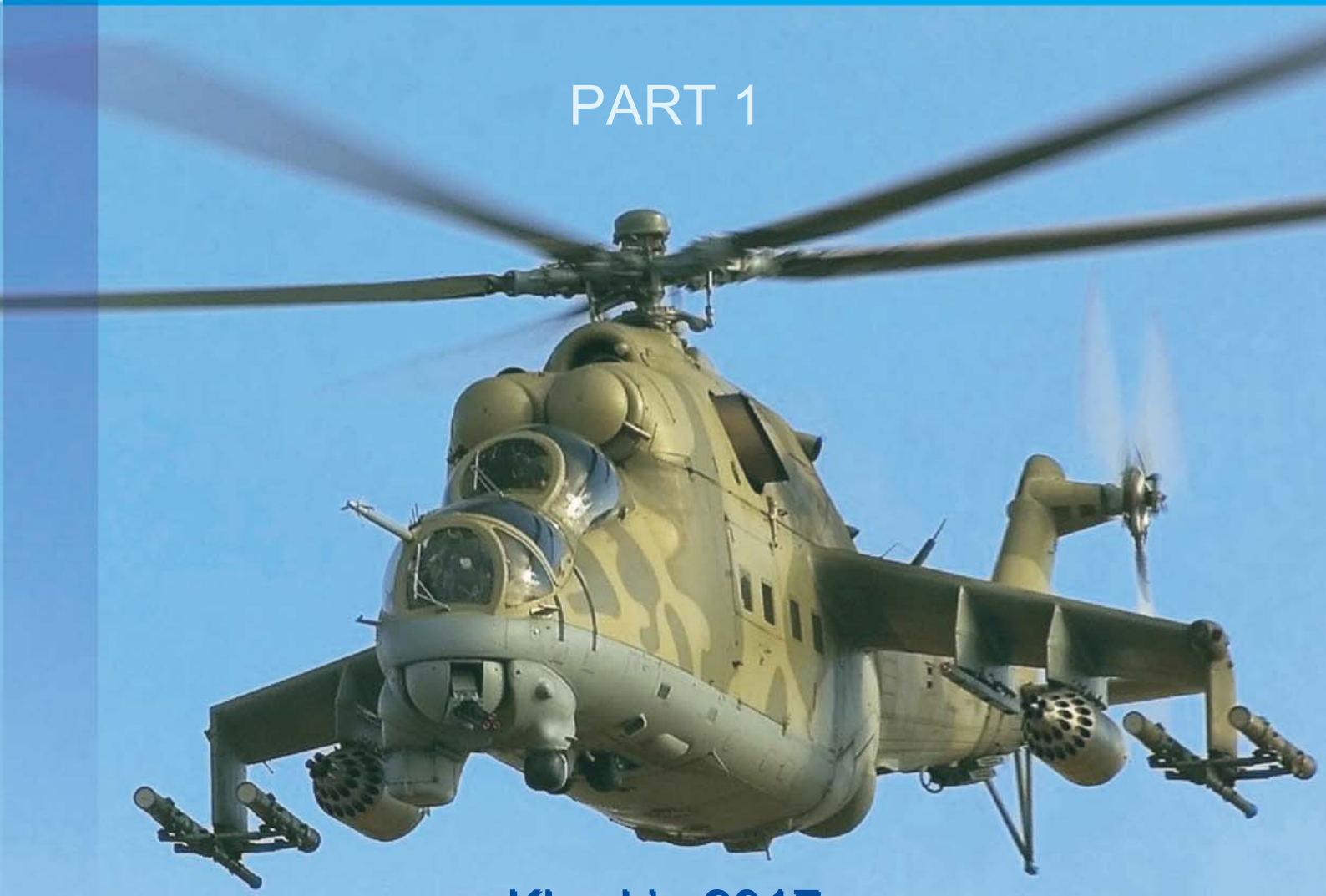


HEAVY SINGLE-ROTOR HELICOPTERS AND THEIR TRANSMISSIONS. DESIGN BACKGROUND

PART 1



Kharkiv-2017

MINISTRY OF SCIENCE AND EDUCATION OF UKRAINE
NATIONAL AEROSPACE N.E. ZHUKOVSKIY UNIVERSITY
“KHARKIV AVIATION INSTITUTE”

HEAVY SINGLE-ROTOR HELICOPTERS AND THEIR TRANSMISSIONS. DESIGN BACKGROUND

Part 1

Under the editorship of PhD, prof. V.S. Krivtsov

Kharkiv "KhAI" 2017

UDC 629.735.45.01(075.8)
BBL 39.54Я73
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Розглянуто метод загального проектування трансмісії важкого вертольота, технологію виготовлення і методи забезпечення ресурсу редуктора, особливості технологічної підготовки виробництва для монтажу трансмісії важкого вертольота, методику вибору оптичних і лазерних засобів монтажу і контролю трансмісії, типові операції монтажу і контролю трансмісії на базі оптичних і лазерних систем. Викладено методику розроблення керівництва з технічної експлуатації трансмісії (на прикладі вертольота Мі-26). Описано метод забезпечення надійності трансмісії вертольота при проектуванні і в експлуатації. Розроблено методологію інтегрованого проектування і моделювання вертольотів і їхніх трансмісій.

Для студентів спеціальностей "Літаки і вертольоти", "Технологія літако- і вертольотобудування" і фахівців в області проектування і виробництва вертольотів.

