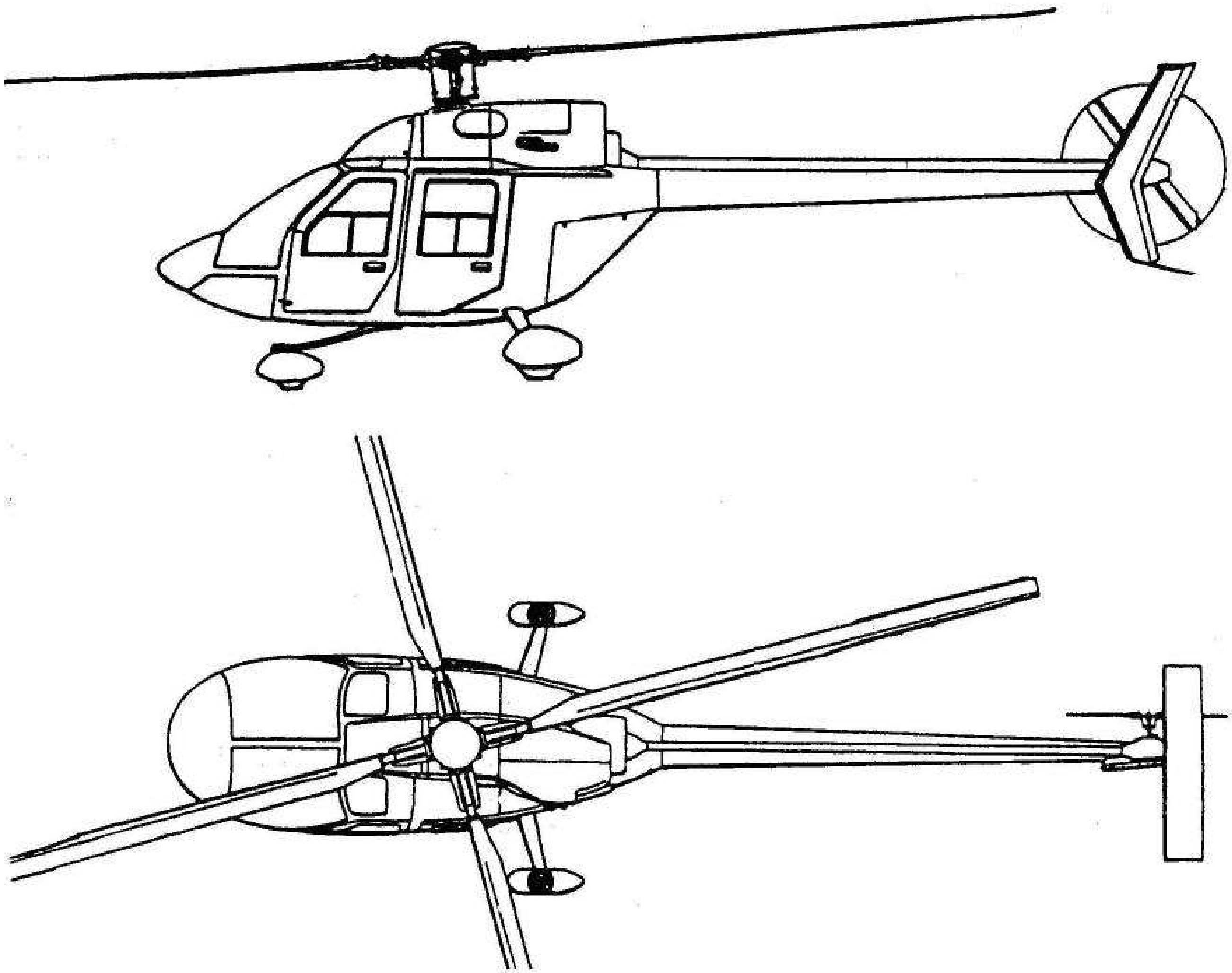


Fig. A.5.18. Structure of Helicopter Ми-52

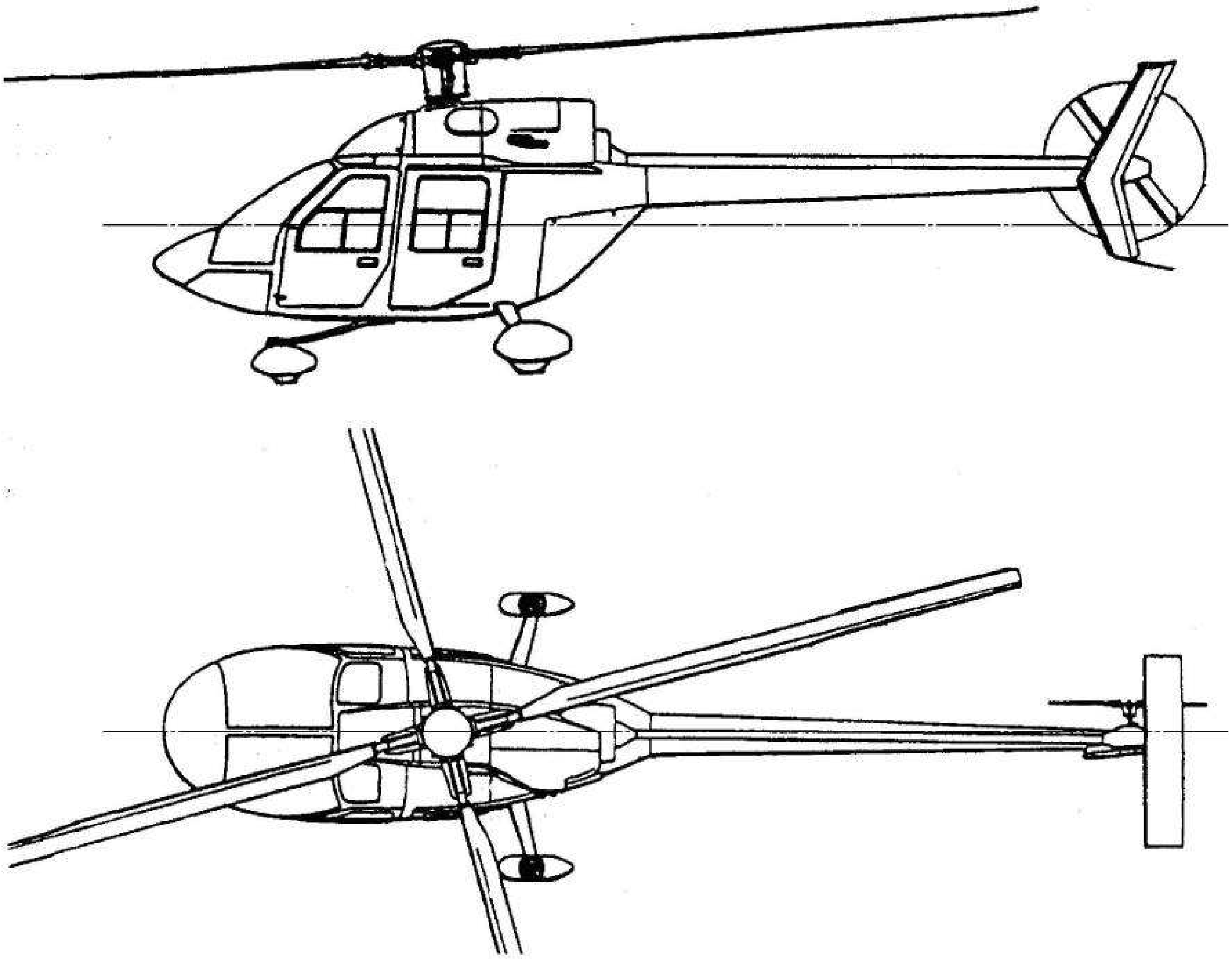


Fig. A.5.19. Structure of Helicopter Ми-52-1

Fig. A.5.20. Helicopter ″Humming Bird″

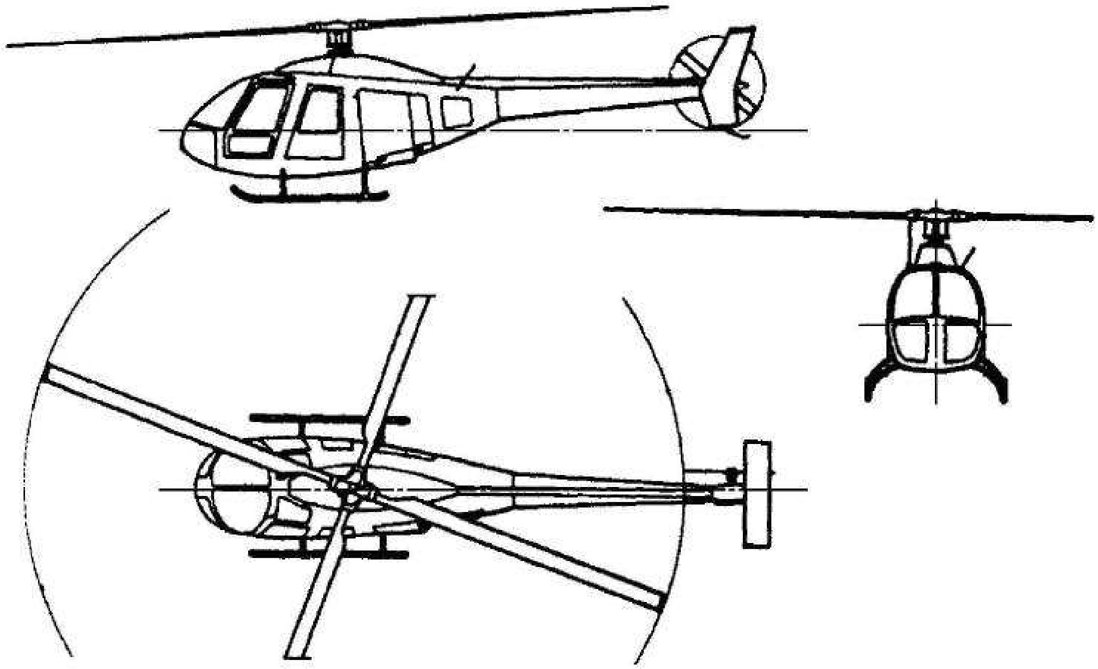


Fig. A.5.21. Structure of Helicopter Ми-34

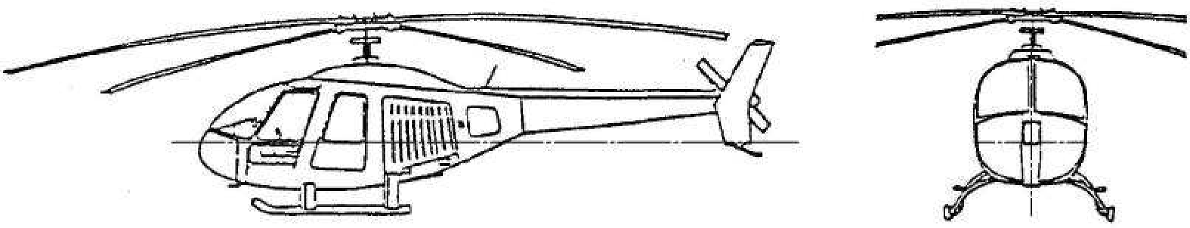


Fig. A.5.22. Structure of Helicopter Ми-34С

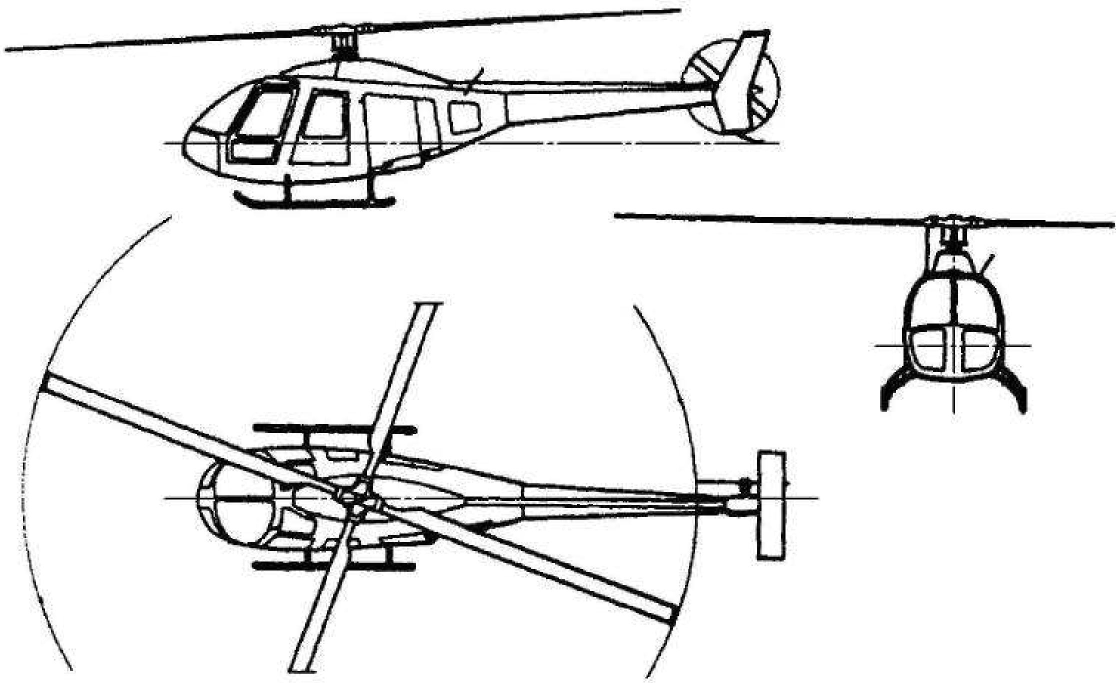


Fig. A.5.23. Structure of Helicopter Ми-34М

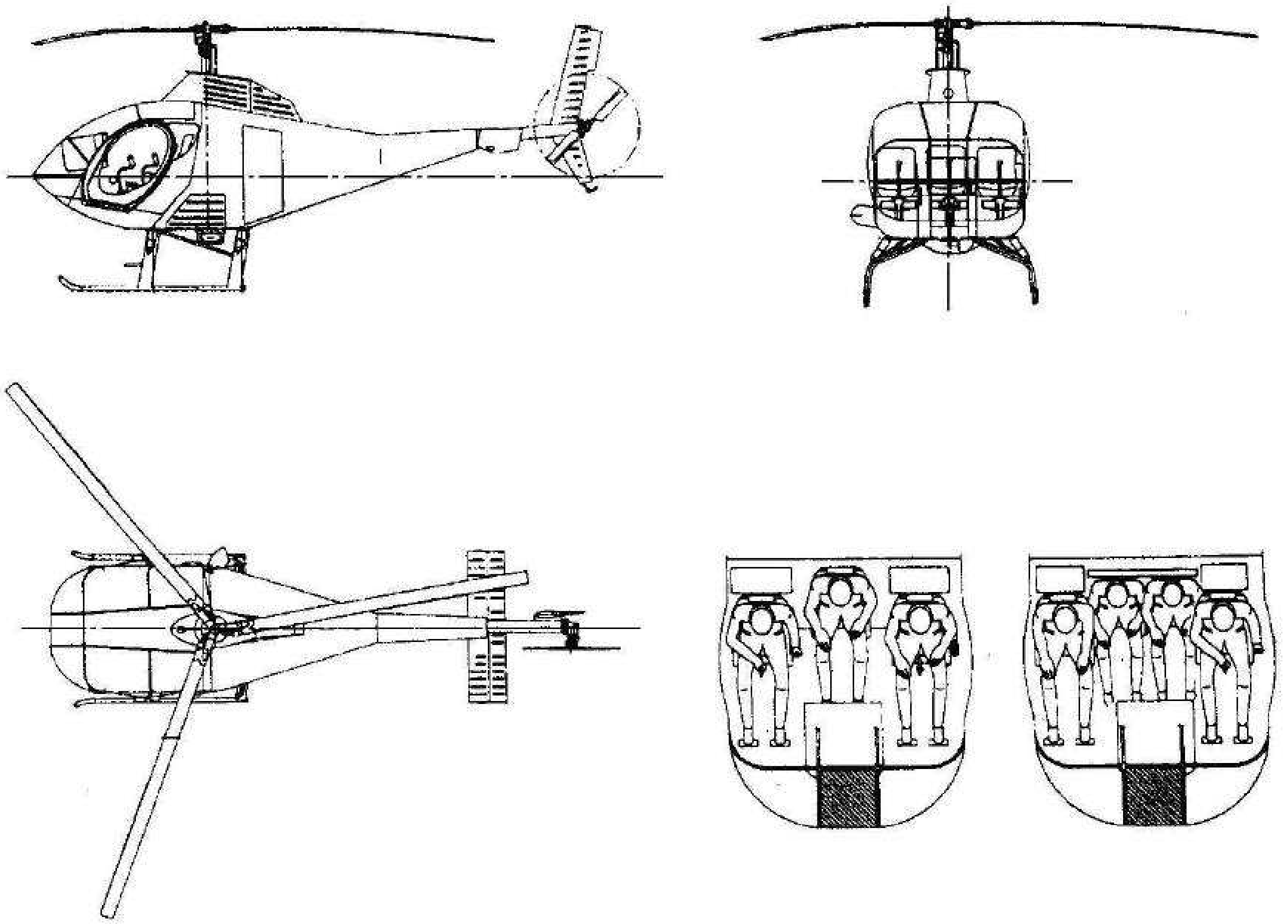


Fig. A.5.24. Structure of Helicopter ″Schweizer 330″

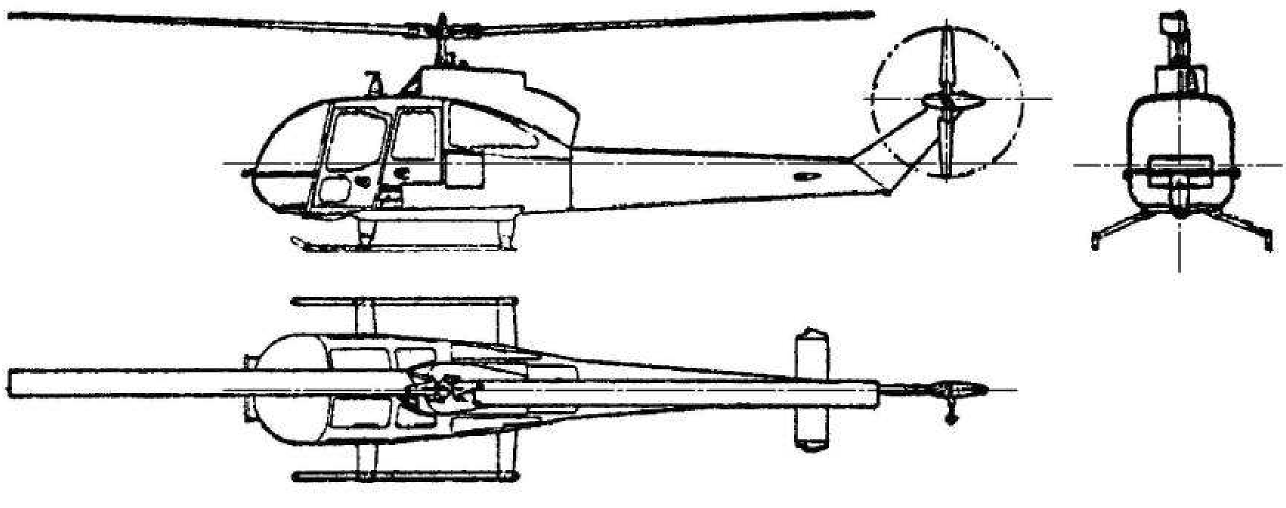


Fig. A.5.25. Structure of Helicopter ″Fairchild Hiller FH-1100″

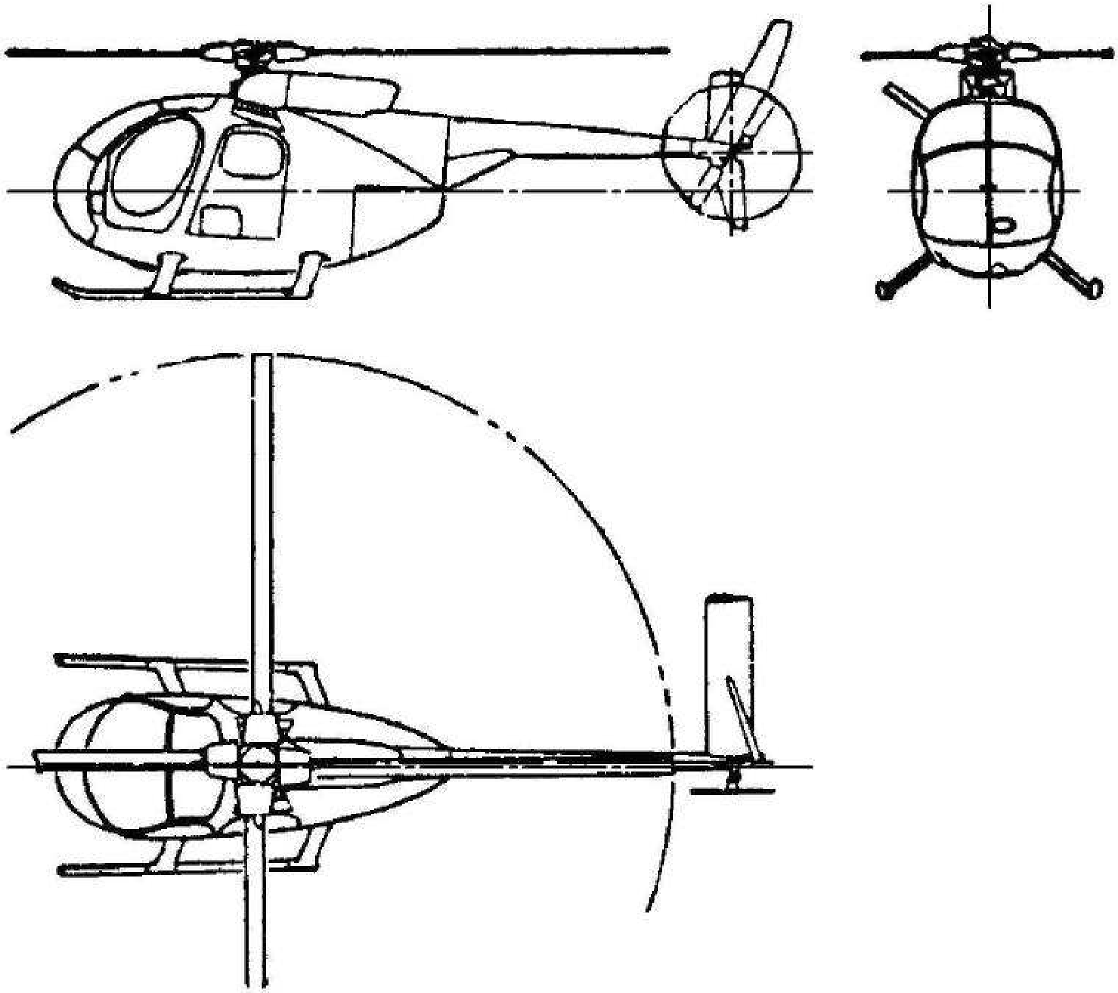


Fig. A.5.26. Structure of Helicopter ″Hughes 369″

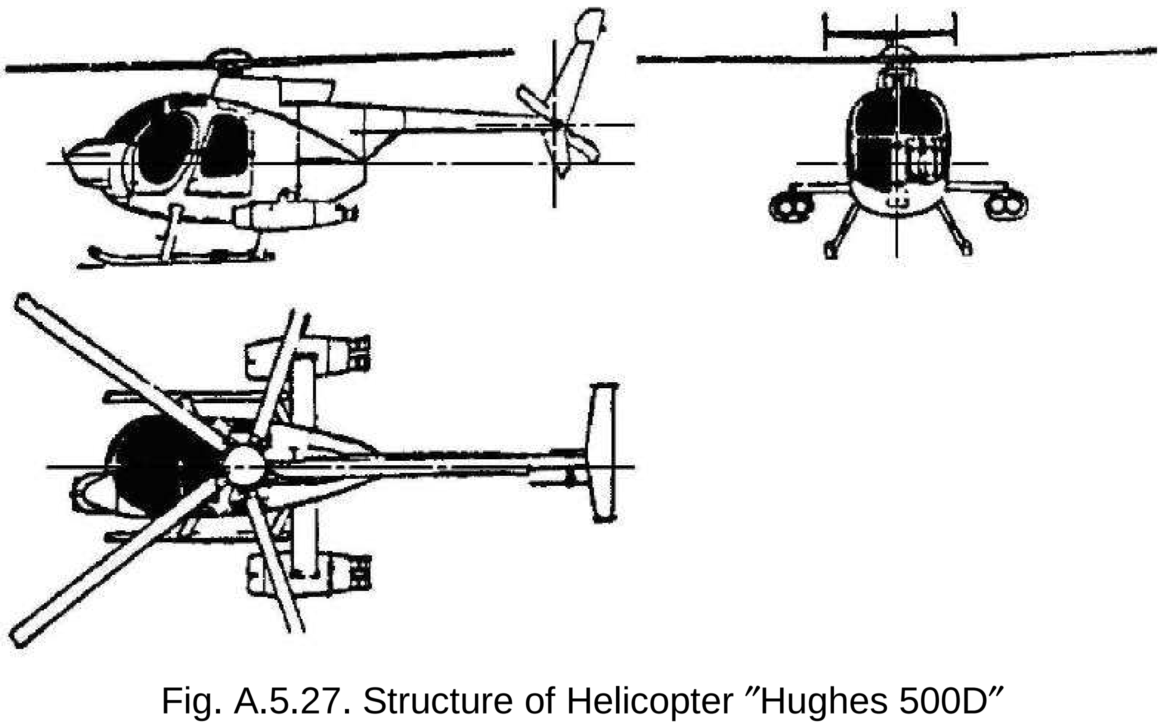


Fig. A.5.27. Structure of Helicopter ″Hughes 500D″

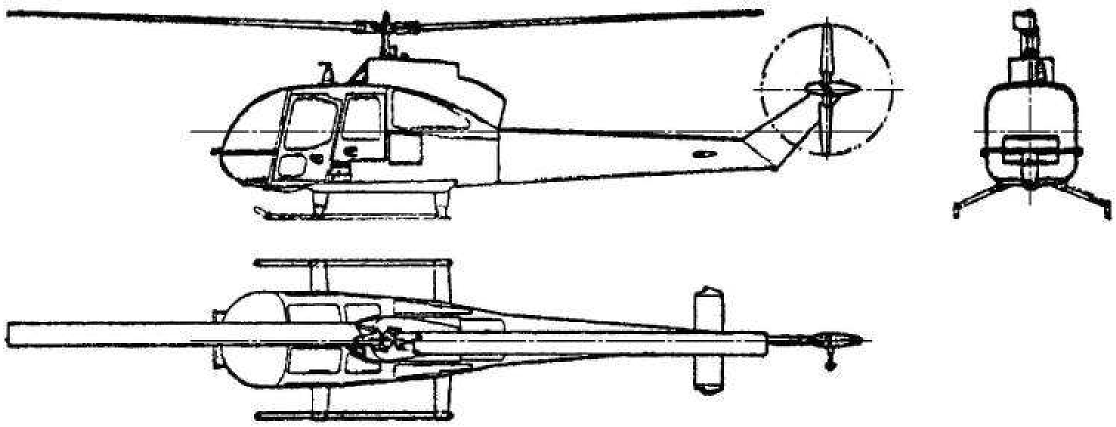


Fig. A.5.28. Structure of Helicopter ″ Rogerson Hiller RH-1100″

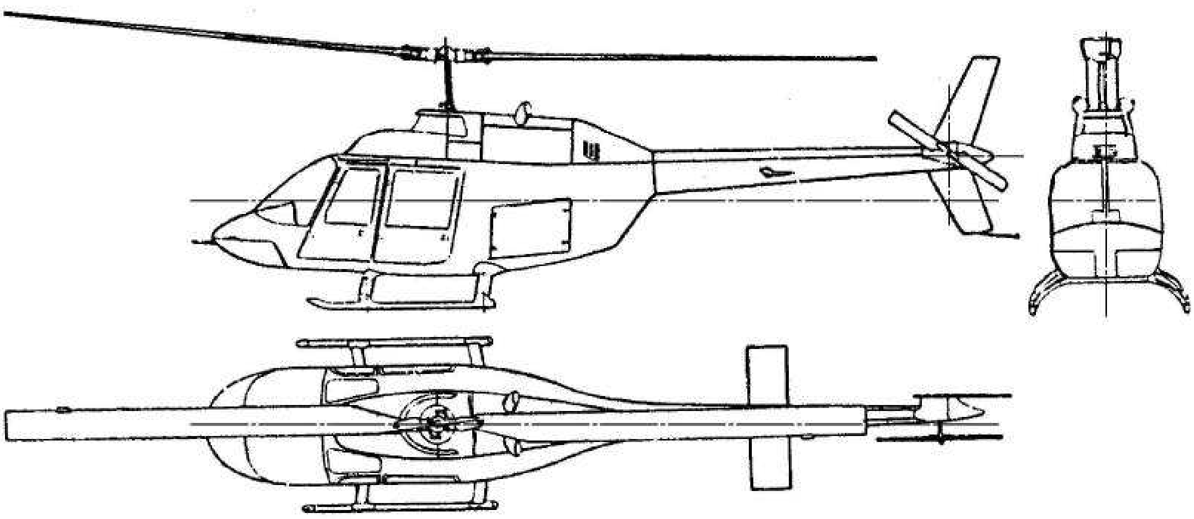


Fig. A.5.29. Structure of Helicopter ″Bell 206A Jet Ranger″

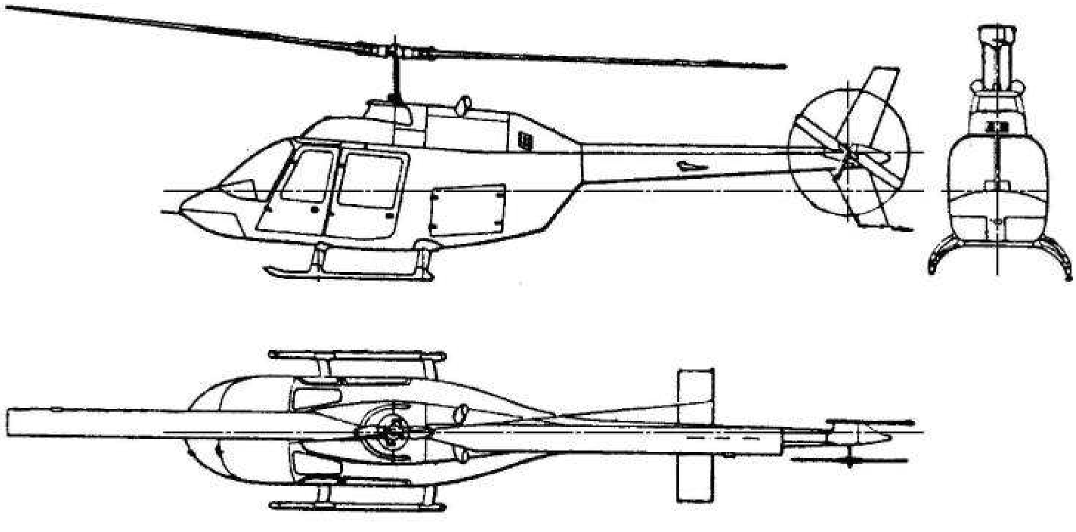


Fig. A.5.30. Structure of Helicopter ″Bell OH-58A Kiowa″

Appendix 6

LOW-POWERED PISTON AND TURBINE AIRCRAFT ENGINES. STATISTICAL DATA

Sr. No.	Engine Performance	Engines			
1	Engine type	Piston (Austria)	Piston (Austria)	Piston (Austria)	Piston (Austria)
2	Engine Trade Mark	Rotax 462 UL CB	Rotax 912 A	Rotax 535	Rotax 532UL 2V СБ
3	Helicopter power plant and engine features	- two-cylinder - two-cycle - water-cooled - intake control: by slide valve	-four-cylinder - four-cycle - mixed cooled - intake control: by valve	- two-cylinder - two-cycle - water-cooled - intake control: by slide valve	- two-cylinder - two-cycle - water-cooled - intake control: by slide valve
4	Engine weight (basic section)	26	56	35	28.4
5	Maximum power	38 (52) at 6500	59.0 (80) at 5500	44 (60) at 7200	47 (64) at 6600
6	Specific fuel consumption under maximum power	(20)	(22)	(29)	(28.4)
7	Engine specific weight	0.684	0.949	0.795	0.604
8	Theoretical/efficient pressure ratio	11.5/6.7	-/9.5	11.5/-	11.5/6.3
9	Engine rotational speed	6500	5800	7200	7200
10	Engine dimension, m	-	-	-	-

Appendix 6 (Continued)

Sr. No.	Engine Performance	Engines			
1	Engine type	Piston (Austria)	Piston (Austria)	Piston (Austria)	Piston (Austria)
2	Engine Trade Mark	Rotax 562 UL IV CB	Rotax 503UL-D.C.D.I	Rotax 503UL-D.C.	Rotax 503UL-D.C.
3	Helicopter power plant and engine features	- two-cylinder - two-cycle - water-cooled - intake control: by slide valve	- two-cylinder - two-cycle - air-cooled (free, forced) - intake control	- two-cylinder - two-cycle - air-cooled (free, forced) - intake control	- two-cylinder - two-cycle - controlled exhaust - fluid cooling - intake control
4	Engine weight (basic section)	27.4	31,.4	30.0	50.5
5	Maximum power	45.0 (60) at 6600	37.4 (49.6) at 6500	34.1 (45.6) at 6500	55.0 (73.8) at 67500
6	Specific fuel consumption under maximum power	(25)	(22)	(22)	(29)
7	Engine specific weight	0.61	0.839	0.879	0.918
8	Theoretical/efficient pressure ratio	11.5/6.27	10.8/6.2	10.8/6.2	11.5/5.75
9	Engine rotational speed	6600	6500	6800	6050
10	Engine dimension, m	-	-	(0.556 x 0.513 x x 0.574)	-

Appendix 6 (Continued)

Sr. No.	Engine Performance	Engines			
1	Engine type	Piston (Poland)	Piston (Poland)	Piston (Poland)	
2	Engine Trade Mark	PSL-3S	ASZ-621R	FYA-235B PZL	F6A-350C PZL