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Heavy Single-Rotor Helicopters and Their Transmissions. Design Background /
H43 A. G. Grebenikov, A. M. Gumenniy, A. I. Dolmatov. – Guide-book. – Kharkov: Nat. Aerospace Univ. "Kharkiv Aviation Inst.", 2017. Part 1. – 345 p.

Presented are the method of designing background of heavy helicopter transmissions, production technology and the methods of gearbox service life provision, features of preproduction planning for installation of heavy helicopter transmissions, the method of choice of optical and laser tools for transmission assembly and control, typical operations of transmission installation and control based on optical and laser systems. The technique for development of transmission maintenance manual (on an example of the Mi-26 helicopter) is given. The method of helicopter transmission reliability in the process of its designing and operation is described. Methodology of integrated design and simulation of helicopters and their transmissions has been developed.

For students of specialty "Planes and Helicopters", "Technology of Plane and Helicopter Manufacturing" and specialists in the design and manufacture of helicopters.

Ill. 260. Tables 18. Bibliography.: 98 references.

UDC 629.735.45.01(075.8)
BBL 39.54Я73

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INTRODUCTION

Every year the areas of application of helicopters are expanding, the volumes of their work are steadily growing. At the same time, the cost of producing and operating helicopters is already becoming noticeable on a countrywide scale. In this regard, it is necessary to carefully approach to the choice of parameters for newly designed helicopters, especially heavy aircraft.

The proposed textbook is devoted to the design of heavy single-rotor helicopters and their transmissions. The brief review of helicopters in the modern world, state and prospects of their development, aerodynamic configurations of helicopters and their analysis are given.

The paper outlines the basics of general design, including choice of efficiency criteria, aerodynamic design, calculation of main parameters, calculation of take-off weight of the helicopter in a whole and its units, including transmission elements. The issues of engine selection, design of layout and calculation of helicopter center of gravity position are considered. Taking into account the requirements of the rotorcraft airworthiness standards established for the rotor drive system (AP-29), methods for determining the loads acting on helicopter and its units and assemblies are described. Definition of aerodynamic and performance characteristics of a heavy single-rotor helicopter is also considered. After that, the methodology for designing the transmission is given. The textbook contains the analysis of transmission layout schemes, methods of designing of reducers (main reducer with high power transmission and high gear ratio, intermediate and tail reducers), transmission shafts and rotor shaft. Algorithms for designing such parts and assemblies as gears, clutches, bearings are given. Particular attention is paid to issues of increasing the transmission lifetime and ensuring its reliability. As part of an integrated approach, the information on technology of transmission assembly and development of transmission operation manual (on the example of the Mi-26 helicopter) is also given. The methodology for integrated design and simulation of helicopter and its transmission is given as part of generalizing section.

The authors hope that the book will be useful for students of aviation specialties and workers involved in the design of helicopters.

Chapter 1. HELICOPTERS IN MODERN WORLD, CURRENT STATE AND TRENDS OF DEVELOPMENT

100 years has passed from the first in the world helicopter manufactured by the brothers Breguet and professor Richet had took off, and then real lifting of the Cornu (France) helicopter. But the beginning of helicopter engineering belongs to the 40th years of the last century.

Over the years the helicopter engineering has arisen in the Soviet Union (then in CIS), United States, and also in Great Britain, France, Italy, Germany, Japan and other countries as a part of the aviation industry and is under successful development, combining experimental design offices, production plants and research institutes with strong production and experimental bases. During this time the helicopter received the widest application in the civilian sector, and especially in the armed forces, where production of military helicopters was comparable to the production of military airplanes. Military aviation fleet changed qualitatively, in which number of military helicopters of various types exceeds number of military airplanes.

As a rule helicopters are dual-purpose: they are used for military and civilian applications as a lightweight multi-purpose, medium and heavy transport aircraft and flying cranes. Passenger helicopters got limited use. But specialized military helicopters, including combat and transport-fighting, anti-submarine, rescue, AWACS, reconnaissance and communication helicopters are extensively used. In turn, the military and civilian helicopters are divided into the ground-based and ship-based helicopters.

Total number of helicopters manufactured around the world, according to updated data for 1997 [1], is close to 110000, with more than 24000 helicopters produced in the USSR and 6200 - under license from Russia in China and Poland, that is total production of helicopters in CIS countries has exceeded 30000. The largest production of helicopters of more than 55000 was achieved in the US and all other countries account for 23000 helicopters, including production in France – 8900, in Italy – 4750, UK – 2400, Germany – 2100, Japan 1600, Canada – 950, China – 600, India – 460, Indonesia – 400, Romania – 360, Brazil – 270, and 1000 in Argentina, Yugoslavia, Taiwan, Switzerland, Austria, Singapore. For more information about serial production of helicopters refer to Table 1.1 [1], which shows distribution of mass production over countries and indicates all manufacturers and helicopters they produce, their take-off weight and capacity. The table shows configurations of helicopters, and state of their production. If this data would be added with 10000 – 12000 helicopters that were supposed to be manufactured for the period 1998-2007 according to predictions of the world helicopter market, so total number of helicopters, produced in the world is 120000.