3	Helicopter power plant and engine features	Direct-drive aircraft engine PSL-3S has been built to meet single-row radial design, 7-cylinder, air-cooled	9-cylinder air-cooled aircraft engine ASZ-621R has been built to meet single-row radial design	4-cylinder air-cooled aircraft engine PZLF.4A-235B has been built to meet two-dimensional design	Air-cooled direct-drive carburetor unsupercharged engine has been built to meet two-dimensional design
4	Engine weight (basic section)	405	580	101	150
5	Maximum power	440 (660) at 2200 rpm	735 (1000) at 230 rad/s	93 (125) at 2800 rpm	162 (220) at 2800 rpm
6	Specific fuel consumption under maximum power	376…417	380…407	312	272
7	Engine specific weight	0.971	4/5	4/5	4/5
8	Theoretical/efficient pressure ratio	6.5/1	6.4/1	6.4/1	10.5/1
9	Engine rotational speed	-	-	-	-
10	Engine dimension, m	-	-	(0.556 x 0.513 x x 0.574)	-

Appendix 6 (Continued)

Sr. No.	Engine Performance	Engines			
1	Engine type	Piston(Poland)	Piston"LOM" Prague	ПД "LOM" Прага	Piston (Ukraine)
2	Engine Trade Mark	PZL F.2A-120C	M137A, A3	M337A, AK	"Vulcan" MP-70
3	Helicopter power plant and engine features	2-cylinder air-cooled direct drive carburetor unsupercharged engine with dual ignition system	Engine has fuel injection system, lubrication system circulated under pressure with dry crankcase. Spark-plug ignition by dual magneto. Supercharged by compressor with a drive from crankshaft and fuel injection under low pressure upstream of intake valve		- design modularity; - high engine balance due to opposite cylinder arrangement diagram
4	Engine weight (basic section)	76	141	153	45
5	Maximum power	45 (60) at 3200 rpm	134	154	50 (68) at 5900 rpm
6	Specific fuel consumption under maximum power	312	-	-	367
7	Engine specific weight	4/5	4/5	4/5	4/5
8	Theoretical/efficient pressure ratio	8.5/1	6.3/1	6.3/1	10.5 geometrical
9	Engine rotational speed	-	2400 rpm		-
10	Engine dimension, m	-	-	-	-

Appendix 6 (Continued)

Sr. No.	Engine Performance	Engines			
1	Engine type	Rotor-piston engine, СКБ ВАЗ			
2	Engine Trade Mark	ВАЗ-416	ВАЗ-4161	ВАЗ-4162	ВАЗ-426
3	Helicopter power plant and engine features	Rotor-piston internal combustion engine uses four-stroke cycle under rotor rotational motion free of valve distributor. Low vibration level, small weight and dimensional values are the main advantages of the rotor-piston engine in compare with piston engine assumed to hve equal power. RPE is specified by less oil consumption and fuel consumption to be at 20% less in compare with two-cycle engines. RPE waste gas toxic problem is solved by application of catalyst or thermal type neutrolizers. Liquid type closed cooling system is used with "wet" and "dry" crankcase			
4	Engine weight (basic section)	125 (with reducer)	125 (with reducer)	125	145
5	Maximum power	110.3 (150)	132.4 (180)	147.1 (200)	154.4 (210)
6	Specific fuel consumption under maximum power	205	205	205	205
7	Engine specific weight	4/5	4/5	4/5	4/5
8	Theoretical/efficient pressure ratio	-	-	-	-
9	Engine rotational speed	1900…2800^x	1900…2800^x	1900…2800^x	1900…2800^x
10	Engine dimension, m	(0.835x0.6x0.6)	(0.835x0.6x0.6)	(0.835x0.6x0.6)	(0.835x0.6x0.6)

General Views and Some Characteristics of Engines

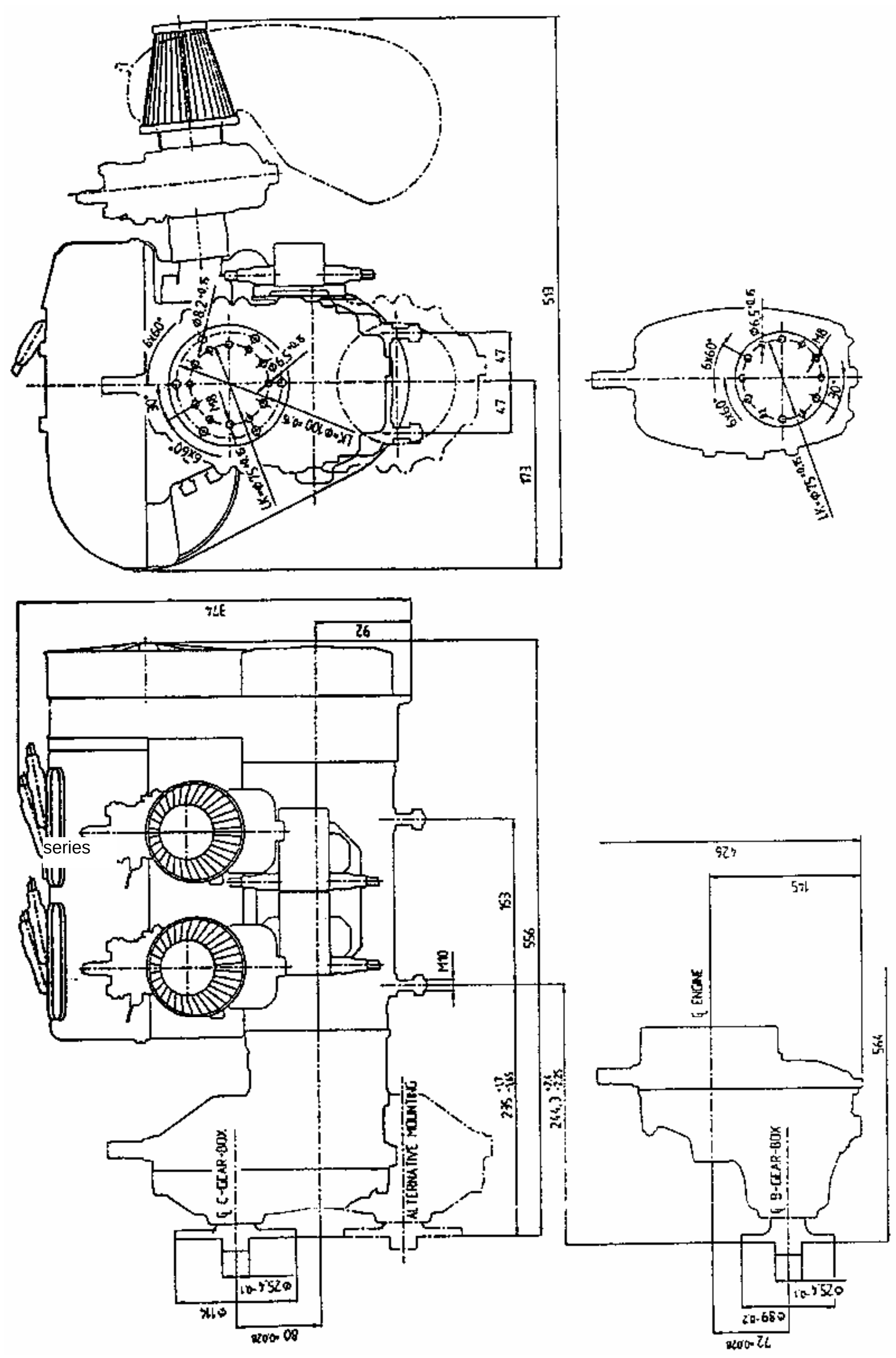
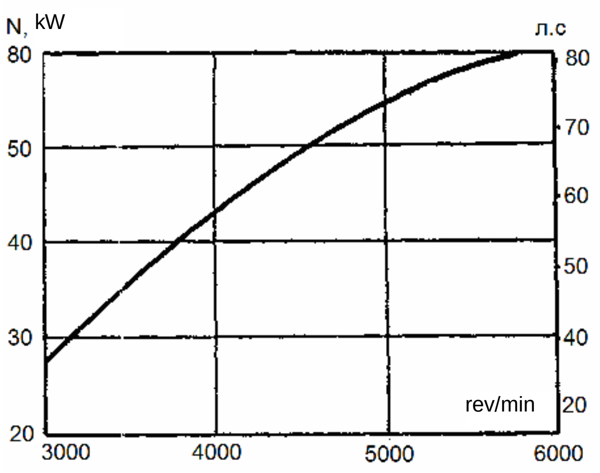
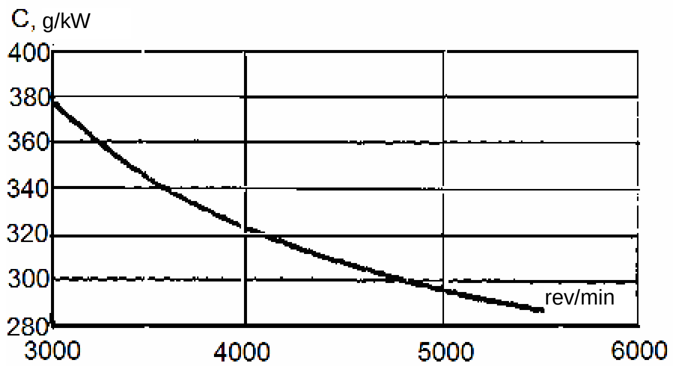


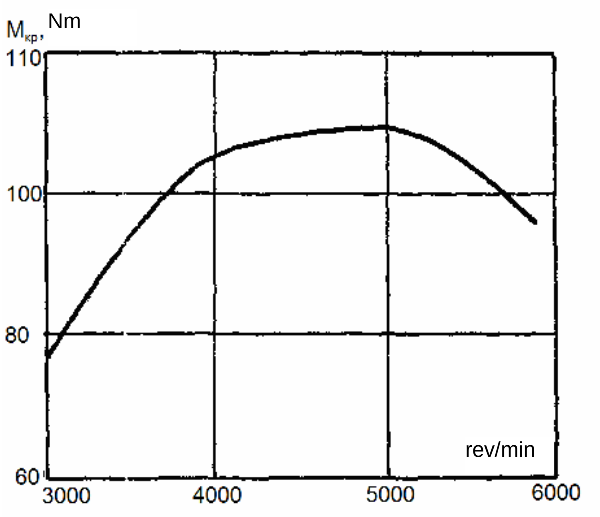
Fig. A.7.1. General View of Engine 503UL-D.C.D.I



a



c



b

Fig. A.7.2. Dependence of Power (a), Torque (b) and Fuel Consumption (c) in Relation to R-912 Engine Number of Revolutions (Rotax)

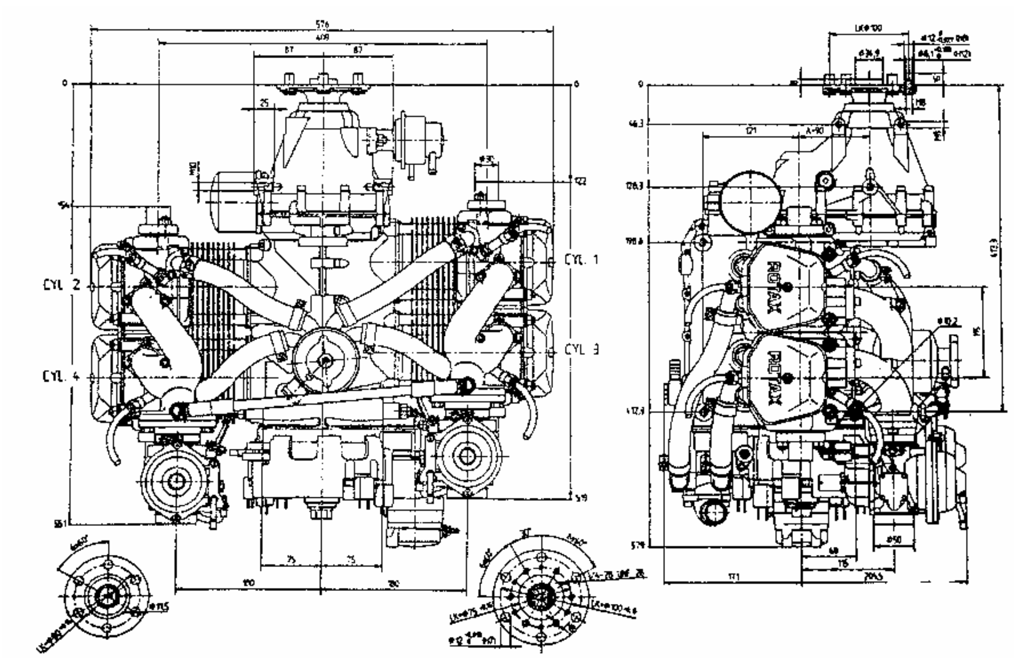


Fig. A.7.3. R-912 Engine Overall View (Rotax)

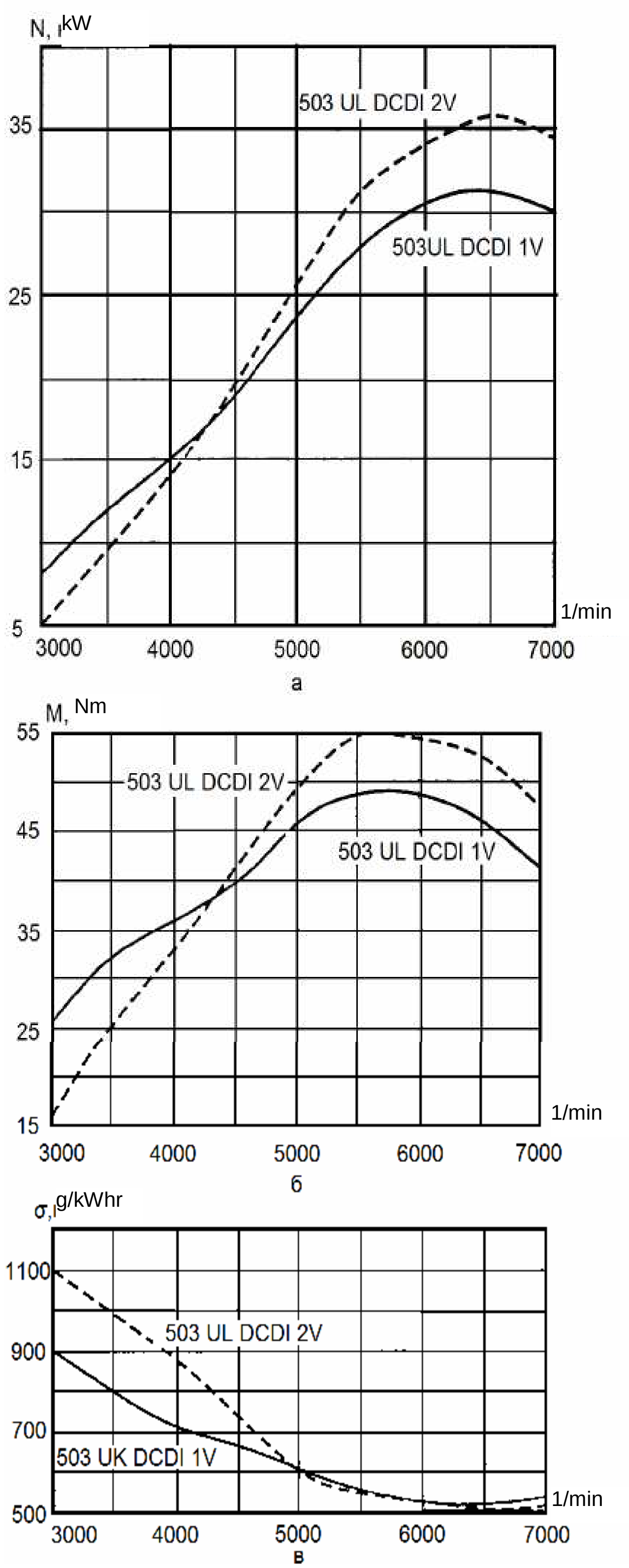


Fig. A.7.4. Dependence of Power (a), Torque (b) And Fuel Consumption (c) in Relation to 503UL-D.C.D.I Engine Number of Revolutions

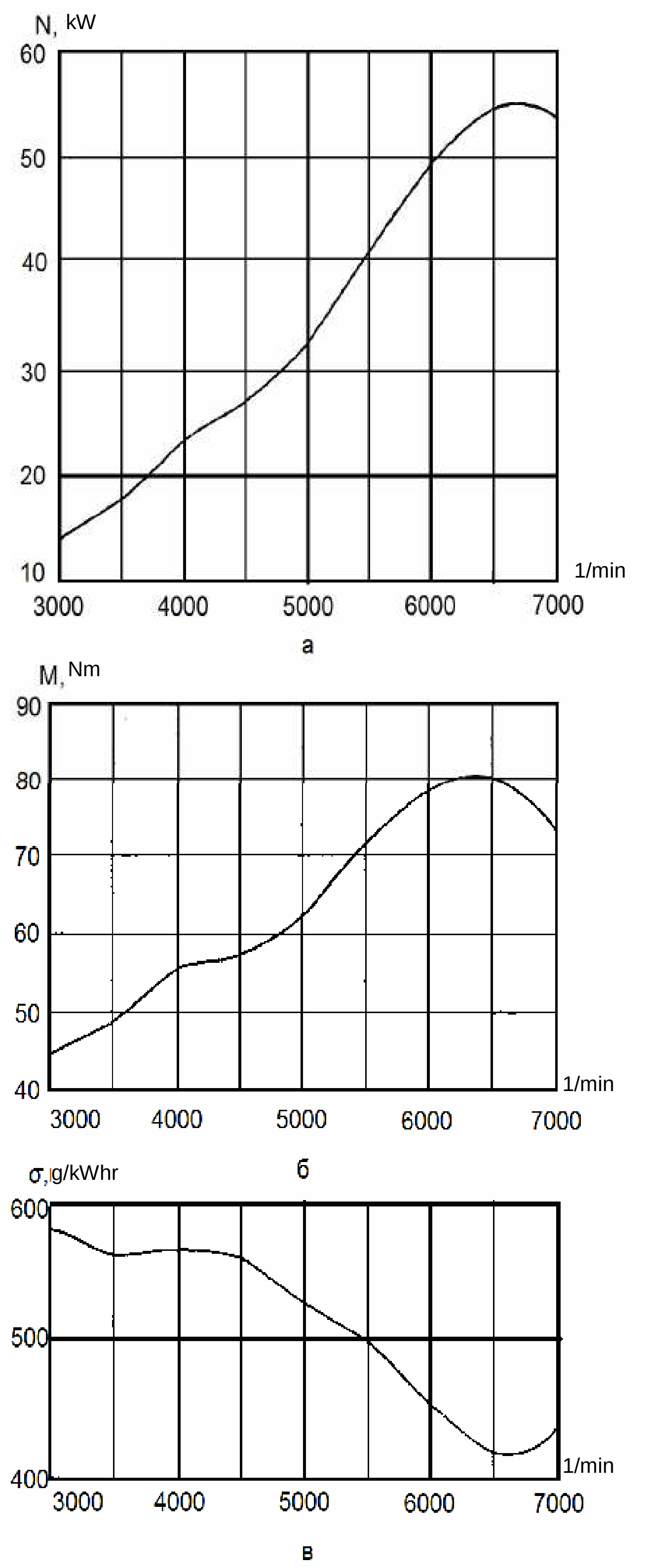


Fig. A.7.5. Dependence of Power (a), Torque (b) And Fuel Consumption (c) in Relation to 618UL-D.C.D.I Engine Number of Revolutions

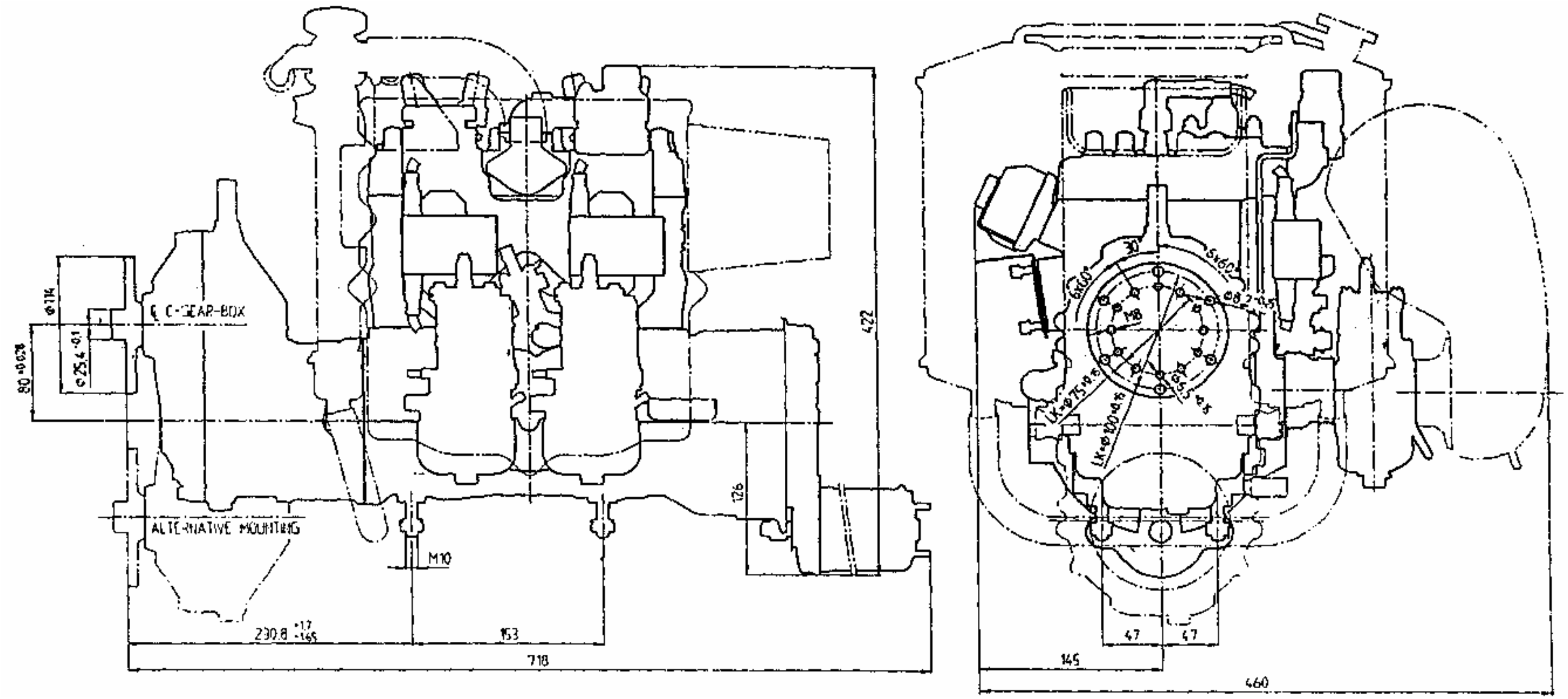


Fig. A.7.6. 618UL-D.C.D.I Engine General View

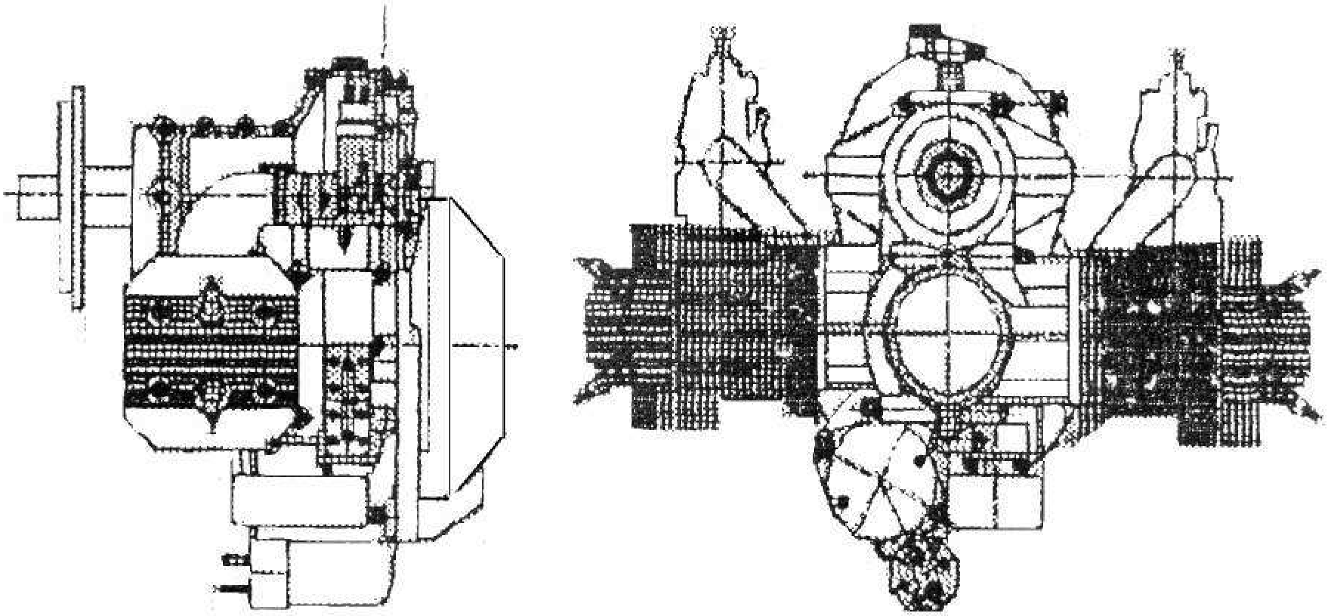


Fig. A.7.7. MP-70 «Volcano» Engine General View

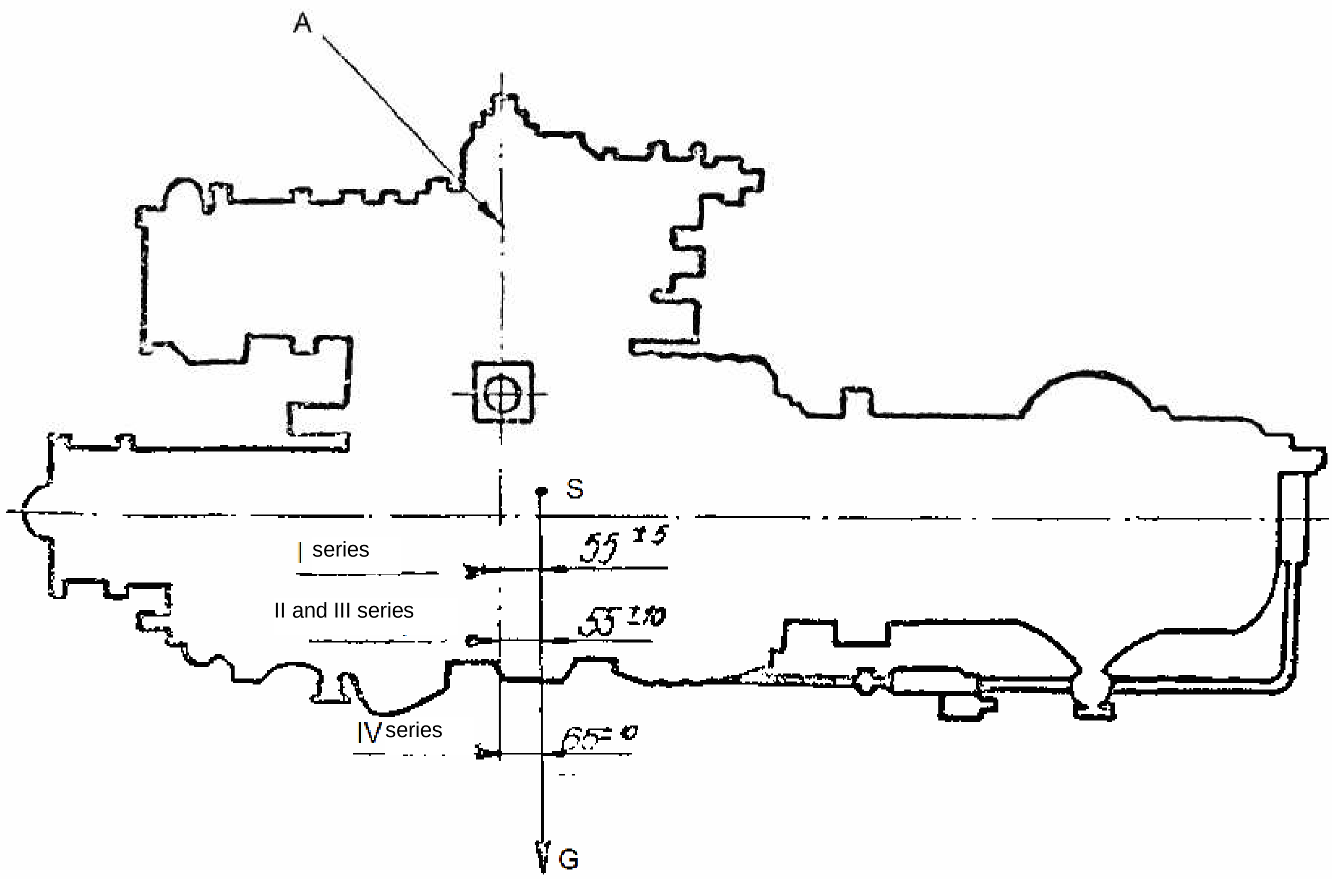


Fig. A.7.8. CG Position ***S*** of ГТД-350 Engine Relatively to Mount Plane A (View From Left Hand)

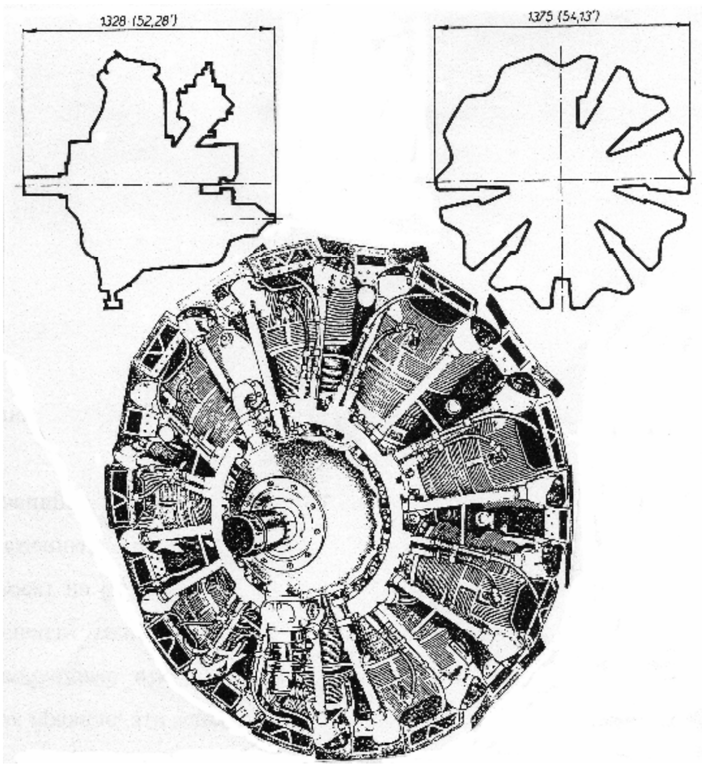
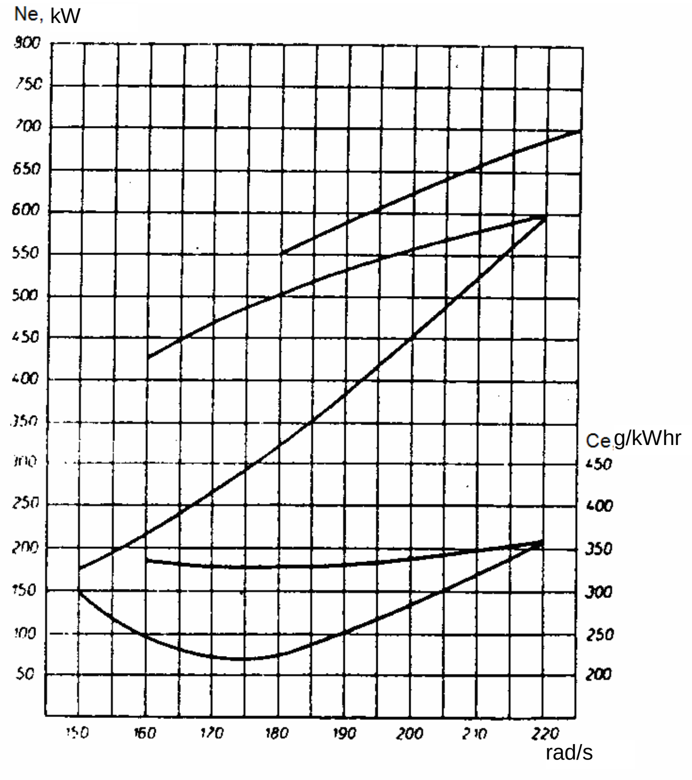
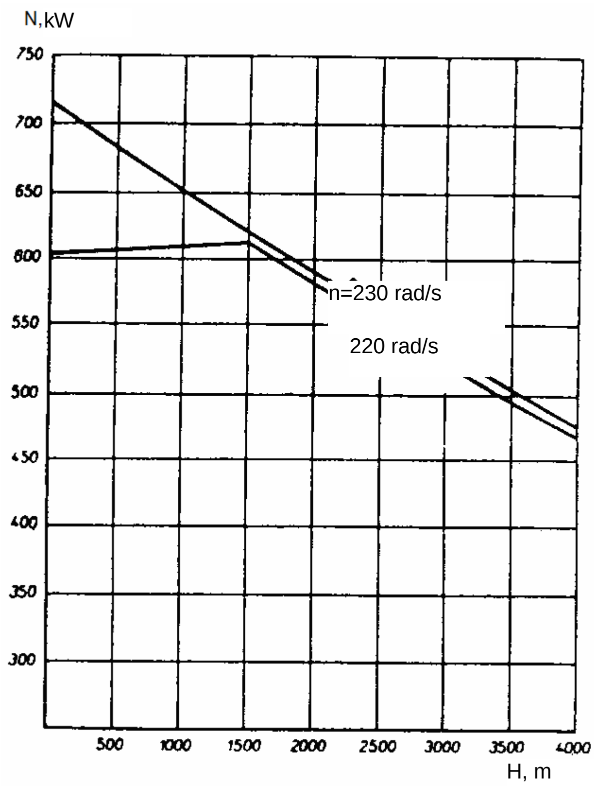


Fig. A.7.9. Overall Dimensions and View of AS_z-621R Engine



a



b

Fig. A.7.10. Dependence of Power, Fuel Consumption in Relation to Number of Revolutions (a) and Power in Relation to Flight Altitude (b) of AS_z-621R Engine

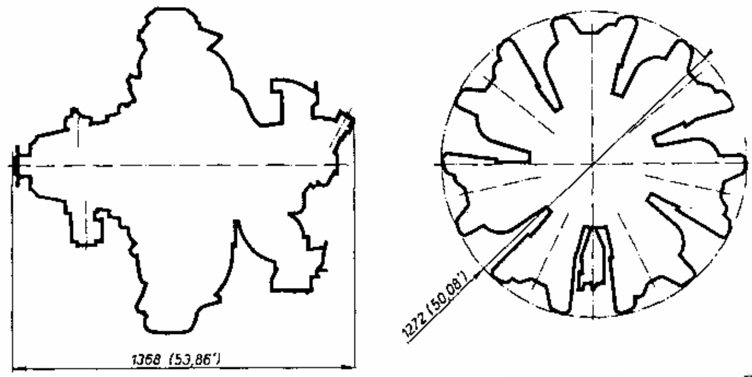
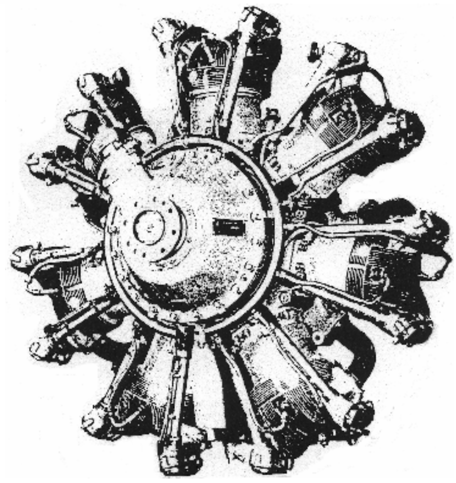
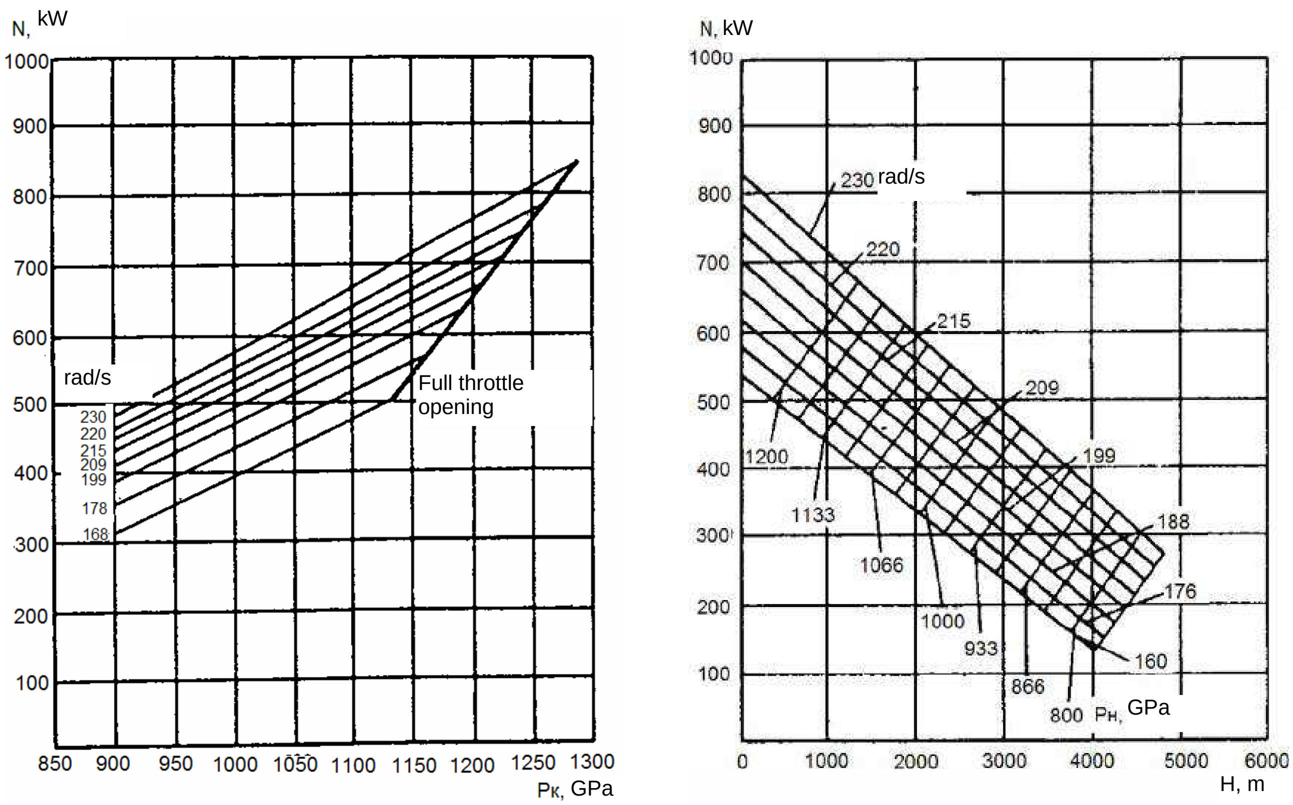


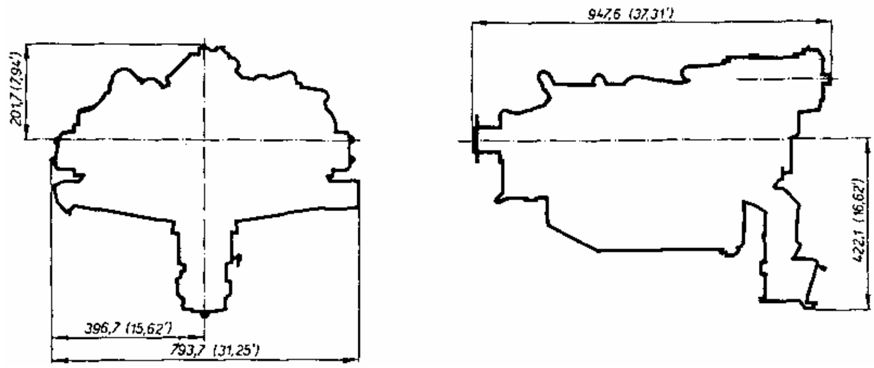
Fig. A.7.11. Overall Dimensions and View of PZL-3S Engine





а б

Fig. A.7.12. Dependence of Power on Pressure $P_к$(a) and Power on Flight Altitude (b) and RPM of PZL-3S Engine



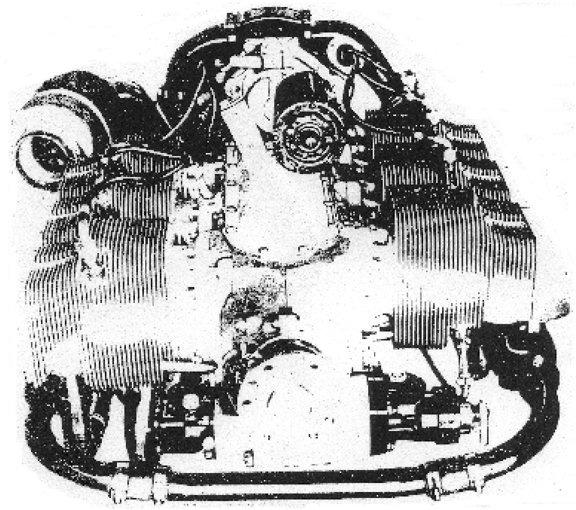


Fig. A.7.13. Overall Dimensions and View of PZL F.6A-350C Engine

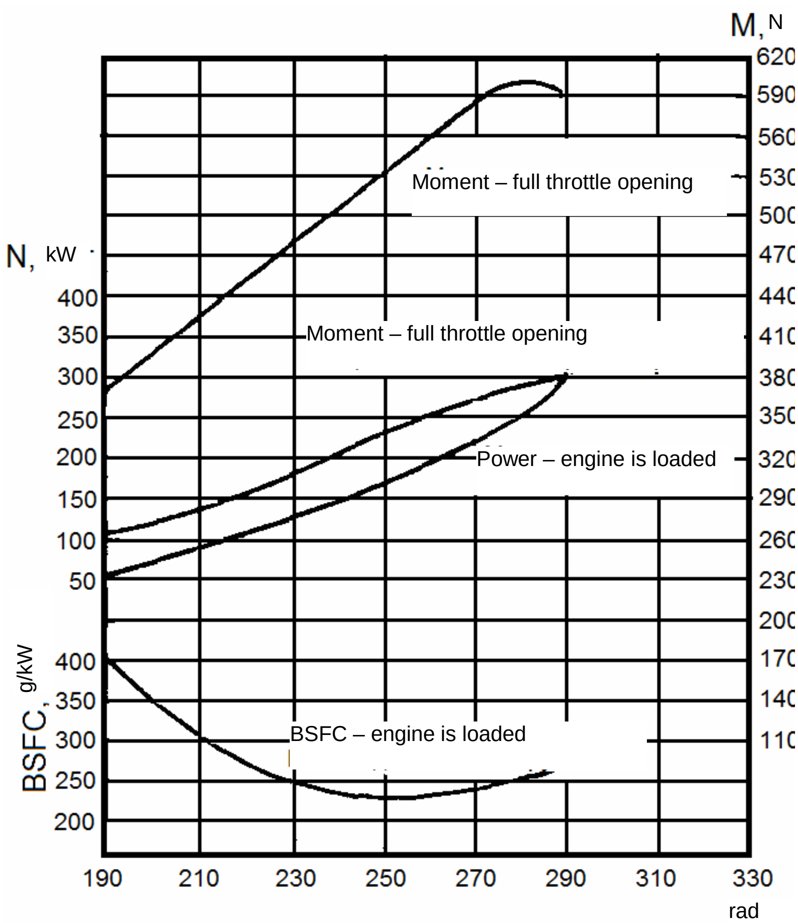


Fig. A.7.14. Dependence of Power, Torque and Fuel Consumption of RPM of PZL F.6A-350C Engine