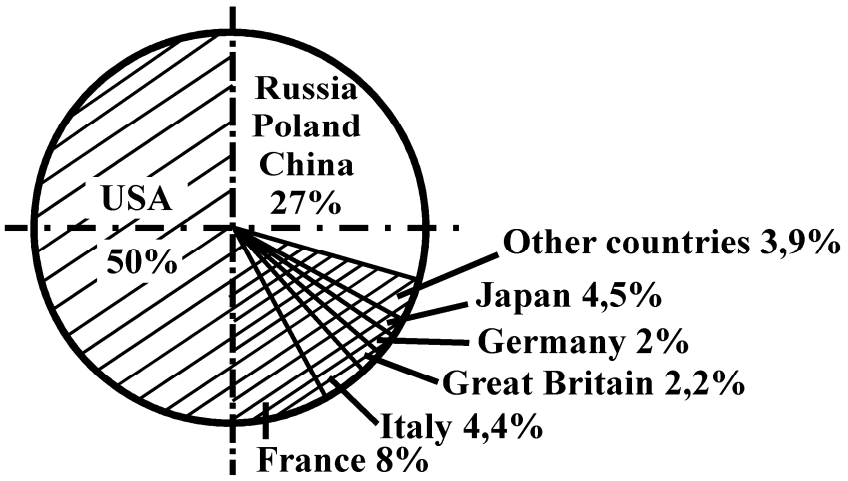
Table 1.1 also comprises number of helicopters manufactured under license, showing that only the United States and Russia began large-scale production of helicopters, while in other countries, even those developed such as UK, France, Italy and Germany, which have built their original helicopters, large-scale production began under license got from the United States. In future, these countries have provided licenses for serial production of own helicopters to other countries: Australia, Argentina, Brazil, India, Indonesia, Spain, China, Singapore, Taiwan and Japan. Production under license of Russian helicopters was carried out in China, Poland and Romania.

Similar to airplane engineering where monoplane configuration has the greatest

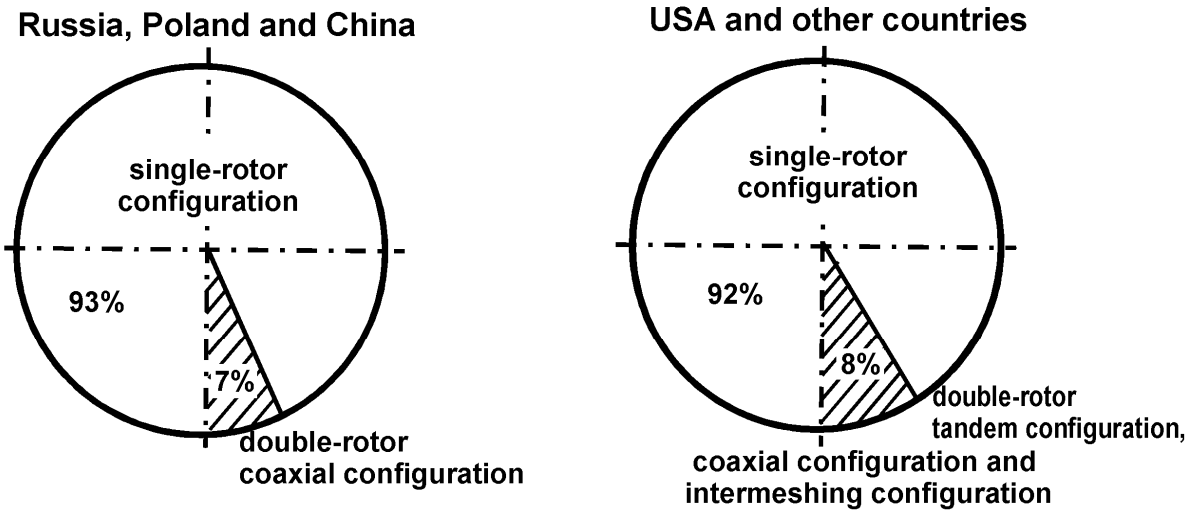
application after years of research and development of various configurations, among helicopters those dominate based on single-rotor configuration with tail rotor, which is realized in 93% of all manufactured helicopters. The remainder is accounted for co-axial and tandem, side-by-side and intermeshing configurations.

When analyzing distribution of helicopters with regard to their carrying capacity and number of seats given in Table 1.1 it is evident that light 2-3-seat and 4-5-seat helicopters make up a large part of all manufactured helicopters, about 58 %, while in Russia they constitute only 13 %; 6-16-seat helicopters constitute 36 % and in Russia about 45 %, 20-40-seat aircraft abroad constitute less than 6 %, and in Russia more than 38 %, that is almost 7 times more. Heavy helicopters with 50-60 seats such as Mi-6 and Mi-26 were manufactured only in the USSR.

Distribution of helicopters manufactured in the USSR and other countries with regard to configuration and carrying capacity is shown in Figs 1.1 and 1.2.



a



b

Fig. 1.1. Distribution of Helicopter Production:
a – by countries, b – by configurations

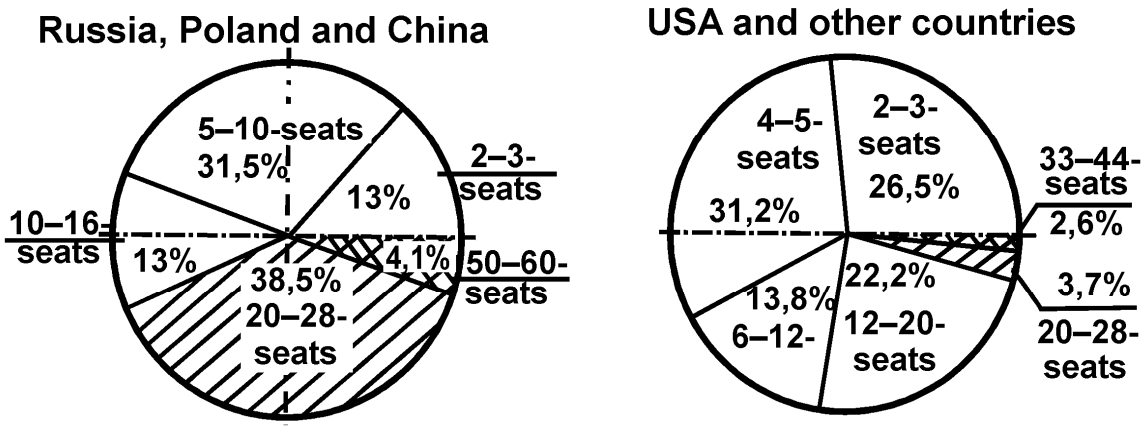


Fig. 1.2. Helicopter Distribution by Carrying Capacity

1.1. SERIES PRODUCTION OF HELICOPTERS AROUND THE WORLD (by statistics data given for middle of 1997) (Table 1.1.) [1]

Table 1.1

Country, company and helicopter name	Configuration	Numb. of seats	Maximum takeoff mass, tons	Years of production	Number of manufactured helicopters	State of production
USSR – Russia Mil Design Office						
Mi-1	Single-rotor	3	2.55	1950-1965	2694	Series production is stopped
Mi-2	Was manufactured under license in Poland					
Mi-4	Ditto	14-18	7.55	1952-1979	3307	Ditto
Mi-6	Ditto	65	42.5	1959-1981	926	Ditto
Mi-8	Ditto	24-28	11.1	From 1965	8160	In series production
Mi-10	Ditto	28	43.5	1967-1975	55	Series production is stopped
Mi-14	Ditto	12	14	1975-1986	273	Ditto
Mi-17	Ditto	26-30	13	From 1982	2900	In series production
Mi-24	Ditto	2-10	11.5	From 1970	3334	Ditto
Mi-26	Ditto	80	56	From 1980	280	Ditto
Mi-28	Ditto	2	11.5	Pilot version	5	Under flight tests
Mi-34	Ditto	4	1.45	From 1993	15	In series production
					21949	
«Kamov»						
Ka-8 and Ka-10	Two rotors, coaxial	1	0.4	1952-1954	15	Not produced in series
Ka-15 and Ka-18	Ditto	2-3	13	1953-1964	450	Ditto
Ka-25	Ditto	2	7.3	1964-1975	275	Ditto
Ka-26	Ditto	2-8	3.25	1964-1977	816	Ditto
Ka-126 and Ka-226	Ditto	2-8	3.1	Experimental	10	Under tests
Ka-27, -29 and -31	Ditto	3	11.5-12.5	From 1977	350	In series production
Ka-32	Ditto	3-18	11-12.6	From 1985	132	Ditto

Ka-50/52	Ditto	2	10.8	From 1992	10	Ditto
					2058	

Table 1.1 (continued)

Country, company and helicopter name	Configuration	Numb. of seats	Maximum takeoff mass, tons	Years of production	Number of manufactured helicopters	State of production
Yakovlev Design Office						
Yak-1A	Two rotors, tandem	20	14	1952-1955	40	Not produced in series
		-	Totally over all Russian Design Offices 24047			
USA						
«Sikorsky»						
R-4 and R-6	Single-rotor	2-3	1.2	1943-1946	441	Not produced in series
S-51hS-52 (H-5h H-18)	Ditto	4-5	2.6	1947-1951	415	Ditto
S-55 (H-19)	Ditto	6-8	3.6	1951-1960	1282	Ditto
S-56 (H-37)	Ditto	26	14.6	1956-1960	156	Ditto
S-58 (H-34)	Ditto	12-16	5.9	1956-1964	1821	Ditto
S-61 (H-3)	Ditto	22-28	8.5-9.8	1962-1980	796	Ditto
S.62 (H-52)	Single-rotor	10-12	3.7	1962-1969	151	Not produced in series
S-64 (H-54)	Single-rotor, flying crane	9	19	1964-1971	100	Ditto
S-65 (H-53)	Single-rotor	24-38	19-22	1967-1976	522	Not produced in series, under modernization
S-80 (H-53E)	Ditto	44-45	31.6	From 1979	222	Ditto
S-70 (H-60)	Ditto	12-20	8-10	From 1978	2000	In series production
S-76 (H-76)	Ditto	8-12	4.4-5.3	From 1979	460	Ditto
1RAH-66	Ditto	2	5.8-7.9	From 2001	8 experimental aircraft	Under tests, 1292 helicopters ordered
			8372			
«Bell Helicopter»						
Bell 47 (H-13)	Single-rotor	2-3	1.35	1946-1975	5127	Not produced in series
Bell 61 (HS1-1)	Two tandem rotors	10-12	7	1953-1955	78	Ditto
Bell 204 (UH-1A)	Single-rotor	8-10	3.7-4.3	1958-1970	2098	Ditto
Bell 205 (UH-ID)	Ditto	10-14	4.3-4.9	1963-1984	10732	Not produced in series, under modernization
Bell 212/412 (UH-1N)	Ditto	12-16	5.1-5.4	1970-1991	723	«
Bell 214	Ditto	16-20	7.95	1975-1991	512	Series production is transferred to Canada
Bell 206 (OH-58)	Single-rotor	4-5	1.4-1.85	1967-1991	7305	Series production is transferred to Canada, under modernization
Bell 209 (AH-1G, Q, S, F)	Ditto	2	4.3-4.6	1967-1979	1560	Not produced in series, under modernization