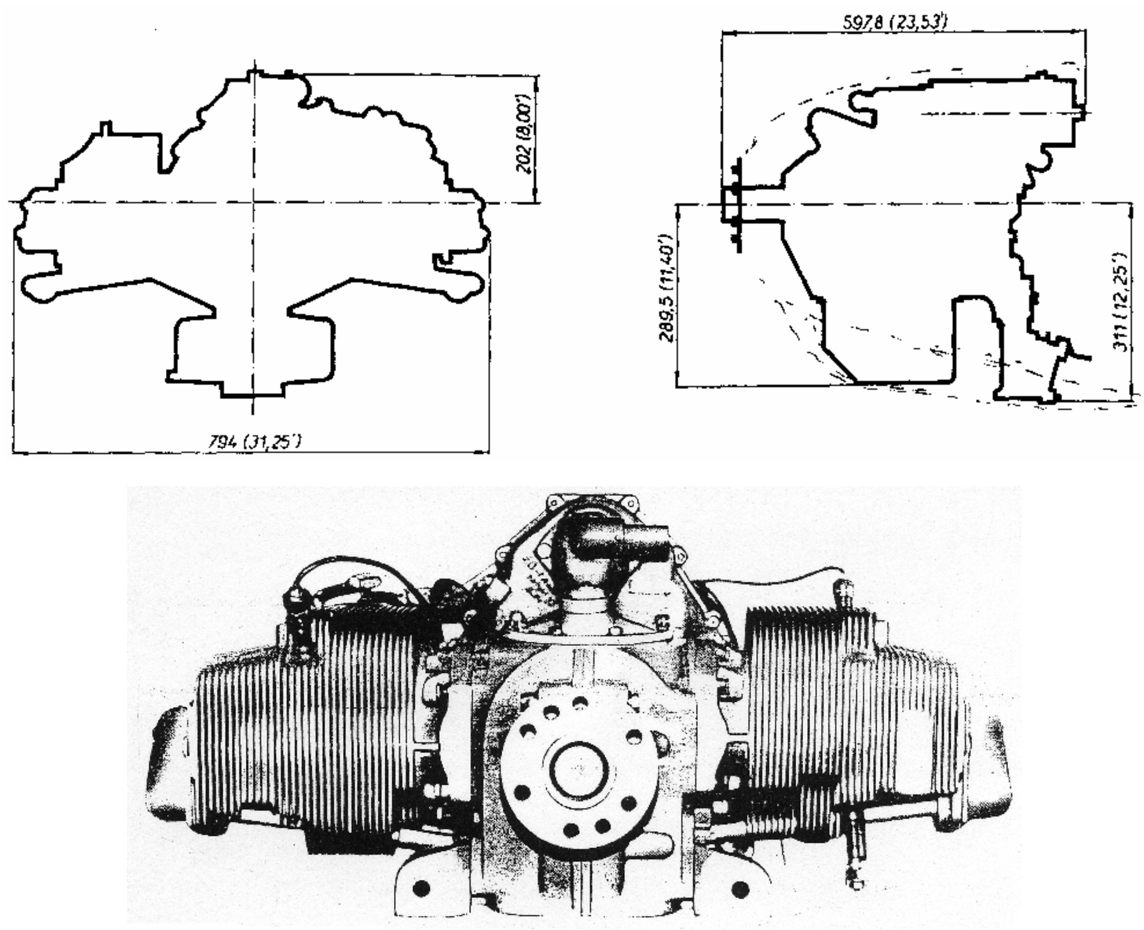


Fig. A.7.15. Overall Dimensions and View of PZL F.2A-120C Engine

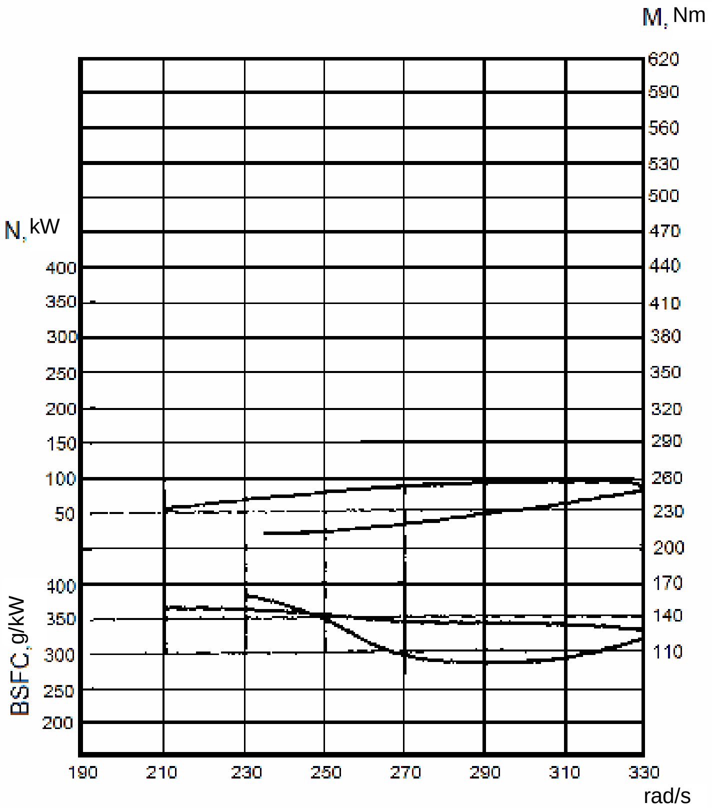


Fig. A.7.16. Dependence of Power, Torque and Fuel Consumption of RPM of PZL F.2A-120C Engine

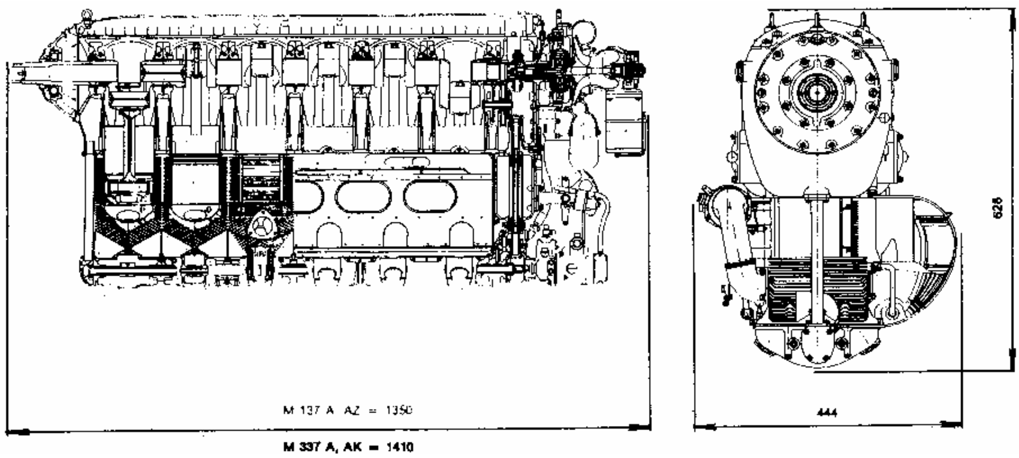


Fig. A.7.17. Dimensions and Variants of M 137 A, M 337 A Engines:

M 137 A – Engine Basic Variant M 337 Without Turbo Charging (for Acrobatics);
M 137 A3 – Modification M 137 A, with Air Filter;
M 337 A – Engine with Turbo Charging for Basic Acrobatics, Forbidden Flights at Back;
M 337 Aк – Similar M 337 A, Enhanced Oil System, Allows Acrobatics at Back

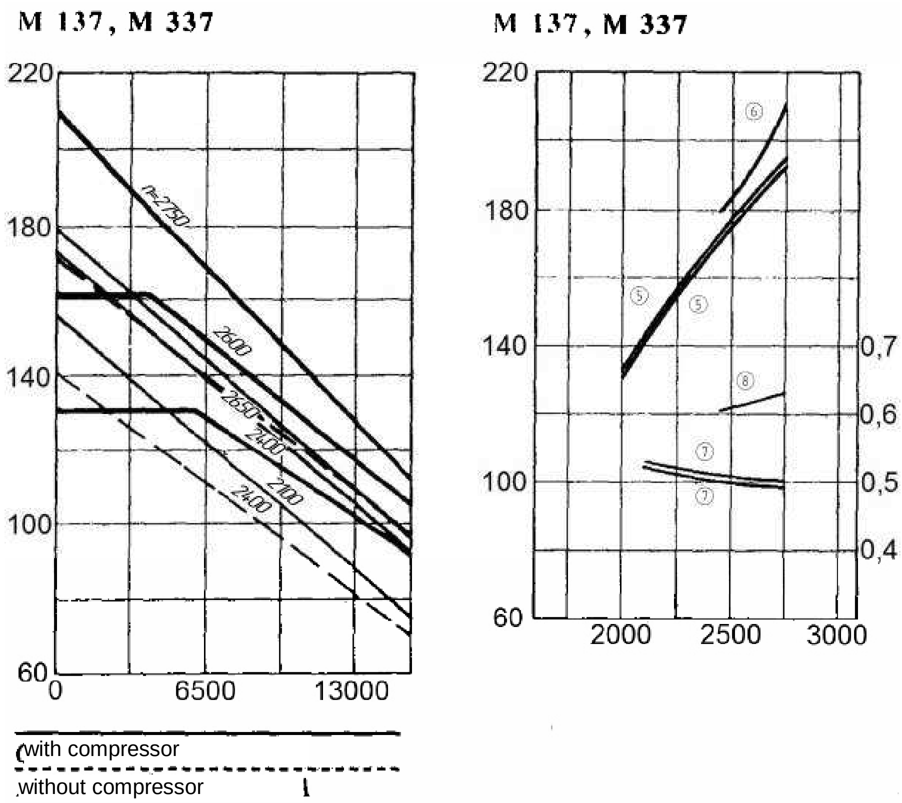


Fig. A.7.18. Engine Characteristics:

1 – power, *kW;* 2 – fuel specific consumption, g*/kW·h;* 3 – number of revolutions, *RPM;* 4 – altitude, *m;* 5 – effective power without compressor, *kW;* 6 – effective power with compressor, *kW;* 7 – fuel specific consumption without compressor, g*/kW·h;* 8 – fuel specific consumption with compressor, g*/kW·h*

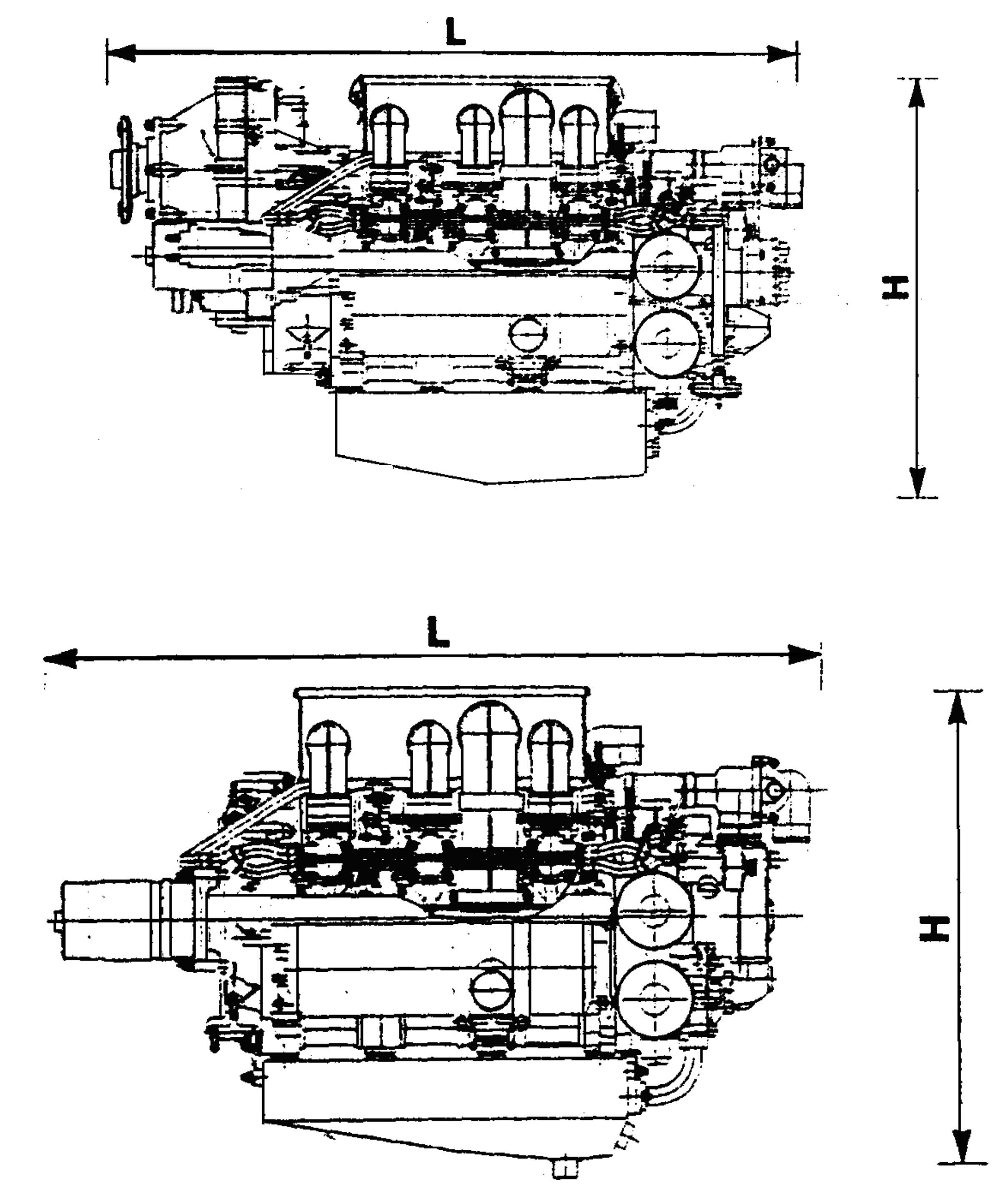


Fig. A.7.19. Engine Overall Dimensions

Aviation Rotary Engines Designed According to Requirements of Aviation Regulations АП-33 (Corresponding to FAR-33)

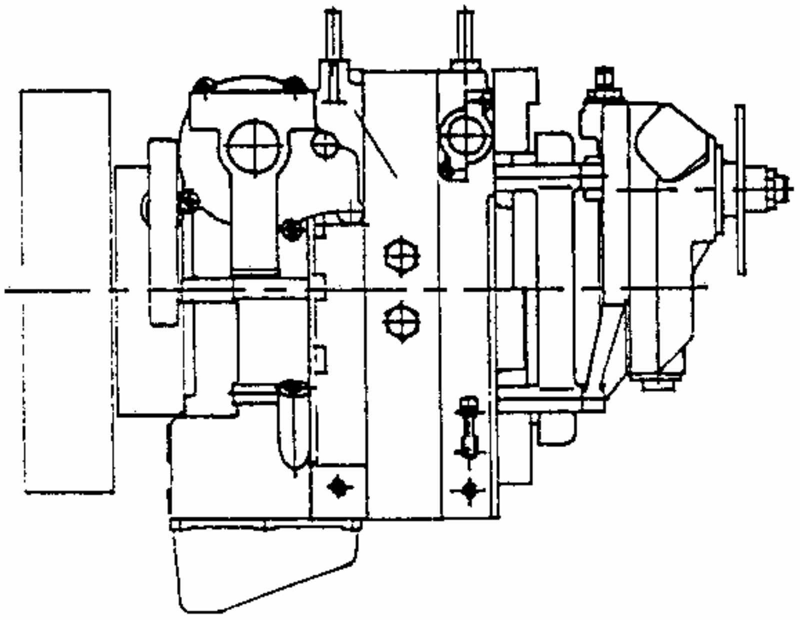


Fig. A.7.20. Overall View of BA3-1187 Engine

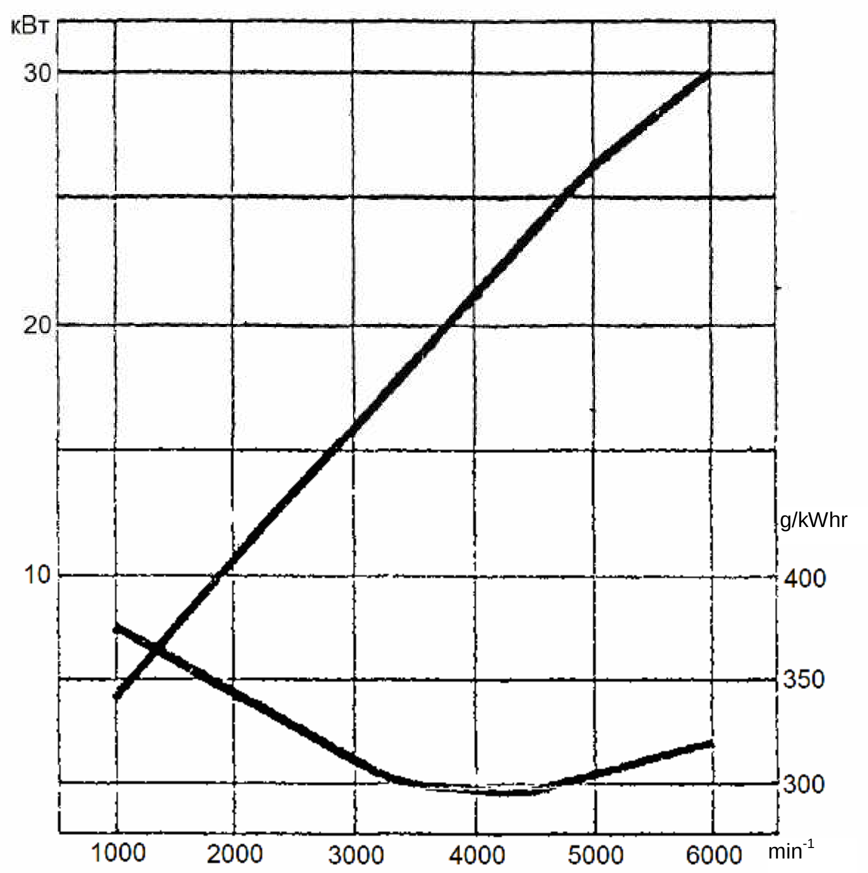


Fig. A.7.21. Dependence of Power and Fuel Consumption in Relation to RPM of BA3-1187 Engine

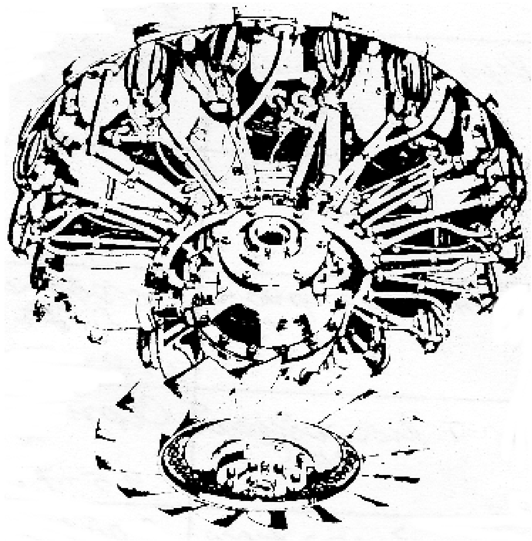


Fig. A.7.22. Overall View of M14 B26 Engine

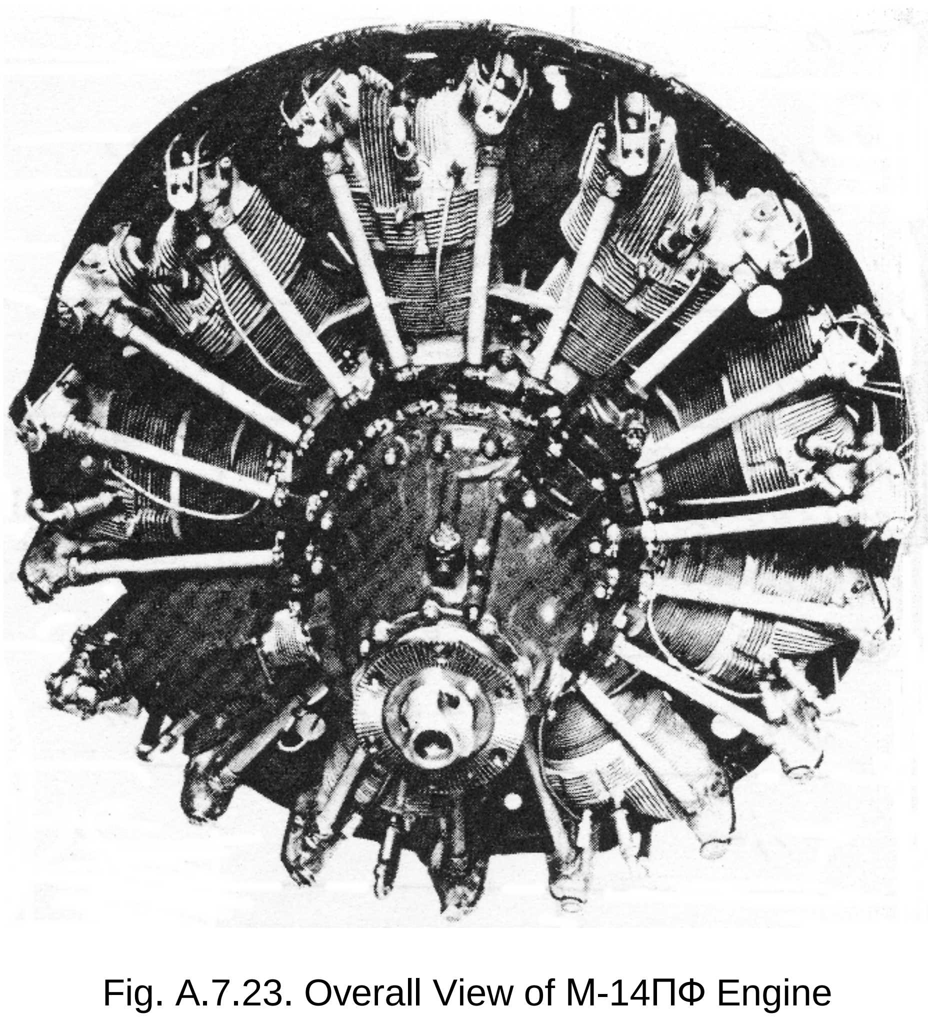


Fig. A.7.23. Overall View of M-14ПФ Engine

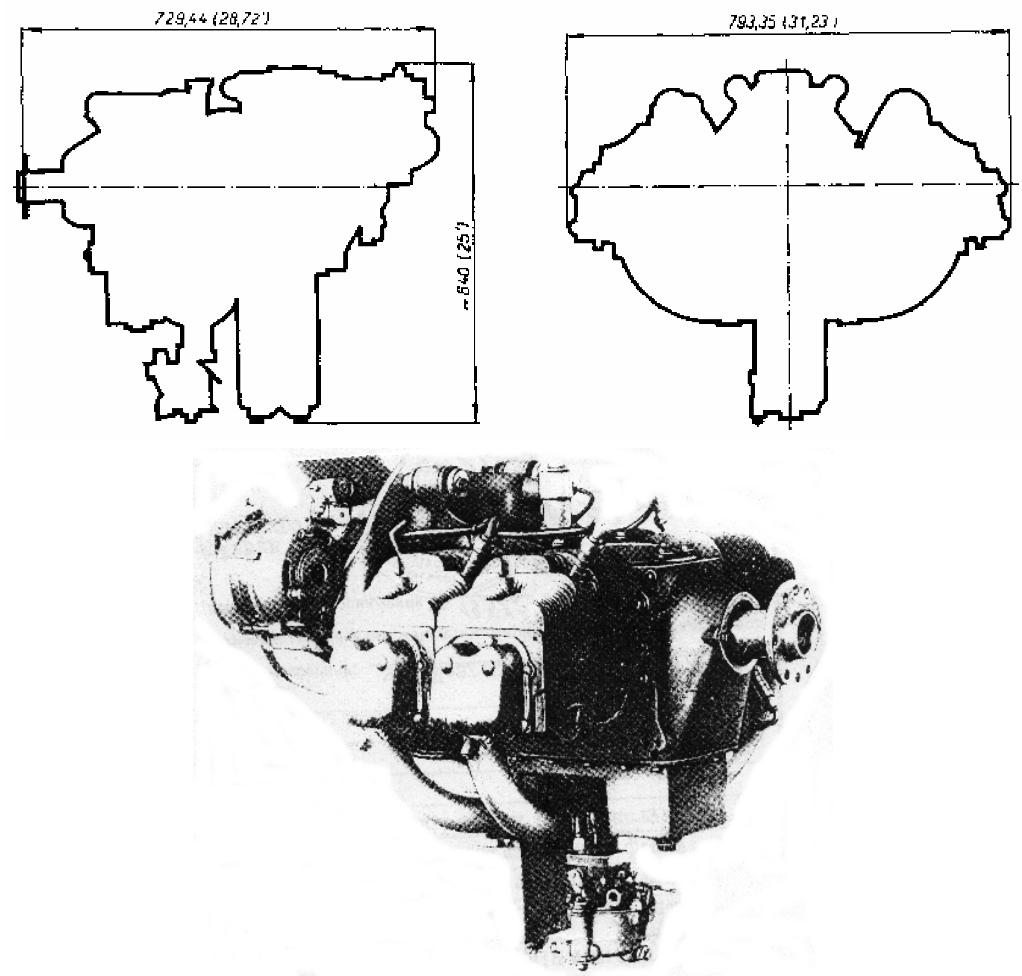
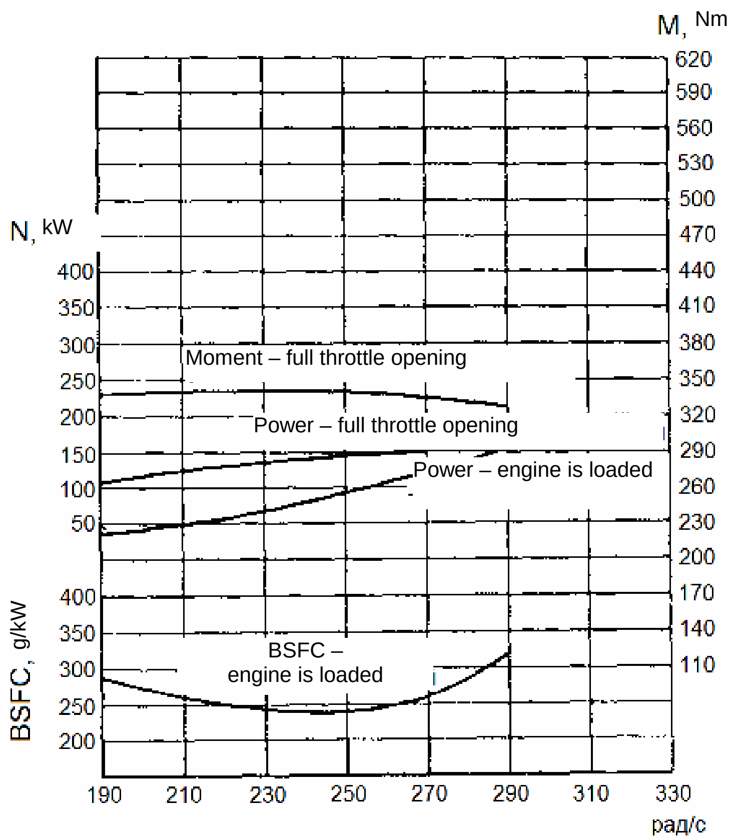