Table 1.1 (continued)

Country, company and helicopter name	Configuration	Numb. of seats	Maximum takeoff mass, tons	Years of production	Number of manufactured helicopters	State of production
Bell 209 (AH-1J, T and W)	Ditto	2	4.5-6.4	1971-1991	350	Ditto
Bell 222/230	Ditto	6-11	3.8	1980-1991	200	Series production is transferred to Canada
					26672	
«Boeing-Vertol-Pyasetsky»						
PV.3/44 (H-21)	Two tandem rotors	12-16	6	1947-1966	715	Not produced in series
PV. 14/18 (H-25)	Ditto	4-5	2.6	1951-1955	390	«
V.107 (H-46)	Ditto	25	9-10.5	1958-1970	677	Not produced in series, under modernization
V.114 (H-47)	Ditto	33-44	15-21	1962-1978	805	Ditto
					2597	
«Brantly – Haino»						
B-2/B-305M	Single-rotor	1-2.	0.8-1.3	1959-1974	400	Not produced in series
«Jirodan»						
QH-50	Two coaxial rotors	Drone	1	1962-1968	800	Not produced in series
«Kamai»						
HTK/HOK/H-43	Two intermeshing rotors	3-8	1.4-4.1	1952-1959	387	Ditto
H-2	Two intermeshing rotors	6-12	5-6.1	1960-1974	242	Not produced in series, under modernization
«K-Mako»	Two intermeshing rotors, flying crane	2	5.2	1994-present time	6	In series production
					635	
«Robinson»						
R-22	Single-rotor	1-2	0.6	From 1979	2500	In series production
R-44	Ditto	2-4	1.06	From 1993	270	Ditto
					2770	
«Rogerson» – «Fairchild» – «Himer» H.12 (H-23)	Single-rotor	2-3	1.25	1952-1966	2391	Not produced in series
FH.1100 (H-5)	Ditto	4-5	1.45	From 1965	274	In series production
					2585	
«Enstrom»						
F-28/F-280	Single-rotor	2-3	1	From 1966	1000	In series production
«Hughes» – «McDonnell» «Douglas»						

Table 1.1 (continued)

Country, company and helicopter name	Configuration	Numb. of seats	Maximum takeoff mass, tons	Years of production	Number of manufactured helicopters	State of production
H-269/300 (H-55)	Single-rotor	1-2	0.9	1960-1978	3300	Not produced in series
MD 500 (H-6)	Ditto	4-5	1.35	From 1960	4940	Not produced in series, under modernization
MD 520/600	Ditto	5-8	1.75-2.7	From 1991	80	In series production
AH-64	Ditto	2	9.5	1979-1994	937	Not produced in series, under modernization
					9257	
«Cessna»	Single-rotor	2-3	1.35	1956-1960	28	Not produced in series
				Totally over all USA Companies	55,116	
Great Britain «Westland»						
Bell 47	Single-rotor	2-3	1-3	1964-1971	100	Not produced
WS-51 «Widgen»	Ditto	4-5	2.6	1948-1963	139	Ditto
WS-55 «Wirluind»	Ditto	6-8	3.6	1955-1960	400	Not produced, removed from army
«Wassex»	Ditto	10-12	5.7	1958-1970	376	Not produced
«Si King» and «Commando»	Ditto	22-28	9.75	1966-1994	328	In series production
«Skaut» and «Wasp»	Ditto	4-5	2.9	1963-1968	293	Not produced
«Links»	Ditto	6-12	3.75	From 1971	440	In series production
					2076	
«Bristol» 171	Single-rotor	4-5	2.45	1949-1959	178	Not produced, removed from army
«Sikamor» 173/192	Double-rotor tandem	12-25	8.6	1960-1963	26	Ditto
					204	
«Saunders-Ro» «Skiter»	Single-rotor	1-2	1	1950-1957	130	Not produced, removed from army
				Totally over all Great Britain companies:	2410	
France «SNCASE»/ «Aerospatiale» /«Eurocopter» S-55 (under license)	Single-rotor	6-8	3.6	1952-1956	86	Not produced, removed from army
S-58 (under license)	Ditto	10-12	5.6	1958-1965	166	Ditto

Table 1.1 (continued)

Country, company and helicopter name	Configuration	Numb. of seats	Maximum takeoff mass, tons	Years of production	Number of manufactured helicopters	State of production
«Gin»	Single-rotor	1-2	0.8	1953-1957	178	Not produced, removed from service
«Aluett» I and II	Ditto	4	1.6	1952-1975	1305	In series production
«Aluett» III	Ditto	6-8	2.12	1962-1982	1398	Ditto
«Lama»		4-5	1.95	1970-1996	407	«
«Super Frelon»		30	12.5	1965-1966	97	Not produced
«Puma»		12-18	6.4-7.4	1968-1980	714	«
«Super Puma»		21-25	8.6-9	From 1980	400	In series production
«Gasel»		4-5	1.8	1969-1990	1223	Not produced
«Dofin» and «Pantera»		10	3	From 1975	565	In series production
«Ecureuil» AS.350		4-5	1.9	From 1978	1816	«
«Twin Star» AS.355		5-6	2.3	From 1980	550	«
Totally over all France Companies					8905	
Poland «PZL»						
Mi-2 (under license)	Single-rotor	8-10	3.7	1965-1992	5440	Not produced
W-3 «Sokol»	Ditto	12	6.4	From 1993	70	In series production
					5510	
Italy Augusto						
AB-47 (under license)	Single-rotor	2-3	1.35	1956-1974	1120	Not produced
AB-206 (under license)	Ditto	4-5	1.45	From 1965	1150	Ditto
AB-204/205/212/412 (under license)	Ditto	6-12	4.3-5	From 1963	1271	In series production
ASH-3D (under license)	Ditto	22-28	9.8	1967-1981	144	Not produced
NH-500 (under license)	Single-rotor	4-5	1.35	FROM 1972	230	In series production
A. 109	Ditto	8	2.75	From 1976	550	Ditto
A. 129	Ditto	2	3.95	From 1986	45	Ditto
					4510	
Meridionally CH-47C	Double-rotor Tandem	35-44	15-21	From 1970	200	In series production
Silvercraft SH-4	Single-rotor	2-3	0.85	1968-1974	45	Not produced
Totally over all Italy Companies					4755	

Table 1.1 (continued)

Country, company and helicopter name	Configuration	Numb. of seats	Maximum takeoff mass, tons	Years of production	Number of manufactured helicopters	State of production
Germany						
Dornje Bell UH-1D	Single-rotor	6-12	4.3	1964-1971	352	Not produced
VFW-Fokker						
S-65 (H-53)	Ditto	24-38	19	1971-1978	112	Ditto
Indonesia (under license)						
NB412	Single-rotor	12-16	5.4	1984-1995	18	Not produced
NSA 330/332	Ditto	21-25	6.4-7.4	1981-1995	27	Ditto
NBo.105	Ditto	4-6	2.3	From 1976	140	In series production
NBK.117	Ditto	6-10	3.2	From 1985	220	
					405	
Romania						
IAR –Eurocopter (under license)						
Aluett III	Single-rotor	4-5	2.2	1968-1986	200	Not produced
IAR 330 Super Puma	Ditto	12-18	6.4-7.4	From 1977	160	In series production
					360	
Brazil						
Gelibraz (under license)	Single-rotor	4-5	1.95	1968-1986	52	Not produced
Aluett/Lama AS.350/550	Ditto	4-5	1.9	From 1980	215	In series production
					267	
Yugoslavia (under license)						
SA.340 и Gazelle	Single-rotor	4-5	1.8		220	Not produced
South Korea (under license)						
MD. 500	Single-rotor	4-5	1.36		198	Not produced
Spain						
CASA (under license)						
CASA Bo.105	Single-rotor	4-6	2.3	1974-1980	143	Not produced
Taiwan (under license)						
Bell 205	Single-rotor	10-14	4.3-4.9		118	Not produced
Argentina (under license)						
MD.500	Single-rotor	4-5	1.36		78	Not produced
Switzeland (under license)						
Aluett III	Single-rotor	6-8	2.2		60	Not produced

Table 1.1 (continued)

Country, company and helicopter name	Configuration	Numb. of seats	Maximum takeoff mass, tons	Years of production	Number of manufactured helicopters	State of production
Philippines (under license)						
Bo. 105	Single-rotor	4-6	2.3		60	Not produced
Australia (under license)						
Bell 206	Single-rotor	4-5	1.36		56	Not produced
Singapur (under license)						
AS.332	Single-rotor	21-25	6.4-7.4		17	Not produced
Totally over all countries and companies abroad					83953	

Modern helicopter engineering is represented with the fourth and fifth generations of helicopters.

Helicopter engineering features with the following helicopter characteristics:

– greater flight speed due to reducing drag and creating high-performance rotors capable of using powerful engines at high speed;

– increased relative weight of payload due to improving MR efficiency in hovering and reducing helicopter empty weight as a result of applying new construction materials;

– reduced fuel consumption per kilometer due to improved fuselage aerodynamics and reduced specific fuel consumption;

– higher efficiency of performing flight missions, increased survivability and reliability due to introduction of structures made of composites;

– reduction of the pilot workload through comprehensive implementation of electronic systems;

– reduced number of flight and technical personnel.

1.2. HELICOPTERS OF THE FOURTH AND FIFTH GENERATIONS

Helicopters of the fourth and fifth generations differ with advanced design widely using composite materials, improved aerodynamic shape, providing reduction of parasitic drag and increase of equivalent lift-to-drag ratio. In Russia a new generation of helicopters includes Mi-26, Mi-28, Mi-34 and Mi-38 created by the Mil Experimental Design Office and Ka-50 Ka-62 created by the Kamov Experimental Design Office (evolution of combat helicopters, including Mi-28 and Ka-50 is given in a separate section). It should be noted that distinguishing of helicopter generations is largely arbitrary. The most successful achievement in helicopter engineering is creation of heavy Mi-26 aircraft.

The Mi-26 arrangement was chosen the same as that of the Mi-6, but its dimensions are smaller than of the Mi-6. The fuselage with the same size and weight as

of Mi-6 has cargo compartment, which volume is twice larger that of the Mi-6 (Fig. 1.3), and which is designed to carry twice more load and equipped with devices to facilitate handling. Overall dimensions of the cabin and load-carrying capacity of Mi-26 allowing transportation of 80 ... 90 % of the military equipment and cargos of motorized division.