Fig. A.7.24. Overall View and Dimensions of PZL F.4A-235B Engine

Fig. A.7.25. Dependence of Power, Torque and Fuel Consumption of RPM of PZL F.4A-235B Engine



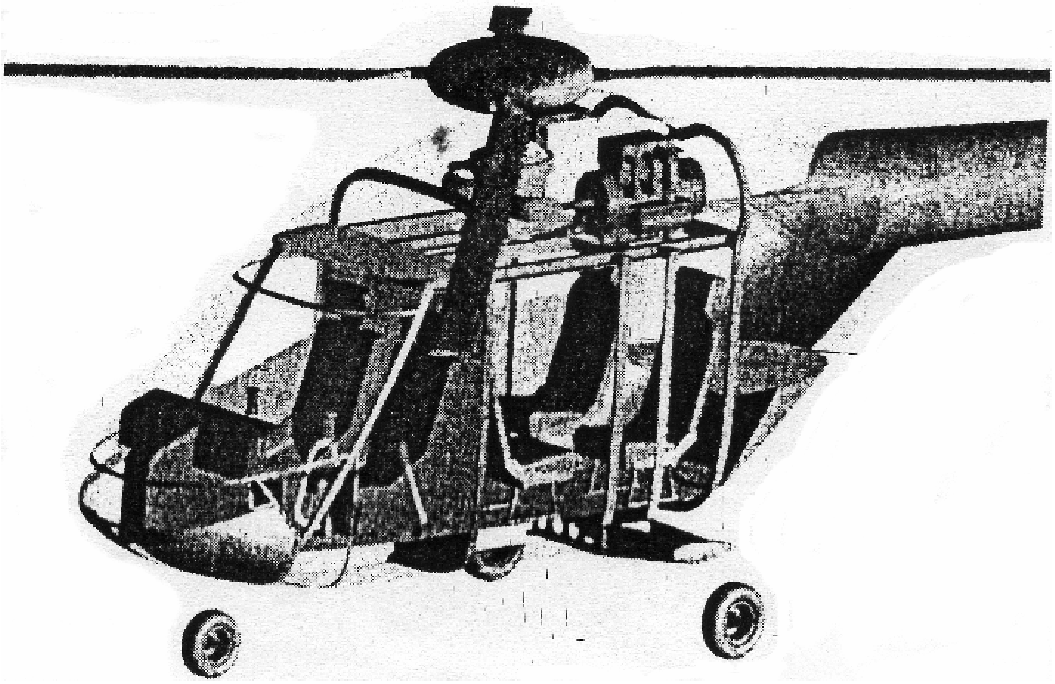


Fig. A.7.26. LMH Layout with 4265 Engines

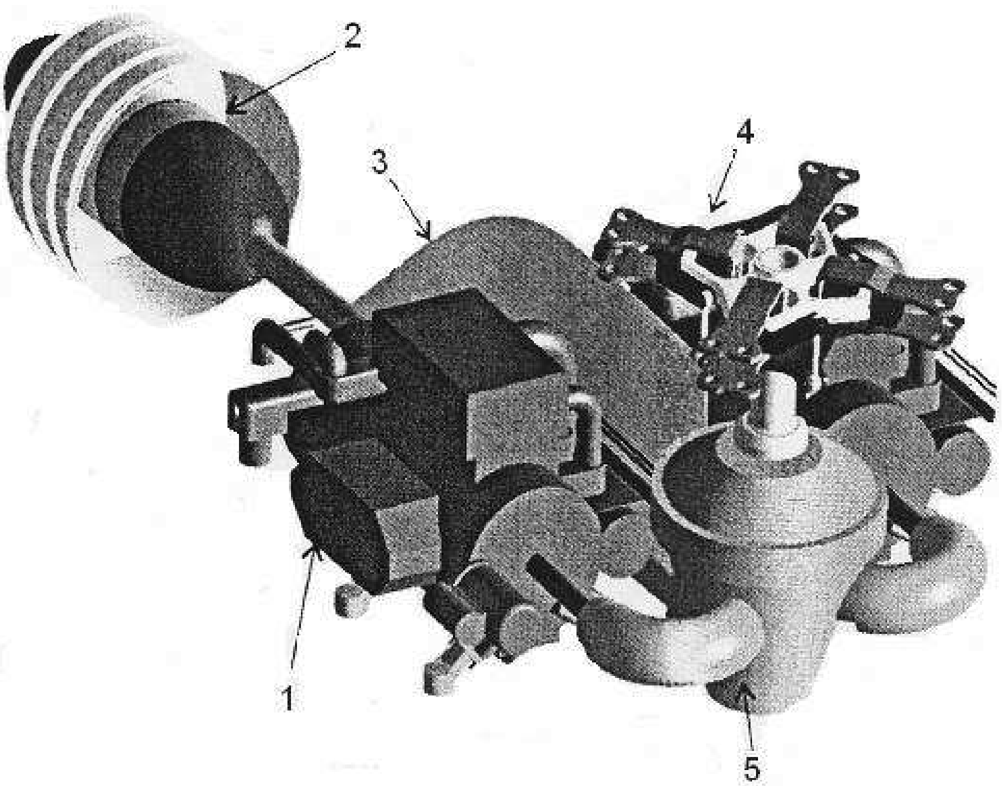


Fig. A.7.27. LMH Power Plant [9]:
1 – engine BA3 4265; 2 – fluid power system fan;
3 – fire partition; 4 – main rotor hub;
5 – main reducer

Helicopter characteristics

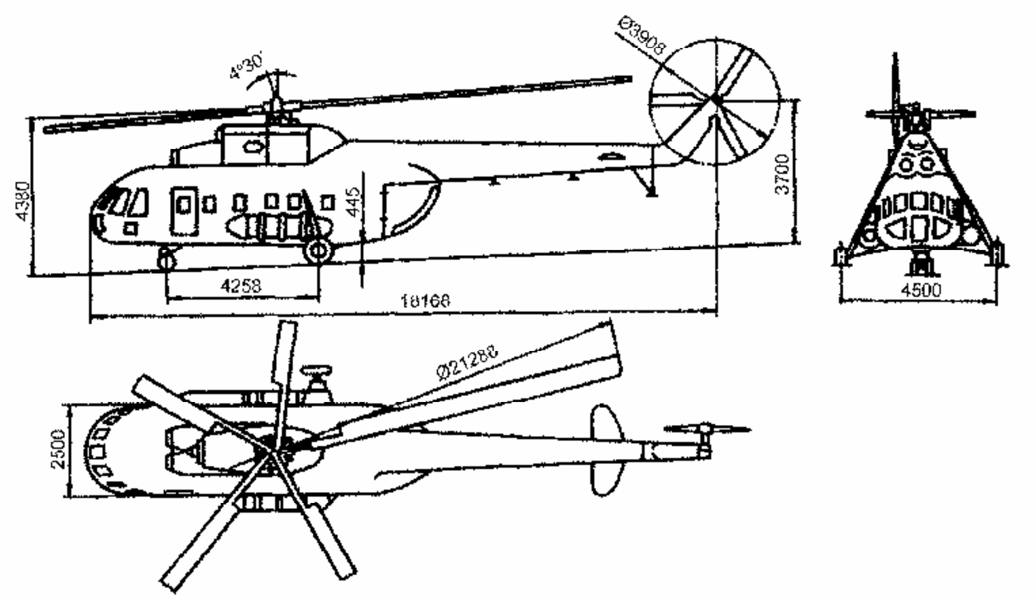


Fig. A.8.1. Mi-8 Helicopter Diagram

Mi-8 HELICOPTER CHARACTERISTICS

Sizes, m:	
helicopter length, rotors turning	25.24
fuselage length without tail rotor	18.17
Fuselage width	2.5
helicopter height:	
to main rotor hub	4.38
with tail rotor turning	5.65
main rotor diameter	21.29
passenger compartment (Mi-8P):	
length	6.36
width	2.34
height	1.8
cargo compartment (Mi-8T):	
length	5.34
width	2.34
height	1.8
Engines:	
for Mi-8T	2 GTE TB2-117A
takeoff capacity, kW/h.p.	2x1250/2x1700
for Mi-8TM, AMT, MTV	2 GTE TB3-117BM
takeoff capacity, kW/h.p.	2x1435/2x1950 (in conditions of MCA to H = 3.6 km)
for Mi-8TG	2 ATRE GV2-117TG
takeoff capacity, kWt/h.p.	2x1104/2x1500

Masses and loads, kg:

(civil Mi-8T)	
maximum takeoff mass	12000
Mi-8AMT and MTV	13000
average takeoff mass	11100
standard mass of empty helicopter	6625
Mi-8P	6800
maximum payload:	
in fuselage	4000
on external attachment	3000
full fuel load, l:	
in main fuel tank	1450
in main and two extra fuel tanks	2870

Performance (civil Mi-8T):

maximum speed near ground:	
under normal takeoff mass, km/h	250
under maximum takeoff mass, km/h	230
cruising speed under normal takeoff mass, km/h	225
hovering ceiling, m	1000
hovering ceiling under normal takeoff mass, m:	
in ground effect	1800
out of ground effect	850
for Mi-8AMT and MTB	3980
dynamic ceiling:	
under normal takeoff mass, m	4500
for Mi-8AMT and MTV, m	6000
flight range with 5% of fuel reserve, km:	
under normal takeoff mass	480
under maximum takeoff mass	460
range with 28 passengers with fuel reserve for 20 min of flight	425
range (Mi-8AMT and MTV) under maximum takeoff mass with payload of 3400 kg with fuel tanks full	580

Mi-26T HELICOPTER CHARACTERISTICS

Sizes, m:	
helicopter length, rotors turning	40.03
helicopter width	
(including external landing gear wheels)	8.15
main rotor diameter	32
propeller disc area, m^2	810
Engines:	2 GTE E-126
	MGB “Progress”, Zaporojie
takeoff capacity in conditions of MCA	
to H=1500 m, kW/h.p.	2x7355/2x10 000
Masses and loads, kg:	
maximum takeoff mass	56000
average takeoff mass	49600
empty helicopter mass	28200
maximum infuselage	
payload	20000
external load:	
on short sling	18150
on sling in length of 30,5 m during flight	
on height of 1000 m under temperature of +30°C	
with fuel reserve for one hour flight	
with reserve for more 30 min	14900
internal fuel, l	12000
Performance:	
maximum speed of horizontal flight, km/h	295
cruising speed under average mass, km/h	255
flight range under maximum takeoff mass	
with full fuel tanking with en-route fuel reserve for 0.5h	
under load of 18000 kg, km	670
with four auxiliary tanks, km	
hovering ceiling out of ground effect	2000
under average takeoff mass	
under conditions of ISA, m	1800
dynamic ceiling, m	4600

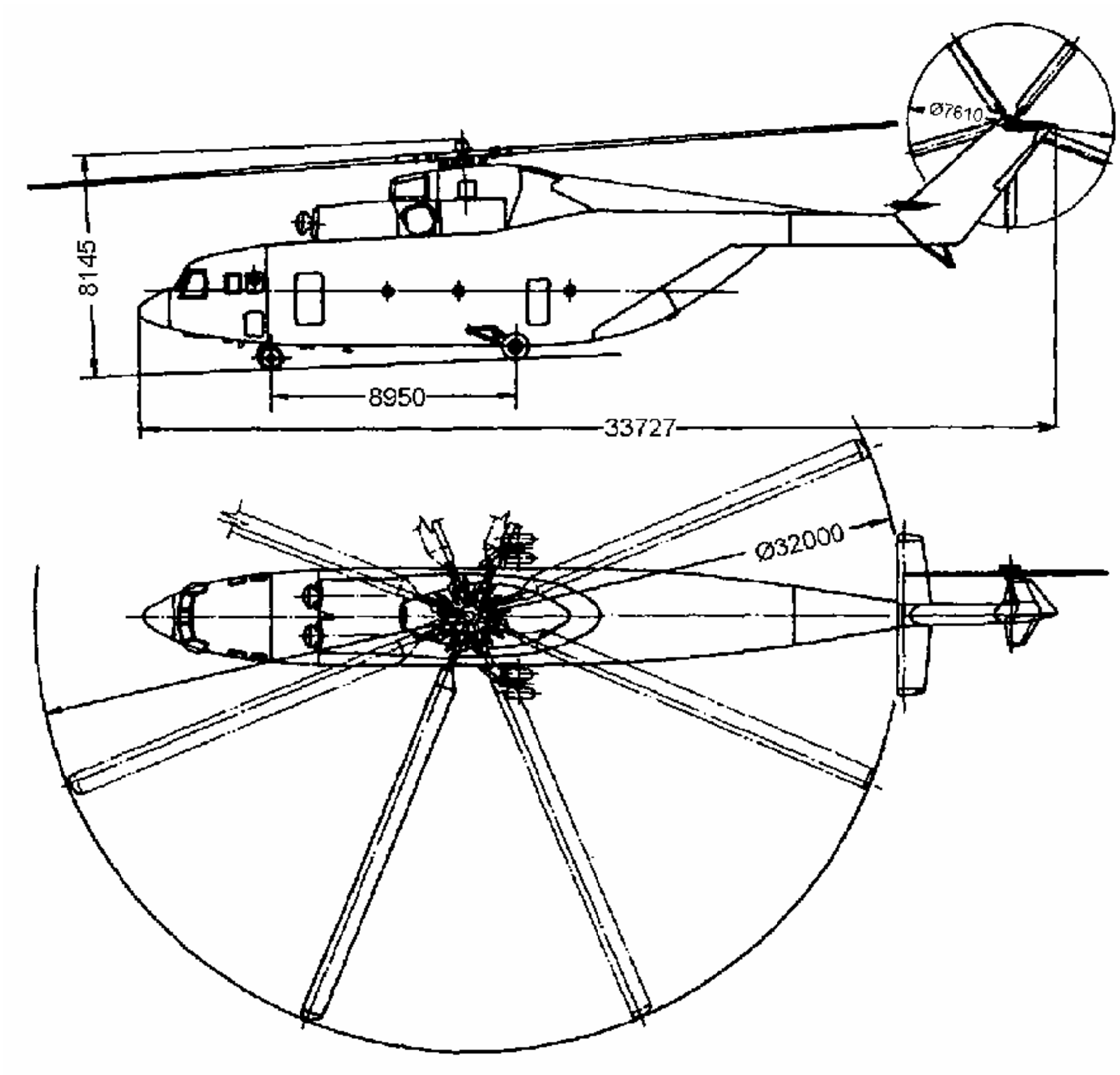


Fig A.8.2. Mi-26T Helicopter Diagram

Mi-34 HELICOPTER CHARACTERISTICS

Sizes, m:	
helicopter length, rotors turning	11.48
fuselage length	8.71
fuselage width	1.42
main rotor diameter	10
propeller disc area, m^2	78.5
Engines:	
Mi-34	PDM-14B26
takeoff capacity, kW/h.p.	242/325
Mi-34 VAZ	2 PD VAZ-40
takeoff capacity, kW/h.p.	2x162/2x220
Masses and loads, kg:	
maximum takeoff mass	1350
average takeoff mass	1260
during sporting contests	1020
empty helicopter mass	800
maximum payload	240
fuel mass	120
Performance:	
maximum speed, km/h	210
maximum cruising speed, km/h	180
average speed, km/h	160
flight range under maximum takeoff mass, full fuel tanking with en-route fuel reserve for 0.5h, with load of 135 kg, km	305
hovering ceiling under average takeoff mass under conditions of ISA, m	700
dynamic ceiling, m	4500
maximum operating overload under takeoff mass of 1020 kg within the speed range of 50…150 km/h,	from+2.5 to –0.5
maximum flight speed on way back under takeoff mass of 1020 kg, km/h	130

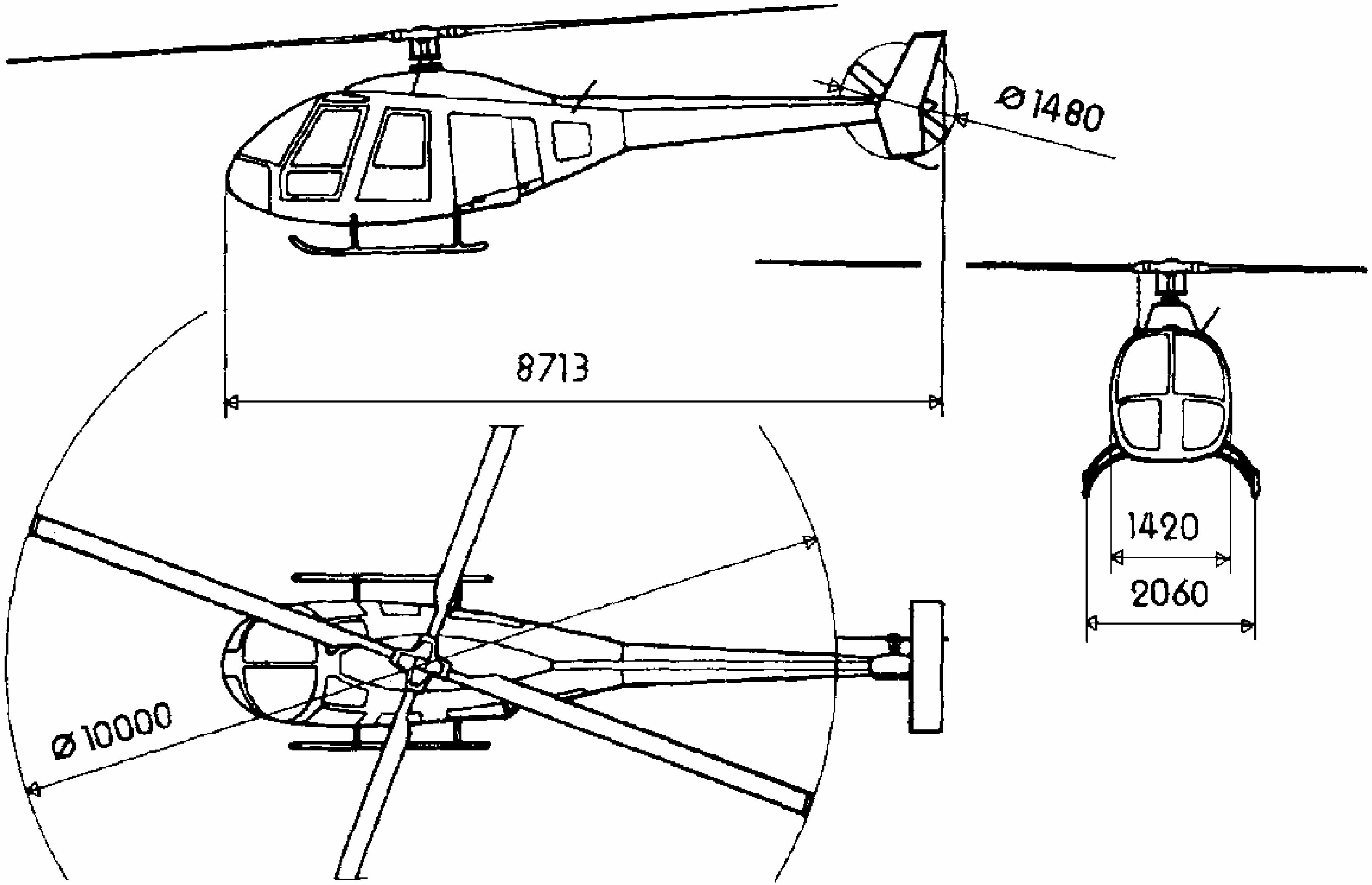


Fig. A.8.3. Mi-34 Helicopter Diagram

Mi-28 HELICOPTER CHARACTERISTICS

Sizes, m:	
helicopter length	
with rotors turning	21.6
without rotors	17.91
wing span	4.88
helicopter height	
with rotors	4.7
till main rotor hub	3.82
main rotor diameter	17.2
propeller disc area, m2	232
Engines:	2 GTE TV3-117VM St. Petersburg SPA in name of V.Y. Klimov
takeoff capacity, kW/h.p.	2x1620/2x2200
Masses and loads, kg:	
maximum takeoff mass	11200
average takeoff mass	10400
empty helicopter mass	7000
maximum combat load	
(with sighting system and weapon system)	3640
internal fuel, l	1337
Performance:	
maximum speed, km/h	300
maximum cruising speed, km/h	270
hovering ceiling, м	3500
dynamic ceiling, m	5800
maximum rate of climb, m/s	13.6
flight range with maximum	
with fuel reserve, km	460
time in flight with maximum	
fuel reserve, h	2
maximum limit load factor	from +3.0 to –0.5