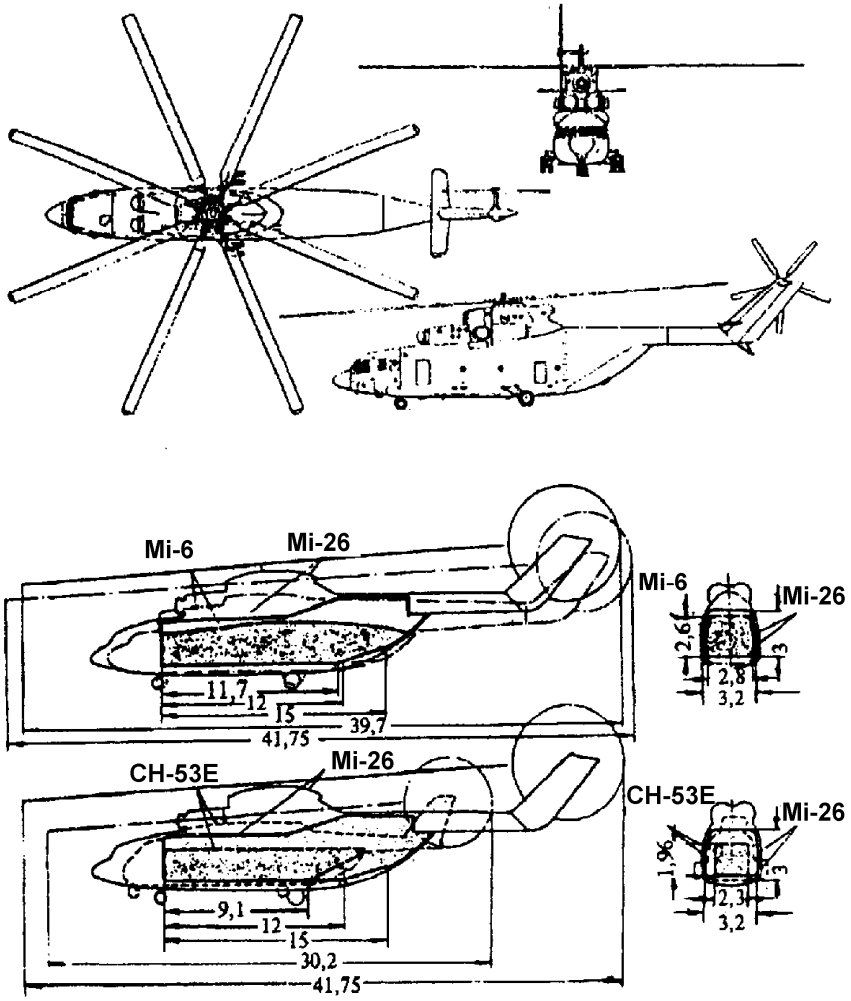


Fig. 1.3. Mi-26 and CH-53E Helicopter Configurations

GTE D-136 with free power turbine having take-off power of 10000 hp was chosen for the Mi-26 power plant. It is not only the most powerful in the world, but characterized with low specific gravity, low specific fuel consumption and low levels of noise and emission of air pollutants.

The main difficulty was introduced by the problem of transmitting an enormous power to the main rotor, which has been successfully solved in the design office by means of developing main gearbox using multiengine non-planetary scheme and having somewhat greater weight than the gearbox of the helicopter Mi-6, with twice the transmission capacity.

Much attention was paid to choosing optimal parameters of the rotor. Designed

eight-blade main rotor having diameter of 32 m allows for much greater thrust than five-blade main rotor having diameter of 35 m of the Mi-6 helicopter, and studies aimed to optimizing the aerodynamic design of the blades have provided significant increase in the rotor efficiency.

Improved aerodynamic contours of the helicopter significantly reduce drag and contribute to essential reduction of fuel consumption per kilometer, as a result – twice higher equivalent transport performance was achieved compared to Mi-6 (Fig. 1.4). Mi-26 helicopter carrying capacity is twice higher than of the largest American helicopter Sikorsky CH-53E (Fig. 1.5).

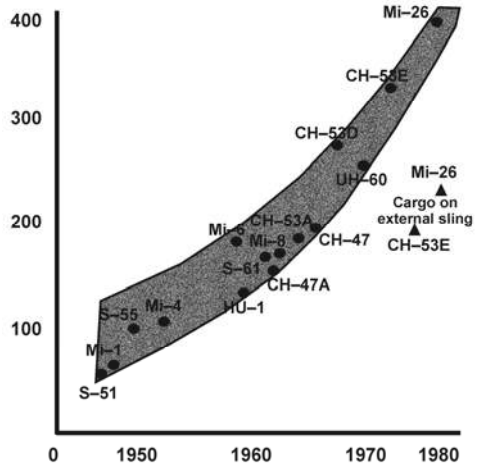


Fig. 1.4. Equivalent Transport Performance of Helicopters

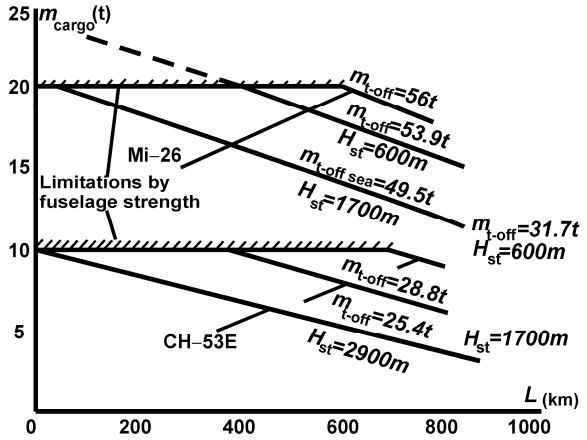


Fig. 1.5. Helicopter Carrying Capacity vs Flight Range

The first flight of Mi-26 experimental helicopter was done in 1977. According to its flight characteristics Mi-26 helicopter was significantly superior comparatively to domestic and foreign helicopters, as evidenced by 14 world records, including the absolute record of lifting 25 tons to altitude of 4,100 meters, 20 tons – to altitude of 4600 m, 15 tons – to 5600 m and 10 tons - to 6400 meters and reaching altitude of 2000 meters with an airborne weight of 56768 kg, established on the 2-4 February, 1982.