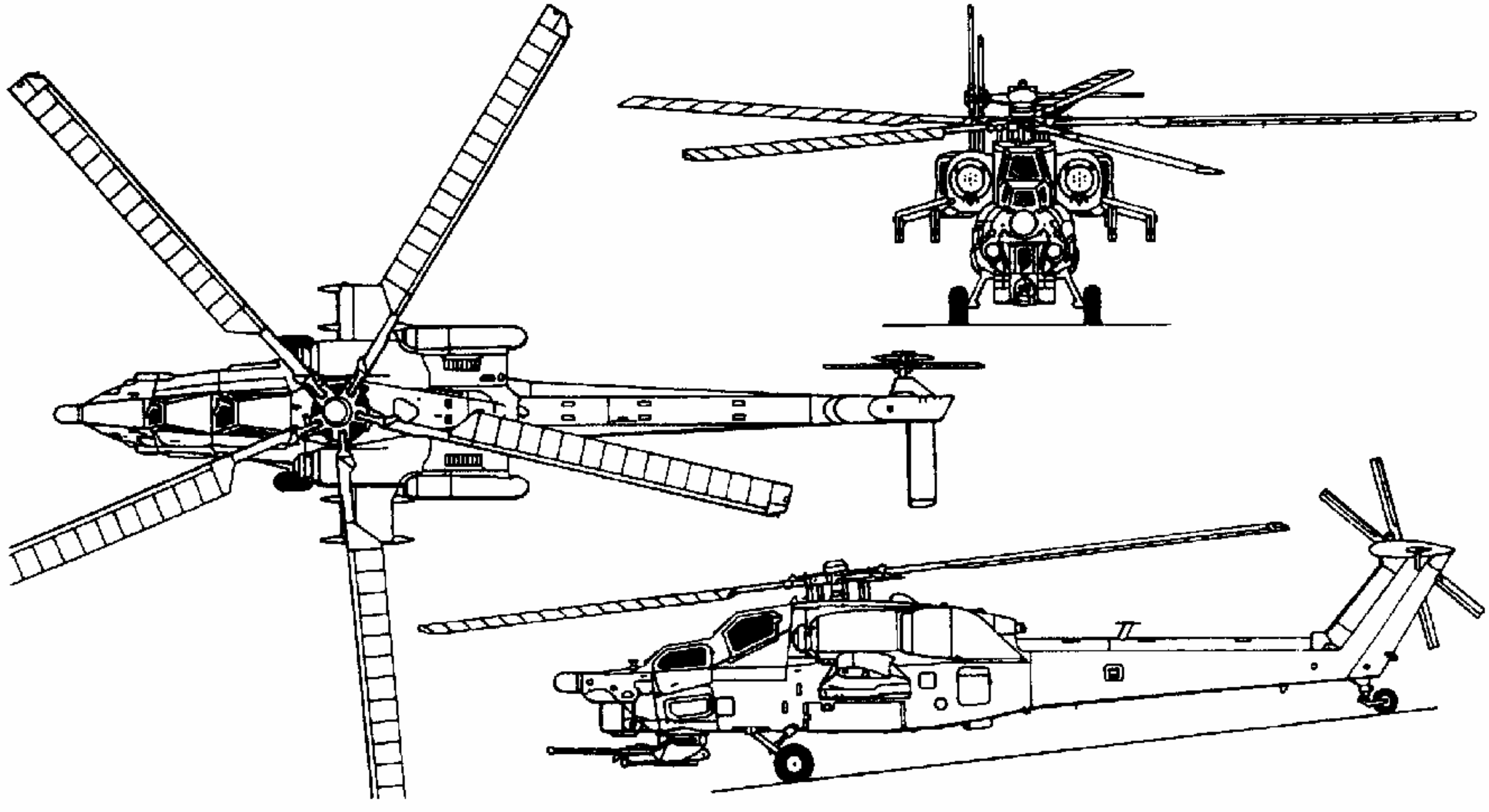


Fig. A.8.4. Mi-28 Helicopter Diagram

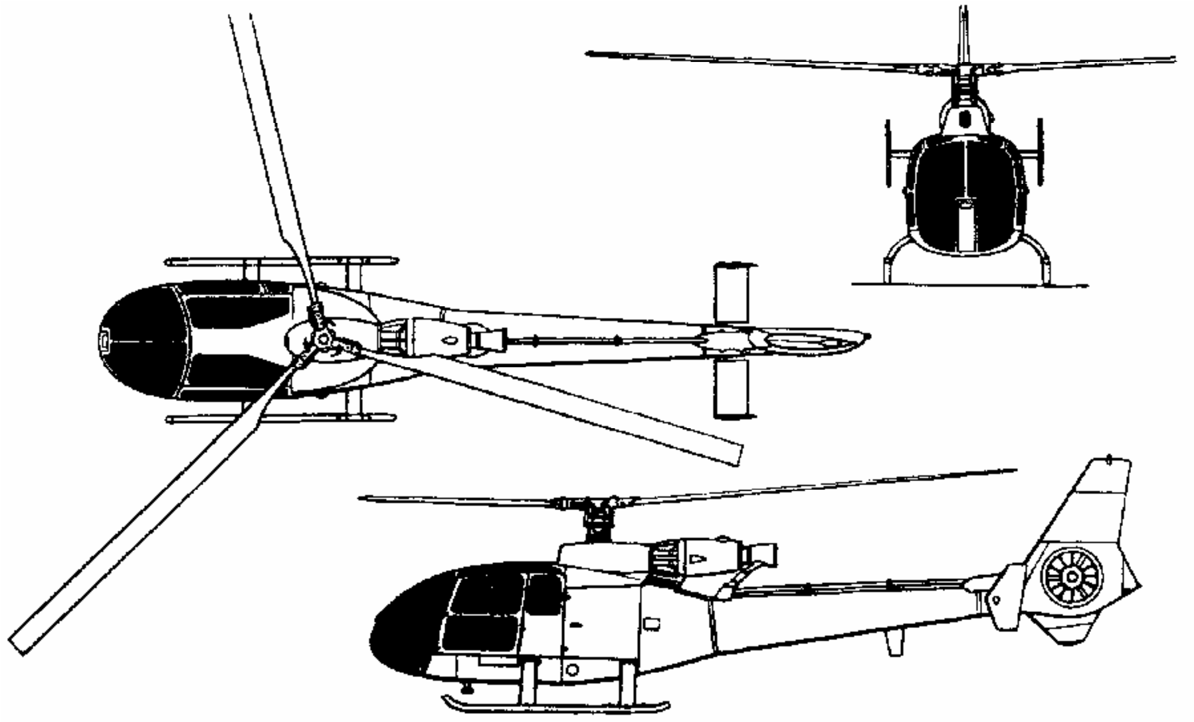


Fig. A.8.5. AS.342 “Gazel” Helicopter Diagram″

AS.342M HELICOPTER CHARACTERISTICS

Sizes, m:	
length with rotors turning	11.97
fuselage length	9.53
width with blades folded back	2.04
helicopter height	3.19
main rotor diameter	10.5
propeller disc area, m2	86.59
Engines:	1 GTE Turbomeca “Astazou” HIN
takeoff capacity, kW/h.p.	640/858
Masses and loads, kg:	
maximum takeoff mass	2000
empty helicopter mass	991
Performance:	
never-exceed speed, km/h	280
maximum cruising speed, km/h	260
economic cruising speed, km/h	238
maximum rate of climb, m/s	7.8
hovering ceiling, м:	
without ground effect taken into account	2370
with ground effect taken into account	3040
dynamic ceiling, m	4300
flight range, km	755

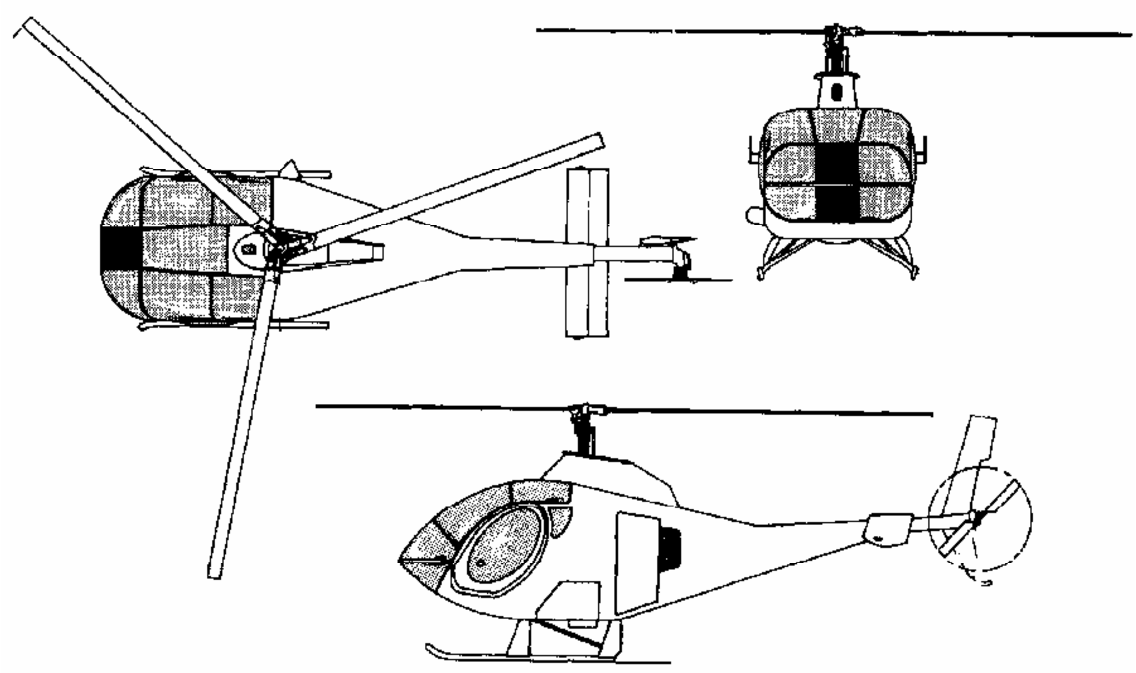


Fig. A.8.6. “Schweitzer 300C” Helicopter Diagram

“SCHWEITZER 300C” HELICOPTER CHARACTERISTICS

Sizes, m:	
length with rotors turning	9.4
fuselage length	6.77
width	2.44
helicopter height	2.66
main rotor diameter	8.18
propeller disc area, m2	52.5
Engines:	1 GTE Avco Lycoming HIO-360-DIA
takeoff capacity, kW/h.p.	142/190
Masses and loads, kg:	
maximum takeoff mass	930
empty helicopter mass	474
Performance:	
maximum cruising speed, km/h	145
economic speed, km/h	124
maximum rate of climb, m/s	12.7
hovering ceiling without ground effect taken into account, м	840
dynamic ceiling, m	3110
ferry flight range, km	370
time in flight	3 h 24 min

“APACHE” AH-64A HELICOPTER CHARACTERISTICS

Sizes, m:	
helicopter length, rotors turning	17.76
helicopter height	4.22
main rotor diameter	12.63
propeller disc area, m2	168.11
Engines:	2 GTE General Electric
takeoff capacity, kW/h.p.	T700-GE-701
maximum	2x1250/2x1695
continuous, kW/h.p.	
	2x1120/2x1522
Masses and loads, kg:	
maximum takeoff mass	9225
design takeoff mass	6670
takeoff mass in general variant	6552
empty helicopter mass	4881
fuel mass	1157
Performance:	
maximum never-exceed speed	
(in process of pitchdown), km/h	365
maximum cruising speed, km/h	296
maximum rate of climb, m/s	14.6
hovering ceiling, м:	
with ground effect taken into account	4570
without ground effect taken into account	3505
dynamic ceiling, m:	
with two engines-on	6400
with one engine-on	3290
flight range (internal fuel), km	
ferry range, km	480
time in flight in case of typical task	1700
on takeoff from field at a height of 1200 m and at t=35°C	
maximum time in flight	1 h 50 min
(with internal fuel reserve)	
	3 h 9 min

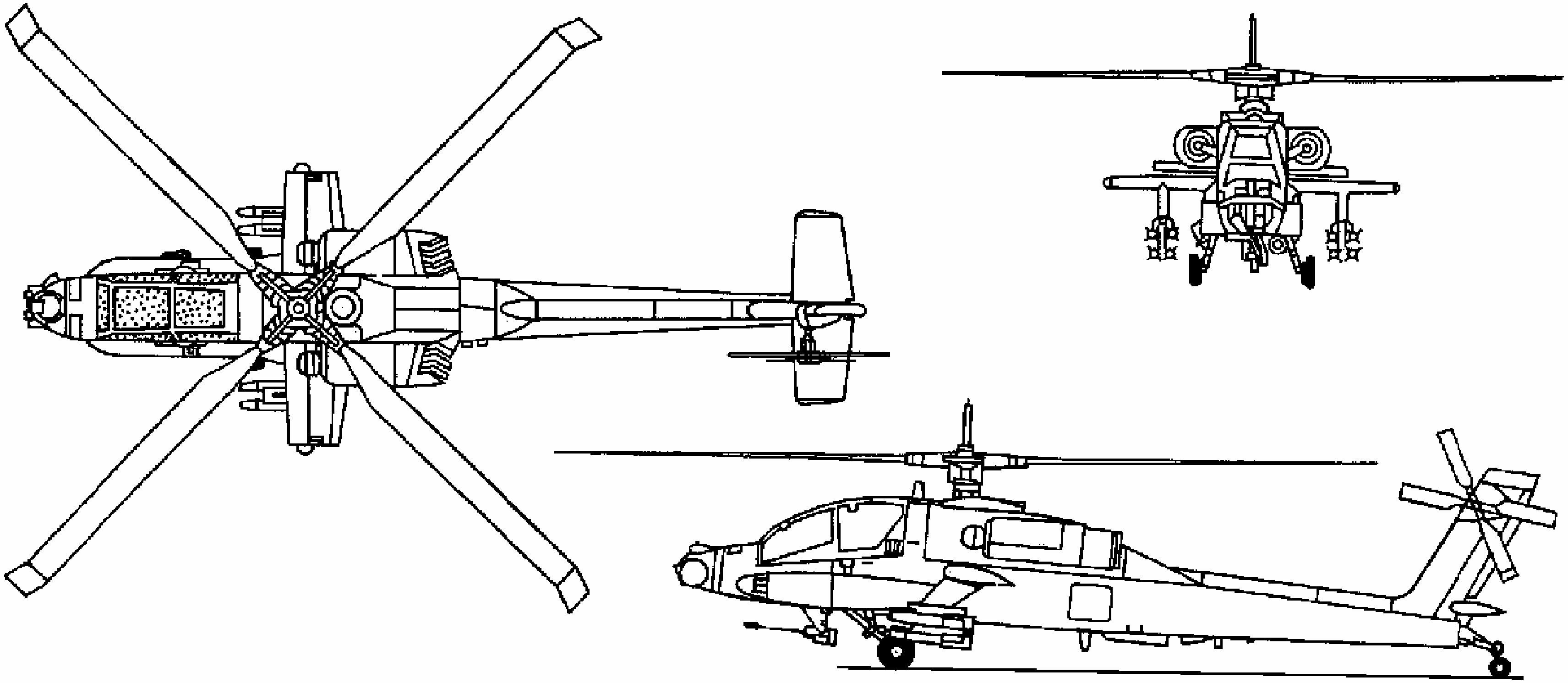


Fig. A.8.7. “Apache” AN-64A Helicopter Diagram

S-61 HELICOPTER CHARACTERISTICS

Sizes, m:

helicopter length, rotors turning	21.91
fuselage length	16.69
length with blades folded back with main rotor and tail rotor blades folded back and tail beam	14.4
helicopter height	4.74
helicopter width	4.97
main rotor diameter	18.9
propeller disc area, m2	280.5

Engines:	2 GTE General Electric T58-GE-10
takeoff capacity, kW/h.p.	2x1035/2x1400

Masses and loads, kg:

takeoff mass under task fulfilling	9300
maximum takeoff mass	9525
maximum load:	
carried in compartment	2720
on external load	3630

Performance:

maximum speed, km/h	267
cruising speed, km/h	222
maximum rate of climb, m/s	11.2
hovering ceiling	
without ground effect taken into account, m	2500
maximum flight range, km	1005

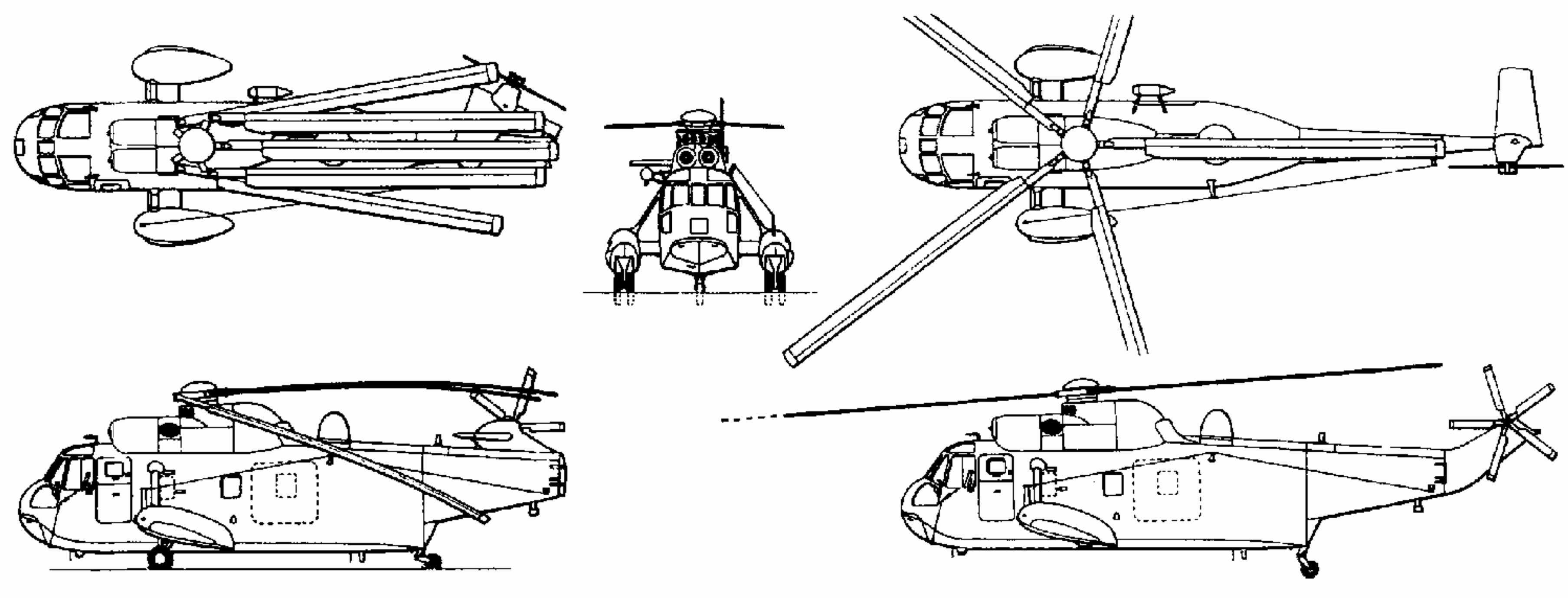


Fig. A.8.8. S-61 Helicopter Diagram

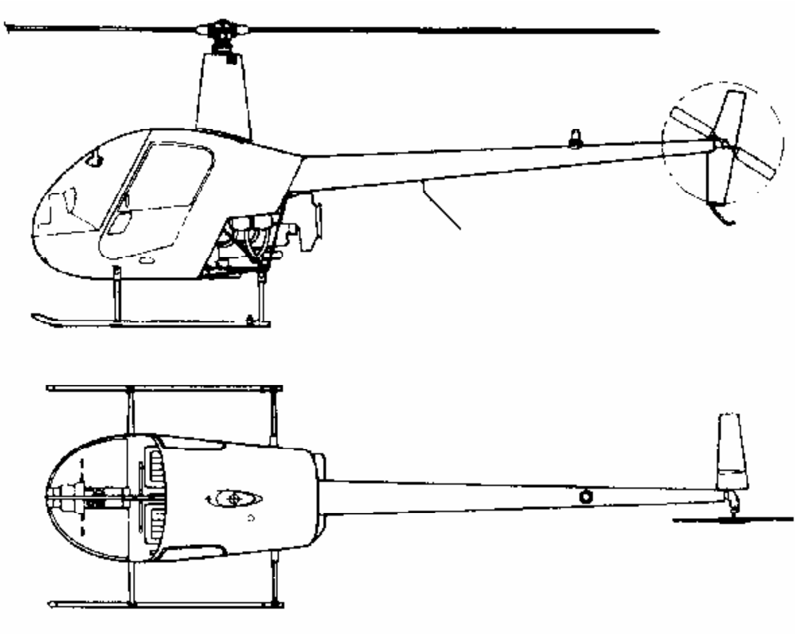


Fig. A.8.9. “Robinson” R.22 Helicopter Diagram

R.22 HELICOPTER CHARACTERISTICS

Sizes, m:	
helicopter length, rotors turning	8.76
fuselage length	6.3
height	2.67
maximum fuselage width	1.12
main rotor diameter	7.67
propeller disc area, m2	46.21
Engines:	1 PD Textron Lycoming 0-320-B20
takeoff capacity, kW/h.p.	119/160
limited to, kW/h.p.	97.5/131
Masses and loads, kg:	
takeoff mass	621
empty helicopter mass	374
Performance:	
never-exceed speed, km/h	190
maximum speed, km/h	180
maximum cruising speed, km/h	153
maximum rate of climb near the ground, m/s	6.1
dynamic ceiling, m	4265
hovering ceiling with ground effect taken into account, m	2125
flight range with auxiliary fuel reserve, km	590
maximum time in flight	3 h 20 min

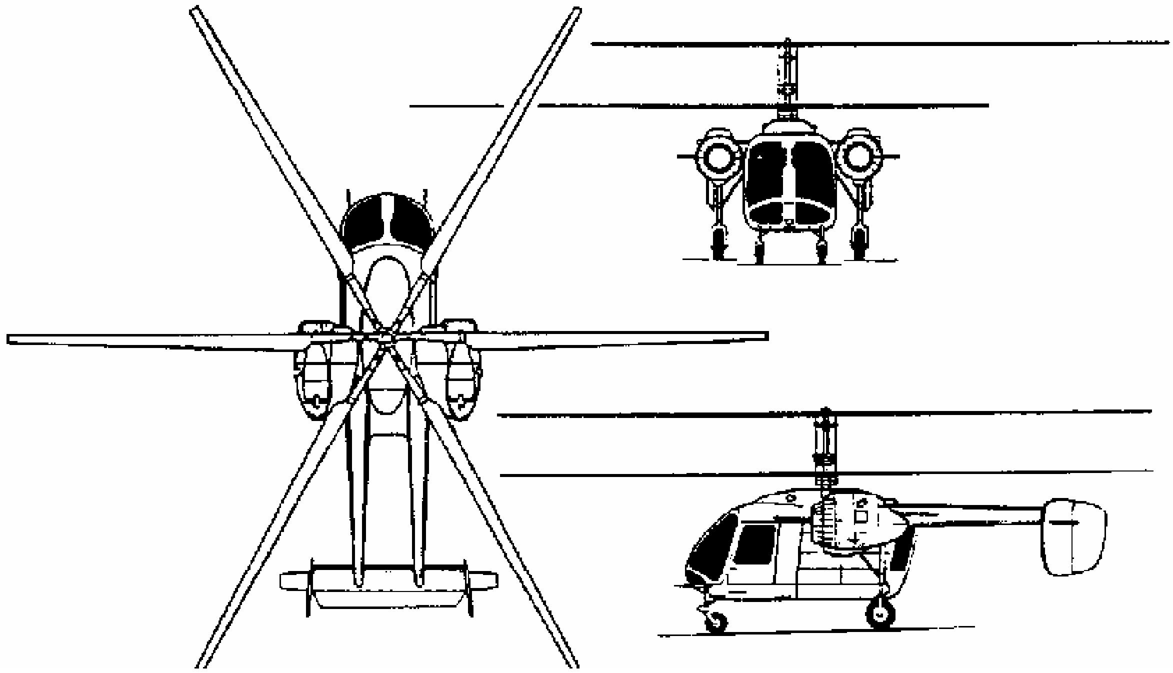


Fig. A.8.10. Ka-26 Helicopter Diagram

Ka-26 HELICOPTER CHARACTERISTICS

Sizes, m:	
fuselage length	7.75
helicopter height	4.05
helicopter width (according to engines nacelles)	3.64
main rotor diameter	13
propeller disc area, m2	132.7
Engines:	2 PD M-14B-26
takeoff capacity, kW/h.p.	2x239/2x325
Masses and loads, kg:	
maximum takeoff mass	3250
maximum payload	900
size of fuel tank and hopper with chemicals, l	800
internal fuel, l	620
Performance:	
maximum speed, km/h	160
cruising speed, km/h	130
service ceiling, m	3000
hovering ceiling with mass 3000 kg, m	800
flight range with tanks full	
main tanks with en-route fuel reserve for 0.5 h of flight, km	465
maximum time in flight, h	3.7

HELICOPTER CHARACTERISTICS EC-135

Sizes, m:	
length with rotors turning	12.1
fuselage length	10.16
fuselage width	1.56
helicopter height:	
maximum	3.75
to main rotor hub	3.23
length of helicopter compartment	4.1
main rotor length	10.2
propeller disc area, m2	81.7
Engines:	2 GTE Turbomeca ″Arrius″ 2 V or Pratt-Whitney HW 206 V
takeoff capacity, kW/h.p.	2x470/2x640
Masses and loads, kg:	
takeoff mass with load in compartment	2500
with external load	2700
of empty helicopter	1300
load, carried in compartment	1100
on external attachment	1400
Performance:	
speed, km/h:	
never-exceed	287
maximum cruising	270…274
economic cruising	240
rate of climb while climbing with horizontal speed, m/s	7.9…9.7
vertical rate of climb with single engine on, m/s	2.7
hovering ceiling, m:	
with ground effect taken into account	4750
without ground effect taken into account in conditions of	4100
MCA	2700
under MCA +20°C	6100
dynamic ceiling, m	
range with one pilot, four passengers and 145 kg of special equipment during flight on 1000 m height with fuel reserve for 20 min, km	750

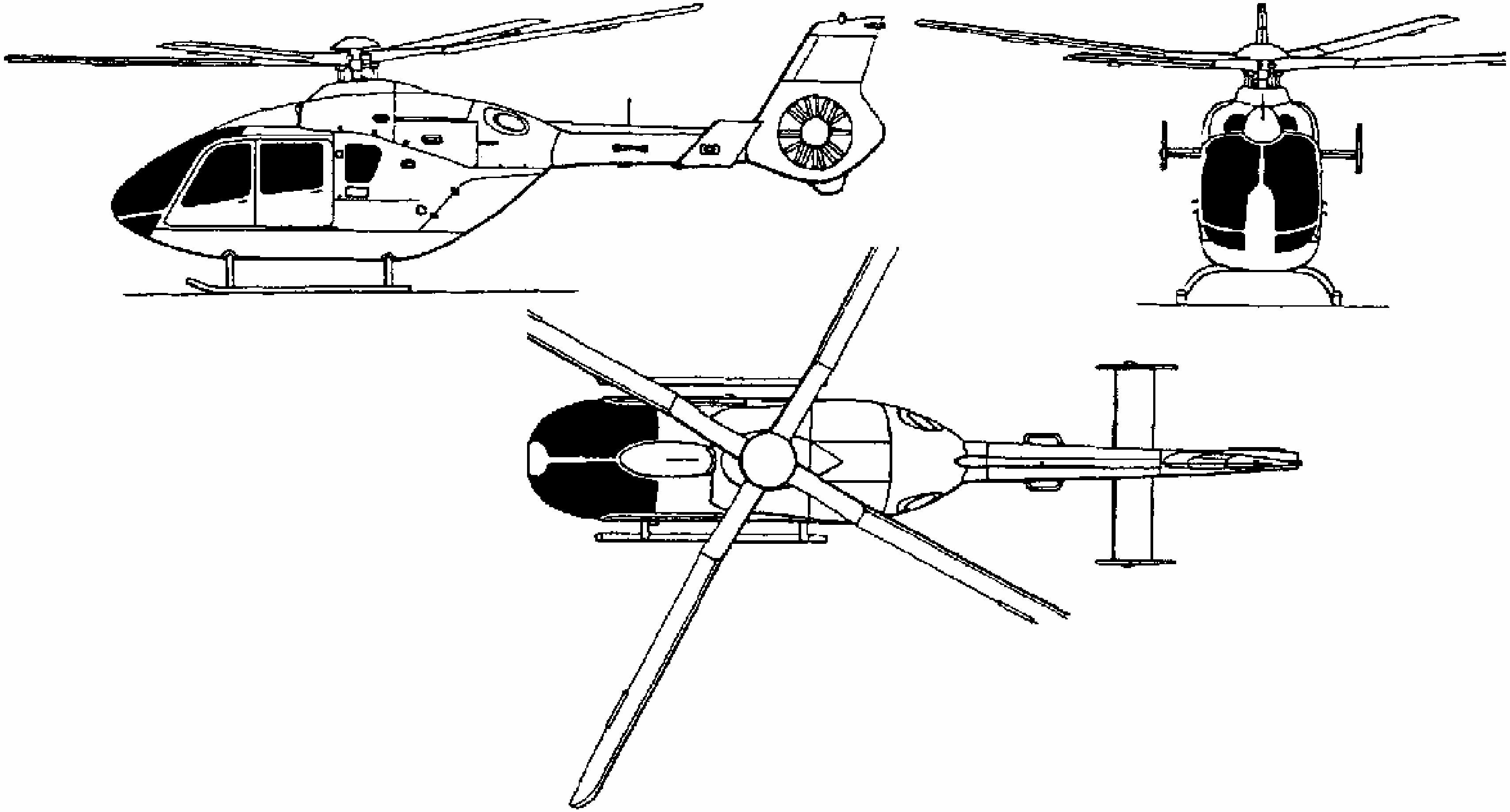


Fig. A.8.11. EC-135 Helicopter Diagram

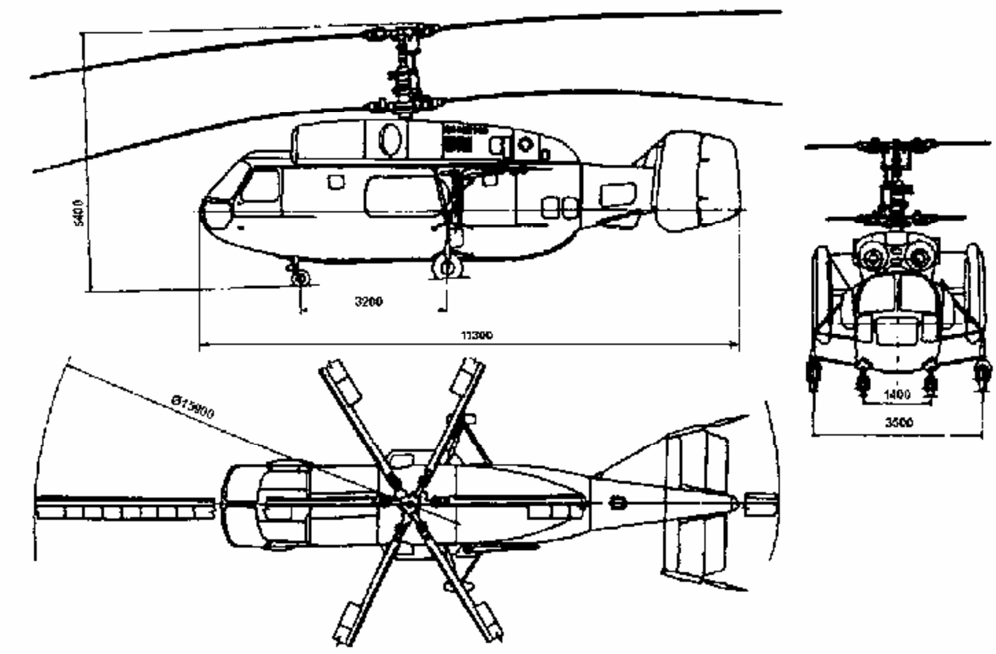


Fig. A.8.12. Helicopter Diagram Ка-32A

HELICOPTER CHARACTERISTICS Ка-32A

Sizes, m:

fuselage length	11.3
helicopter height to hub of upper rotor	5.4
main rotor diameter	15.9
propeller disc area, m2	198.5

Engines:	2 GTE T3-117 St. Petersburg SPA in name of V.Y.Klimov
takeoff capacity, kW/h.p.	2x1618/2x2200

Masses and loads, kg:

maximum flight mass	
с грузом on external attachment	12600
average takeoff mass	11000
maximum payload:	
in fuselage	4000
on external attachment	5000

Performance

(under average takeoff mass):

maximum speed, km/h	250
cruising speed, km/h	230
hovering ceiling, m	3500
maximum flight range, km	800
flight range under maximum takeoff mass	
with payload of 3500 kg	
and with en-route fuel reserve for 0.5 h of flight, km	570
time in flight, h	4.5

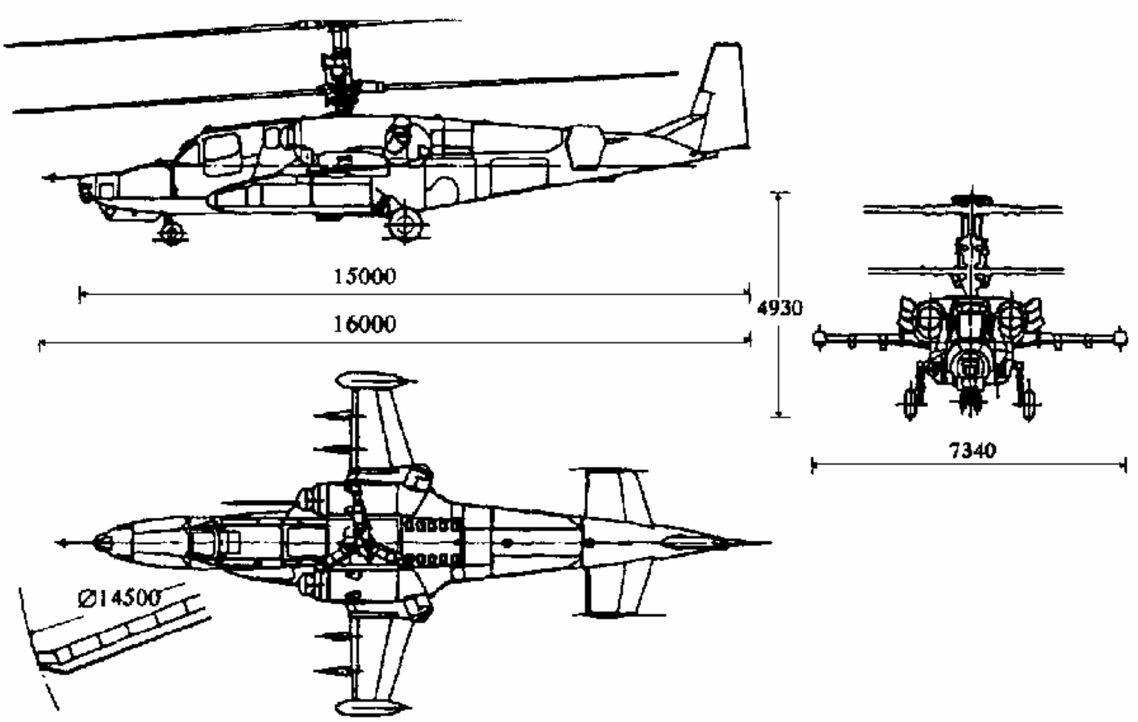


Fig. A.8.13. Helicopter Diagram Кa-50

HELICOPTER CHARACTERISTICS Кa-50

Sizes, m:	
helicopter length, rotors turning	16
main rotor diameter	14,5
propeller disc area, m2	166,5
Engines:	2 GTE TV3-117 St. Petersburg SPA in name of V.Y. Klimov
takeoff capacity, kW/h.p.	2x1618/2x2200
Masses and loads, kg:	
maximum takeoff mass	10800
average takeoff mass	9800
empty helicopter mass	7700
Performance:	
maximum speed during flat dive, km/h	350
maximum speed during horizontal flight, km/h	310
hovering ceiling out of ground effect , m	4000
vertical rate of climb on 2500 m height, m/s	10
speed of sideways flight, km/h	80
speed of rearward flight, km/h	90
service flight range, km	450
ferry range, km	1200
maximum flight maneuvering load	3

BOEING CH-47 HELICOPTER CHARACTERISTICS

Sizes, m:	
length with rotors turning	30.18
fuselage length	15.54
helicopter width	3.78
helicopter height	5.68
distance between axes of main rotors	11.94
main rotor diameter	18.29
propeller disc area, m2	2x262.65
Engines:	2 GTE Lycoming T55-1-L-12
emergency takeoff capacity, kW/h.p.	2x3356/2x4400
takeoff capacity, kW/h.p.	2x2796/2x3750
Masses and loads, kg:	
takeoff mass with full load	20865
maximum takeoff mass	22680
design average takeoff mass	14970
empty helicopter mass	10615
maximum transported load:	
within the cabin	6310
on external attachment	10340
Performance:	
maximum speed near the ground, km/h	285
maximum cruising speed, km/h	260
hovering ceiling without ground effect taken into account,	1675
м	3100
dynamic ceiling, m	615
flight range, km	
radius of action:	55
with external load with mass of 9390 kg, km	
with distribution of load with mass of 8165 kg	185
within fuselage, km	

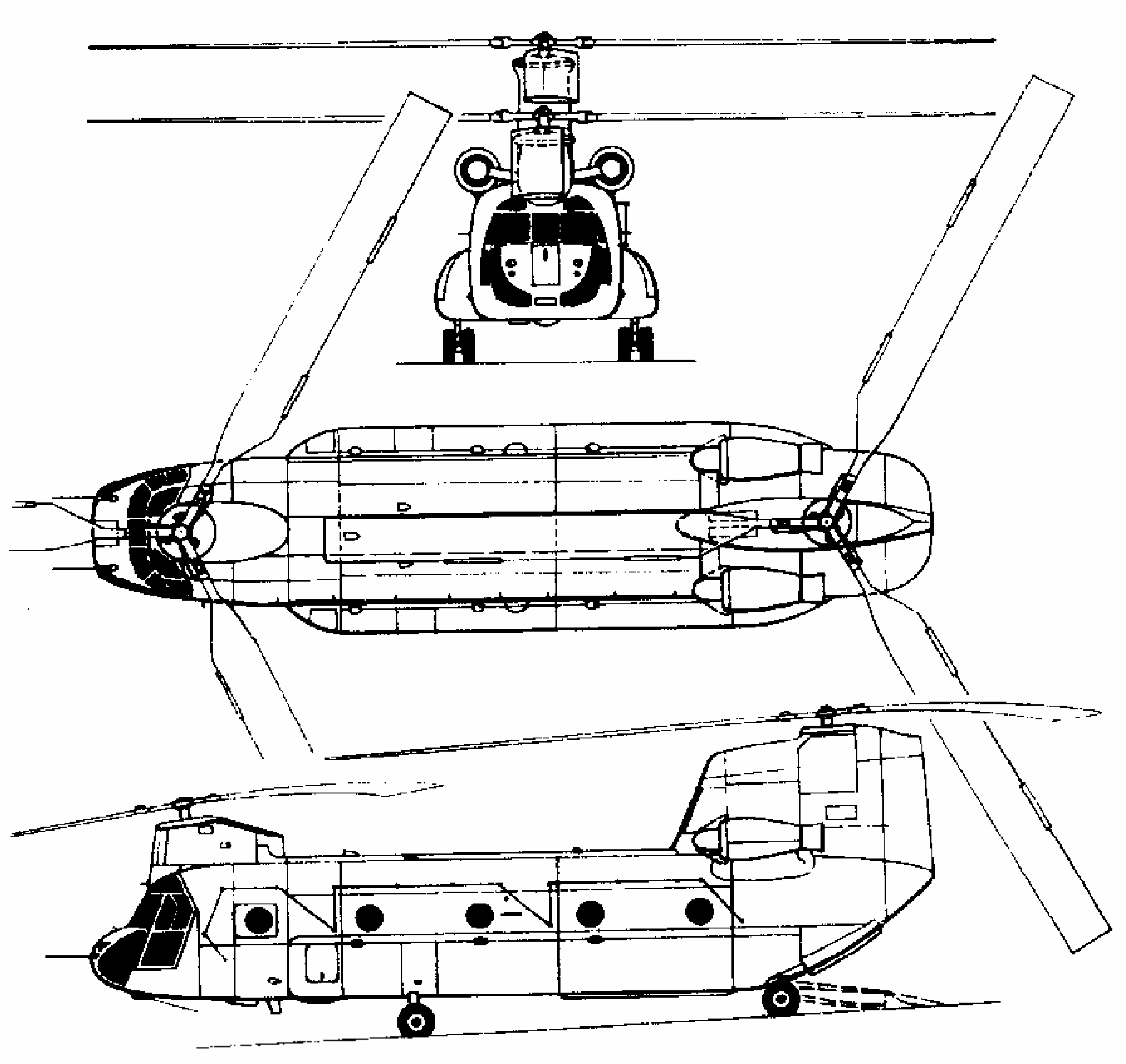


Fig. A.8.15. “Chinook” CH-47 Helicopter Diagram

Table A.8

General characteristics of turboshaft engines

Engine identification, designing country	Maximum rating capacity, kW / Specific fuel consumption while maximum rating, kg/kW·h	Air flow, kg/s / Engine pressure ratio	Mass, kg / Length x diameter, m	Engines installed on following helicopters
T800-LHT-800, USA	905.3 / 0.283	• / •	135 / 0.86×0.51	LHX
T800-APW-800, USA	883.2 / 0.285	• / 15	135 / 0.98×0.47	LHX
T701-AD-700, USA	5946.9 / 0.289	20.1 / 12.8	535 / 1.88×0.94	HLH
T702-LD-700, USA	452.6 / 0.35	2.03 / 8.5	110 / 0.79×0.57	HH-65A
T703-A-700, USA	478.4 / 0.364	2.54 / 8.6	110 / 1.1×0.64	OH-58D
T700-GE-700, USA	1192.3 / 0.285	4.5 / 15	200 / 1.17×0.64	UH-60A
T700-GE-701, USA	1251.2 / 0.285	4.5 / 15	200 / 1.17×0.64	AH-64A
T700-GE-701A, USA	1262.2 / 0.285	• / •	200 / 1.17×0.64	S-70C
T700-GE-701C, USA	1361.6 / 0.277	• / •	200 / 1.17×0.64	
T700-GE-401, USA	1243.8 / 0.285	4.5 / 15	196 / 1.17×0.64	HH-60D
T700-GE-401A, USA	1251.2 / 0.289	• / •	205 / 1.17×0.64	EH-101
T700-GE-401C, USA	1361.6 / 0.277	• / 17.1	200 / 1.17×0.64	
T400-WV-402, USA [1)]	1449.9 / 0.364	• / 7.7	350 / 1.7×1.12	AH-IJ and T
T400-CP-400 и 401, USA [1)]	1324.8 / 0.367	• / 7.4	325 / 1.7×1.12	AH-IJ, UH-IN, VH-IN
T64-GE-419, USA	3496.0 / 0.289	• / 14.9	330 / 2.0×0.51	CH-53E, MH-53E

Table A.8 (Continued)

Engine identification, designing country	Maximum rating capacity, kW / Specific fuel consumption while maximum rating, kg/kW·h	Air flow, kg/s / Engine pressure ratio	Mass, kg / Length x diameter, m	Engines installed on following helicopters
T64-GE-415 и 416, USA	3223.7 / 0.289	• / 14.8	330 / 2.0×0.51	CH-53D and E,RH-53D and E
T64-GE-100, USA	3186.9 / 0.296	• / 14.9	330 / 2.0×0.51	S-65C
T64-GE-7A, USA	2899.8 / 0.289	12.8 / 14.1	330 / 2.0×0.51	CH-53C, HH-53C
T63-A-720, USA	309.1 / 0.401	1.56 / 7.3	70 / 1.05×0.59	OH-58C
T58-GE-16, USA	1376.3 / 0.326	• / 8.6	200 / 1.17×0.64	CH-46E
T58-GE-10, USA	1030.4 / 0.369	• / 8.4	160 / 1.5×0.53	CH-46D, VH-46D and F, SH-3D
T58-GE-8F, USA	993.6 / 0.369	• / 8.2	160 / 1.5×0.53	CH-46A, SH-3G, SH-2F
T58-GE-5, 100 and 402, USA	1104.0 / 0.369	• / 8.4	130 / 1.5×0.53	SH-3E and F, HH-3E and F
T55-L-712, USA	2760.0 / 0.319	• / 8.2	340 / 1.19×0.61	CH-47D
T55-L-11A and E, USA	2760.0 / 0.326	12.25 / 8.2	305 / 1.12×0.62	CH-47C
T53-L-703, USA	1140.8 / 0.369	• / 8.0	250 / 1.21×0.58	AH-1
T53-L-13B, USA	1030.4 / 0.356	5.53 / 7.4	250 / 1.21×0.58	VH-1H
250-C20 F and J, USA	309.1 / 0.401	5.53 / 7.4	70 / 0.98×0.51	Hughes 500D and E
250-C30 L, M and P, USA	478.4 / 0.364	2.54 / 8.6	110 / 1.04×0.64	Hughes 530, S-76A
250-C30S, USA	515.2 / 0.364	2.54 / 8.6	110 / 1.04×0.64	A-109, S-76Mk2

Table A.8 (Continued).

Engine identification, designing country	Maximum rating capacity, kW / Specific fuel consumption while maximum rating, kg/kW·h	Air flow, kg/s / Engine pressure ratio	Mass, kg / Length x diameter, m	Engines installed on following helicopters
250-C34, USA	566.7 / 0.364	2.54 / 8.5	120 / 1.1×0.64	
GEM-2 Mk1001, Great Britain	662.4[2)] / 0.319[2)]	• / 12.0	150 / 1.08×0.6	WG-13
GEM-2 Mk1004, Great Britain	761.8[2)] / 0.326[2)]	• / 11.5	140 / 1.08×0.6	A-129
GEM-41-1, Mk1014, Great Britain	824.3[2)] / 0.299[2)]	• / 12.7	155 / 1.08×0.6	WG-13, WG-30
″Gnome-H1400-1″ Great Britain	1221.8 / 0.374	6.26 / 8.5	150 / 1.08×0.6	″Commando″
″Gnome-H1400″, Great Britain	1104.0 / 0.374	6.26 / 8.4	150 / 1.08×0.6	″Sea King″
″Nimbus Mk1051502″, Great Britain	522.6 / 0.516	• / 6	305 / 1.85×0.89	″Scout″
″ Nimbus Mk1031503″, Great Britain	522.6 / 0.516	• / 6	295 / 1.85×0.89	″Wasp″
″Artust-3V″, France	412.2 / 0.408	• / 5.4	180 / 1.8×0.67	SA-315, SA-316
″Astasu-3″, France	4342 / 0.394	6.26 / 8.4	180 / 1.8×0.67	SA-341
″Astasu-2A″, France	382.7 / 0.383	• / •	• / 1.27×0.48	SA-318C
″Astasu-14B″, France	434.2 / 0.383	• / •	165 / 1.43×•	SA-319B
″Astasu-14H″, France	434.2 / 0.346	• / 7.5	160 / 1.47×0.56	SA-342
″Astasu-18A″, France	434.2 / 0.346	• / •	155 / 1.33×0.7	SA-360
TM333-1M, France	669.8 / 0.319	• / 11.0	135 / 0.94×0.56	SA-365M

Table A.8 (Continued)

Engine identification, designing country	Maximum rating capacity, kW / Specific fuel consumption while maximum rating, kg/kW·h	Air flow, kg/s / Engine pressure ratio	Mass, kg / Length x diameter, m	Engines installed on following helicopters
TM319, France	375.4[2)] / 0.340[2)]	• / •	85 / 0.81×0.52	AS-355, AS-350
″Makila-1A″, France	1405.8 / 0.292	• / •	240 / 2.0×0.57	SA-332
″Turmo-3C7″, France	1185.0 / 0.394	5.9 / 5.9	325 / 1.95×0.72	SA-321
″Turmo-4C″, France	1148.2 / 0.394	• / 5.9	• / 2.18×0.72	SA-330
″Ariel″, France	537.3[2)] / 0.340[2)]	• / 8	110 / 1.2×0.63	SA-365, AS-350
TM-322-01, Great Britain, France	1545.6 / 0.272	• / 14.7	240 / 1.17×0.65	TP-90 and EH-101
MTM385R, FRG, France	1030.4 / 0.285	3.4 / 11.2	190 / 1.3×0.7	PAH-2, HAC-3G
T64-MTU-7, FRG [3)]	2888.8 / 0.296	• / 13	320 / 1.48×0.51	CH-53G
T53-L-13, FRG [3)]	1030.4 / 0.356	5.53 / 7.4	245 / 1.21×0.59	VH-1D
250-MTU-020B, FRG [3)]	309.1 / 0.401	1.54 / 7.2	70 / 0.98×0.51	Bo-105M and P
T58-GE-3, Italy	975.2 / 0.369	• / 8.2	140 / 1.5×0.53	AB-204B
T55-K-712, Japan [3)]	2760.0 / 0.319	• / 8.2	340 / 1.19×0.61	CH47-414
T53-K-703, Japan [3)]	1093.0 / 0.369	• / 8	250 / 1.21×0.59	AN-1S
T53-K-13B, Japan [3)]	1030.4 / 0.356	5.53 / 7.4	245 / 1.21×0.59	VH-1H

Table A.8 (Continued)

Engine identification, designing country	Maximum rating capacity, kW / Specific fuel consumption while maximum rating, kg/kW·h	Air flow, kg/s / Engine pressure ratio	Mass, kg / Length x diameter, m	Engines installed on following helicopters
GTE-350, Russia	$\frac{294.4}{0.469}$	$\frac{2.1}{5.2}$	$\frac{137}{1.35\times0.52\times0.68}$	Mi-2
TP-0-100, engine-building DB of Omsk, Russia	$\frac{530.9}{0.346}$	$\frac{\bullet}{\bullet}$	$\frac{160}{1.275\times0.78\times0.735}$	Ka-126
TP2-117A, SPA in name of V.Y. Klimov, Russia	$\frac{530.9}{0.346}$	$\frac{10.0}{6.6}$	$\frac{332}{2.843\times0.556\times0.748}$	Mi-8
TP7-117B, SPA in name of V.Y. Klimov, Russia	$\frac{1692.8}{0.283^{4)}}$	$\frac{7.95}{6.9}$	$\frac{250}{2.14\times0.94\times0.886}$	Mi-38
TP3-117, SPA in name of V.Y. Klimov, Russia	$\frac{1637.6}{0.299}$	$\frac{9}{9}$	$\frac{285}{2.085\times0.65\times0.728}$	Ka-32, Ka-50, Mi-17
TPE-1500, DB of mechanical engineering of Rybinsk, Russia	$\frac{956.8}{\bullet}$	$\frac{\bullet}{\bullet}$	$\frac{\bullet}{\bullet}$	Ka-62
D-25B (TP-2BM), engine-building DB of Perm, Russia	$\frac{4048.0}{\bullet}$	$\frac{26.2}{5.6}$	$\frac{1200}{2.737\times\bullet}$	Mi-6, Mi-10
D136, engine-building DB of Zaporojie “Progress”, Ukraine	$\frac{8420.0}{0.267}$	$\frac{35.55}{18.4}$	$\frac{1050}{3.964\times1.67\times1.161}$	Mi-26
PW-206A, “Pratt & Whitney” Corp, Canada	$\frac{456.3}{\bullet}$	$\frac{\bullet}{\bullet}$	$\frac{\bullet}{\bullet}$	Ka-118
Alisson 250-020B, ″General Motors″ Corp	$\frac{309.1}{0.400}$	$\frac{1.56}{7.2}$	$\frac{71.6}{0.589\times\bullet}$	Ka-226, Bell-206, Bo-105C, Hughes 500

Note: 1) development of Canadian department of American Company “Pratt & Whitney”; 2) emergency power; 3) American license manufacture; 4) cruising power; 5) numerator or denominator points (columns 2, 3, 4) mean null data.

Appendix 9

Helicopter Specification Requirements

Helicopter pilot project includes overall view drawings, basic parameters and geometry, and results of preliminary estimation for performance characteristics and operation costs.

Basic helicopter configuration developed at the stage of preliminary design serves as a reference point for subsequent procedures. That is why all the results obtained at this stage must be assorted and generalized in the final report as well as basic information.

Predesign department usually provides following documents:

– overall view drawings which tell about configuration, basic components, interior layout, and shapes of helicopter units. Theses drawings represent the arrangement of the helicopter primary load-bearing members;

– description together with basic characteristics and helicopter geometry, which corresponds to typical helicopter description given in aviation handbooks;

– results of preliminary estimation of parameters, flight characteristics, and economic efficiency.

As a rule, pilot projects include basic statistics of similar helicopters.

Pilot project must prove the necessity of a new helicopter and provide possible strategies for its development. At the stage of pilot project, designers must confirm helicopter configuration applicability and design integrity.

Overall view drawings must be large-scale and contain data on helicopter exterior, load-carrying structure, and preliminary general layout.

Here is an approximate helicopter requirements specification:

Adopted abbreviations and designations

Introduction

Basics for development

1. Helicopter purpose
2. Expected operating conditions
   - 2.1. State parameters and impact factors of ambient conditions
   - 2.2. Operational factors
   - 2.3. Distinguishing features of helicopter application
3. Brief description
   - 3.1. General layout
   - 3.2. Flight compartment
   - 3.3. Cargo cabin
   - 3.4. Principal layout of systems and equipment
4. Helicopter geometry characteristics
   - 4.1. Overall view
   - 4.2. Overall dimensions
   - 4.3. Wing
   - 4.4. Fuselage

    4.5. Tail unit
    4.6. Transmission
    4.7. Landing gear
    4.8. Main power plant
    4.9. Streamlined surface
5. Weight and balance characteristics
    5.1. Maximal takeoff mass
    5.2. Maximal payload
    5.3. Maximal fuel amount
    5.4. Weight report of empty equipped helicopter
    5.5. Centre-of-gravity position characteristics
6. Primary flight characteristics
    6.1. Basic performance characteristics
    6.2. Flight range
7. Transporting abilities of helicopter
    7.1. Transporting abilities
    7.2. Variants of loading
8. Limits, lifetime and service life
    8.1. Operational g-forces
    8.2. Maximum flight speed
    8.3. Permissible g-forces of cargo fastening during emergency landing
    8.4. Maximum excessive pressure in pressurized cabin
    8.5. Typical conditions and warranties of providing design lifetime
9. Program of helicopter production
    9.1. Proposals of terms of helicopter production
    9.2. Basic principles of helicopter industrial engineering
    9.3. Proposals for cooperation
    9.4. Programs of serial production
    9.5. Manpower for helicopter production
    9.6. Manpower for production preparation
    9.7. Helicopter development costs
    9.8. First price of the helicopter for different production amounts
10. Flight tests and certification
    10.1. Content of flight tests
    10.2. Helicopter certification
11. Economy and operation
    11.1. Initial data
    11.2. Direct operational costs
    11.3. Conclusions
12. Technical perfection and competitiveness
    12.1. Brief description of market
    12.2. Definition of technical level
    12.3. Comparison with analogues

Table A.9

Scheme of distributing standards of system developing and putting in production aviation products

General data of development and put in production system, terms and definitions	Item Life Cycle Stages				
	Research works, PILOT PROJECT (ways and substantiation)	Development Activity	Production	Providing operation and repair in industrial plants	Phasing out and taking out from service
	Statement about procedure of developing the aircraft				
GOST B 15.001-80. General data **GOST B 15.004-80.** Item life cycle stages **GOST 15,001-82.** **GOST B 15,011-85.** Patent investigation procedure **GOST 15.012-84.** Patent card	**GOST B 15.101-79.** Requirements specification (technical requirements) for research works **GOST B 15.102-84.** Requirements specification (technical requirements) for pilot project. Basics .	**GOST B1 00203-85.** **GOST B 15.201-83.** Requirements specification (technical requirements) for development activity **GOST B 15.203-79.** Development activity procedure. Basics. **GOST B 15.204-79.** Development activity procedure for preproduction model components. Basics.	**GOST B 15.204-79.** Development activity procedure for preproduction model components. Basics. **GOST B 15.301-80.** Putting items into production. General. **GOST B 15.305-85.** Follow-on in production. General. **GOST B 15.306-79.** Warranties. General.	**GOST B 15.701-77** Order of issuing bulletins and working according to them. Basics. **GOST B 15.702-83** Order of assigning service life, storage life. Basics	**GOST B 15.801-79.** Phasing part out. General. **GOST B 15.802-85.** Phasing material out. General.

Table A.9. (Continued)

General data of development and put in production system, terms and definitions	Item Life Cycle Stages				
	Research works, PILOT PROJECT (ways and substantiation)	Development Activity	Production	Providing operation and repair in industrial plants	Phasing out and taking out from service
	Statement about procedure of developing the aircraft				
	GOST B 15.103-84. Pilot project execution procedure. General **GOST B 15.104-84.** Own section of pilot project execution procedure. General **GOST B 15.105-79.** Research work procedure. General	**GOST B 15.205-79.** Research work procedure for developing components of items for interindustry application. General. **GOST B 15.206-84.** Programs providing reliability. General requirements. **GOST B 15.207-90.** Requirements in standartization and unification to items when doing developing activity	**GOST B 15.307-77.** Tests and acceptance of serial products. General. **OST 1 00350-79.** Airplane and helicopter design documentation transfer procedure to serial production plant for manufacturing development series	**GOST B 15.703-78.** Procedure of making and satisfying claims. General. **GOST B 15.704-83.** Spares, tools and accessories. General.	

Table A.9. (Continued)

Item Life Cycle Stages				
Research works, PILOT PROJECT (ways and substantiation)	Development Activity	Production	Providing operation and repair in industrial plants	Phasing out and taking out from service
Statement about procedure of developing the aircraft			**GOST 20436-75.** Aircraft products reliability. General requirements to programs providing reliability of airplane and helicopters. **GOST B 15.708-89** Typical specification of spares.	
GOST B 15.106-79. Research component works procedure. General. **GOST B 15.107-79.** Research works procedure for components having interinductry application. General. **GOST B 15.108-83.** Procedure of developing and putting in production materials. General.	**GOST B 15.208-82.** Unified through plan for creating preproduction models (systems, complexes). General. **GOST B 15.209-85.** Procedure of development, approval and application of limiting lists of products and materials allowed for application while developing and modifying machinery **GOST B 15.210-78.** Tests of preproduction models. General.	**GOST B 15.303-84** Establishing production at doubling enterprises		

End of Table A.9.

General data of development and put in production system, terms and definitions	**Item Life Cycle Stages**					General data of development and put in production system, terms and definitions
	Research works, PILOT PROJECT (ways and substantiation)	Development Activity	Production	Providing operation and repair in industrial plants	Phasing out and taking out from service	
	Statement about procedure of developing the aircraft					
	GOST B 15.708-89 Typical specification of spares. **GOST B 15.110-81.** Reporting documentation of research and development activities. General.	**GOST B 15.211-78.** Procedure of developing programs and techniques of tests, preproduction models. General. **Branch standard 1 00038-87.** Aircraft. Maintenance facilities and engines. Expertise procedure. **GOST B 15.501-90.** Operational and repair documents for helicopter (general requirements to nomenclature, construction, contents, execution, issuing and implementing revisions). **Branch standard 1 02730-92** Procedure of development and submission of mock-up for mock-up approval commission				

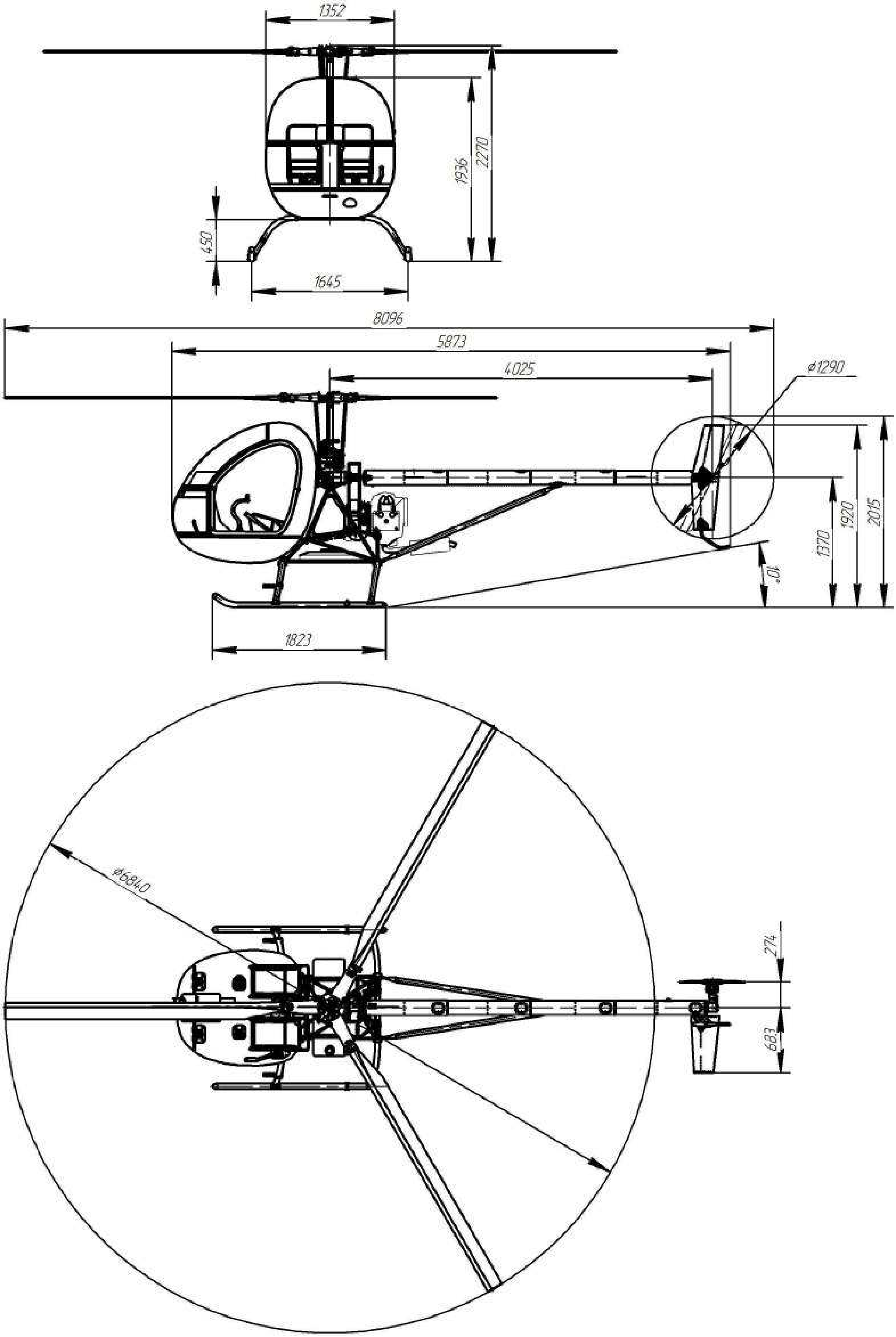


Fig. A.9.1. AK1-3 Light Helicopter Overall View

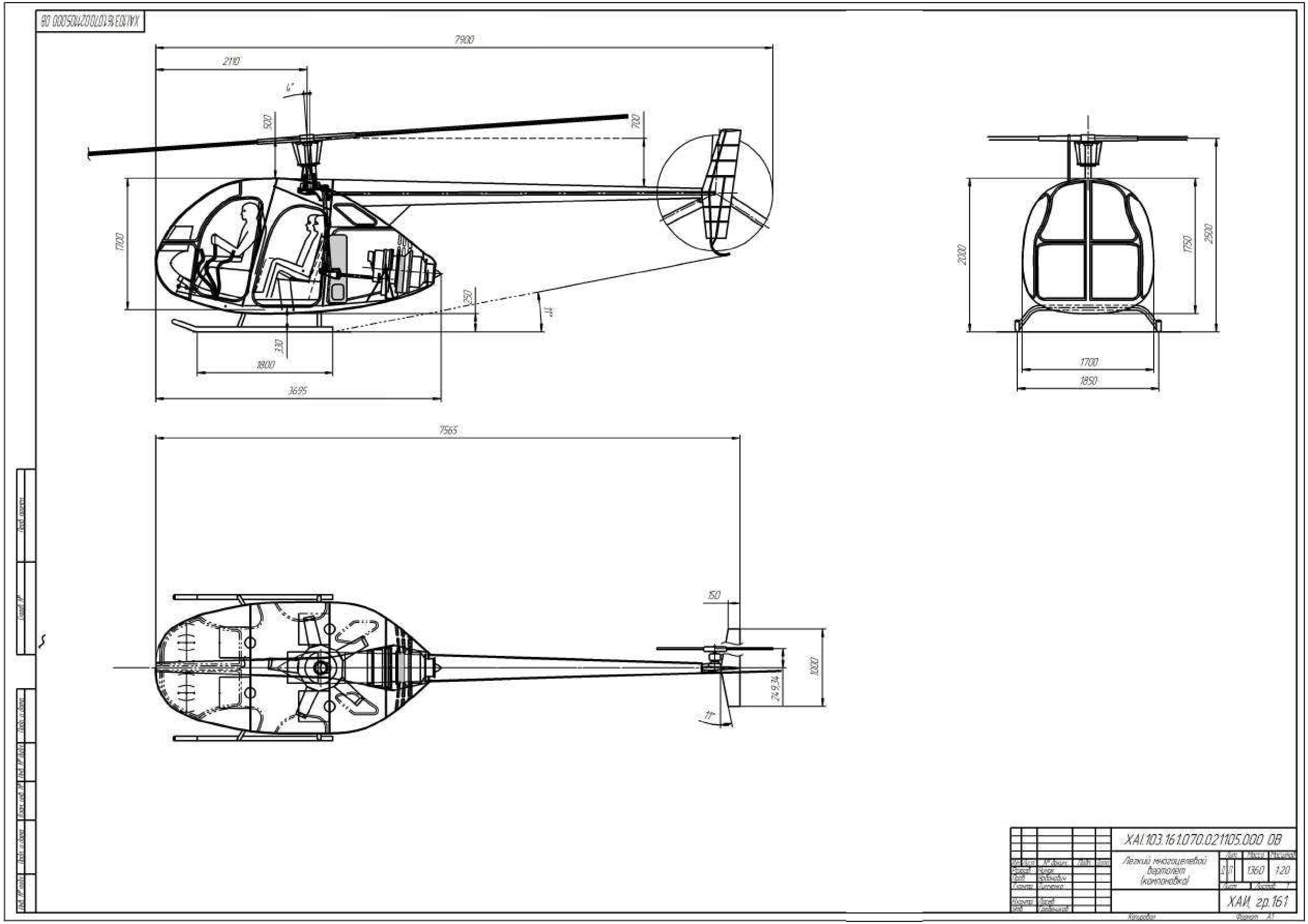
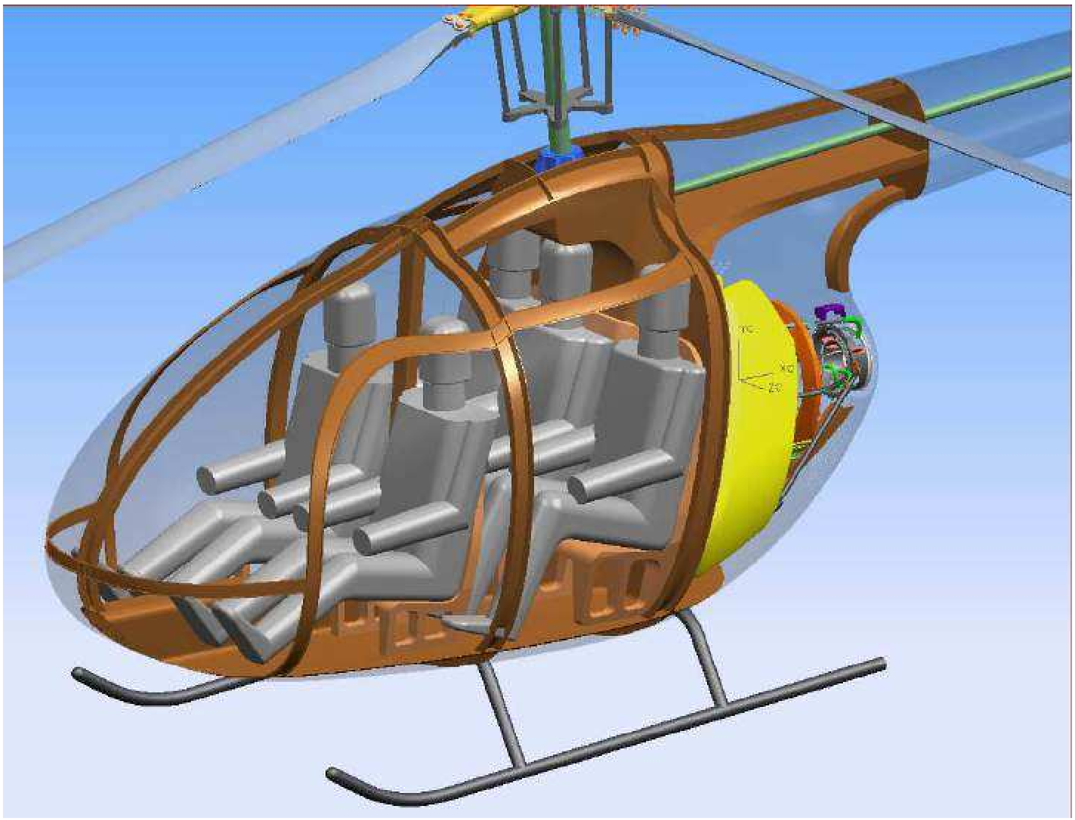
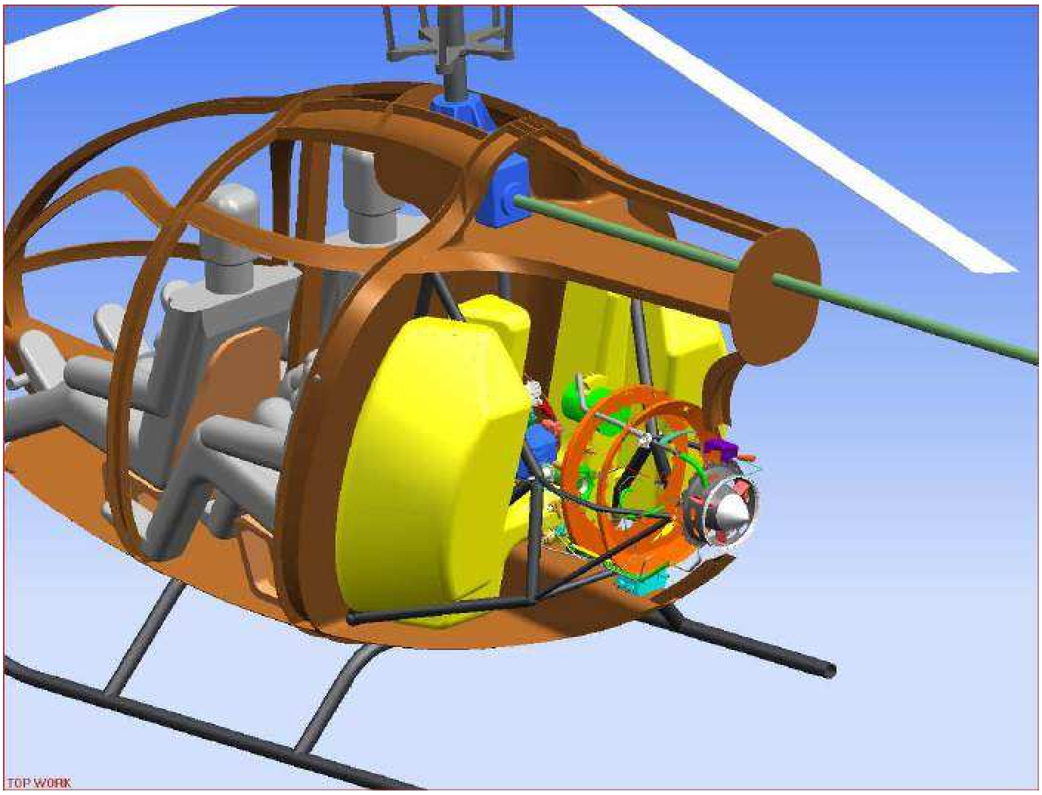


Fig. A.9.2. Light Helicopter General Arrangement

Fig. A.9.3. Light Helicopter Aerodynamic Configuration

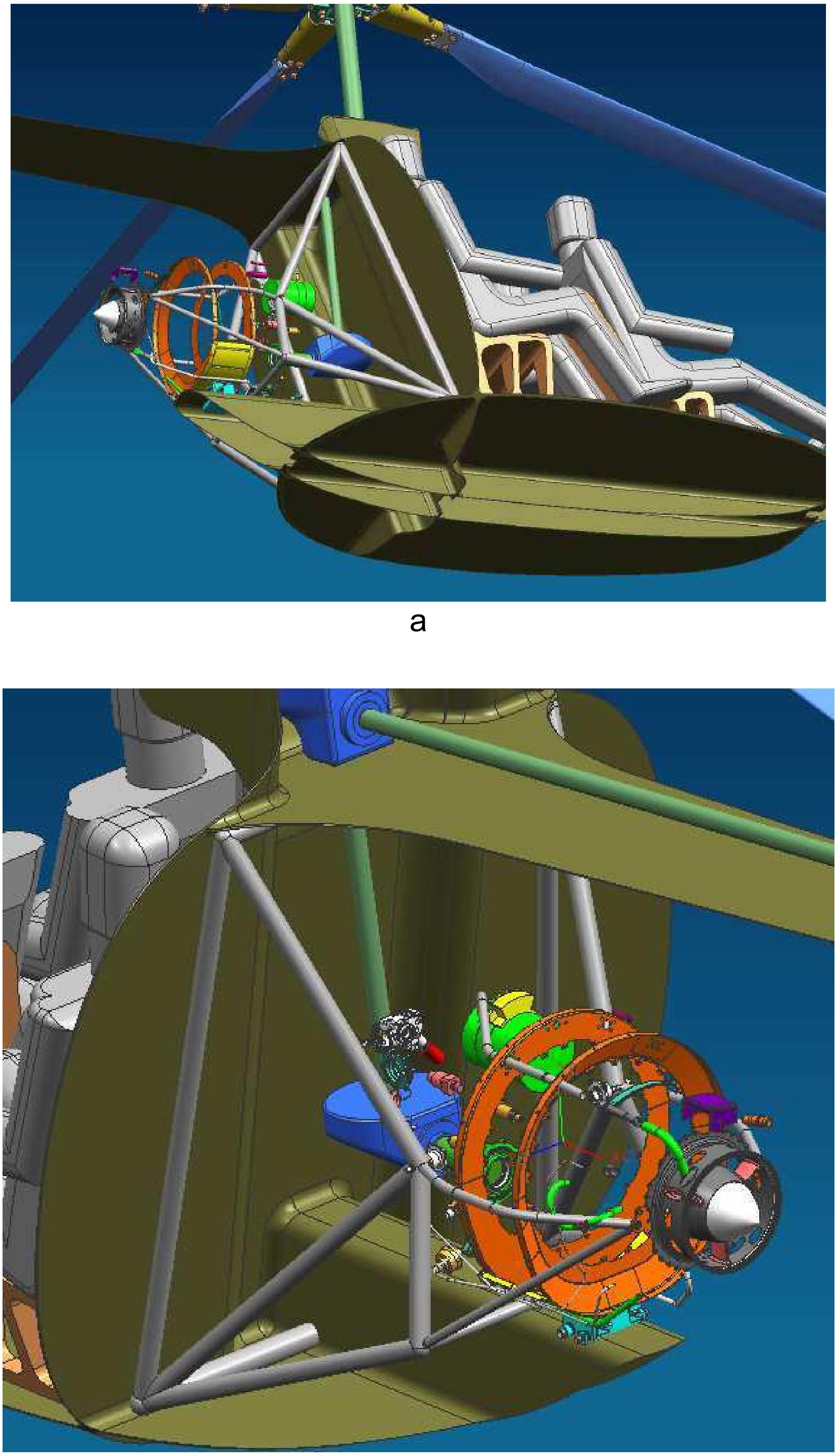


a



b

Fig. A.9.4. Space-Mass Distribution of a Light Helicopter:
a – flight compartment space distribution;
b – fuel tanks arrangement



a

b

Fig. A.9.5. Load-Carrying Structure of a Light Helicopter
a – fuselage ventral part; b – engine mount system

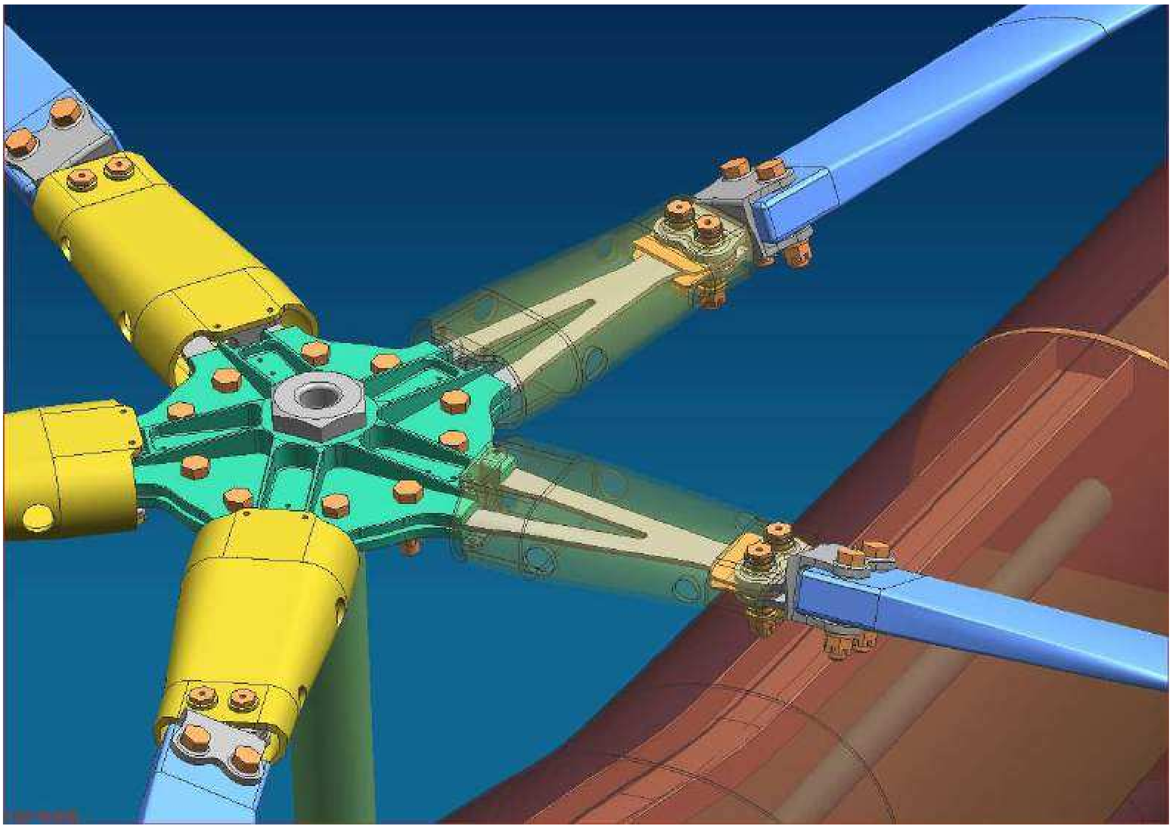


Fig. A.9.6. Load-Carrying Structure of Attachment Fitting
of a Blade-to-Hub Attachment

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