The Mi-26 serial production began in 1984. It was built more than 280 helicopters for civilian and military applications, 40 of which were exported to different countries. The Mi-26 was successfully used to transport goods, people and vehicles in the army and the national economy. They are used for the same purpose in the United Nations played an important role in the aftermath of the Chernobyl accident.

A number of the Mi-26 modifications, including transport, passenger with cabin having 70 seats and the Mi-26 «Salon» having higher level of comfort, fire helicopter and flying crane Mi-26T with the operator cabin under the fuselage behind the front landing gear, or on the rear cargo ramp; Mi-26T can carry loads of 20 tons on the external sling.

Light multi-role helicopter Mi-34 (see. Fig. 1.6) with one piston engine M-14B-26 was developed to substitute the well-proven in operation, but obsolete Mi-1. It flew for the first time in 1986. The helicopter features with the extensive use of composite materials, good aerodynamic shape and high maneuverability: the helicopter (first among Russian helicopters) had demonstrated performance of the Nesterov loop. There were developed versions of the Mi-34 with two rotary-piston engines and Mi-34 with the American GTE Allison 250. It was started to produce in small series.

In 1987 development of new multi-purpose helicopter Mi-38 (Fig. 1.6) with two turbine engine was started to substitute the Mi-8 helicopters. New helicopter should exceed the Mi-8 2-3 times in transport performance and fuel efficiency, and differs with enhanced safety due to the greater reliability of the main units and redundancy of the main systems due to greater power-to-weight ratio and improved equipment. It was assumed that load carried on the external sling will be of 6 tons, which allows the use of Mi-38 helicopters instead of Mi-6 aircraft. The helicopter will be produced as transport, passenger and sanitary versions. Development of the Mi-38 is conducted in cooperation with the «Eurocopter» consortium, but it is delayed much: a construction of the helicopter pilot version and its first flight were postponed several times, and only helicopter models were shown at exhibitions.

Mil Moscow Helicopter Plant has developed a number of projects of new helicopters, including light multi-role three-seat Mi-52 with a rotary-piston engine and passenger 30-seat helicopter Mi-58 (see. Fig. 1.6) with two GTE, which is a modification of the Mi-28 with its five-blade main rotor and dynamic systems, as well as heavy transport helicopter Mi-46T to substitute the Mi-6 aircraft, similar in layout and design to the Mi-26, but smaller; The flying crane Mi-46K was proposed as an option. Models of these helicopters have repeatedly demonstrated at various aerospace trade shows, as evidence of continuing Mil design office work on creation of new helicopters, but the practical implementation was not reached due to lack of funding.

Light multi-role 9-seat helicopter «ANSAT» with two GTE having characteristics close to the Mi-54 aircraft, but featuring with simplified design with ski landing gear, has been developed by the Design office of the Kazan Helicopter Plant (Russia).

The Kamov Design office also developed a number of new generation helicopter projects. The most advanced project is the Ka-62 (Fig. 1.7) having not coaxial twin-rotor, but single-rotor design with a fan in the vertical tail and two GTEs, which arrangement is similar to the French helicopter AS.365 “Dofen”. The Ka-62 is being developed with the participation of foreign companies and must be certified by the American airworthiness standards. Together with the «Augusta» Italian firm the Ka-64 modification having tail rotor instead of a fan was developed. Construction of Ka-62 multi-role military modification with advanced equipment and weapons was shown at the exhibition «MAKS-97». It is expected that helicopter will be put in operation in 2000.

Kamov design office had developed in 1992 the project of light 4-5-seat multi-purpose helicopter Ka-115 (Fig. 1.7) with a twin-rotor co-axial system and advanced design of the blades and hubs and all-composite airframe.

In the US, the major new military programs of the rotary-wing aircraft is the fourth generation combat helicopter reconnaissance program RAH-66 «Comanche» and helicopter-airplane Bell-Boeing V-22 «Osprey» that become not only the most prestigious programs of the armed forces, but also the most expensive.

In the 1990s, the United States started a number of programs to modernize military helicopters of the third generation, such as UH-60L helicopters with more powerful turbine engine and an improved design and equipment, and the development of advanced civil helicopters, which should also include light multipurpose helicopters Bell 222 and Sikorsky S-76.

The «Bell» company has developed Bell 222, Bell 230, Bell 430 and Bell 407 aircraft.

New 8-seat Bell 427 helicopter with two gas turbine engines and large-capacity cabin, developed by the Canadian branch of the «Bell» company in cooperation with the South Korean company «Samsung» became an evolution of the Bell 407 helicopter. The «Bell» company signed an agreement with China on production of 200 Bell 206 helicopters in East China under license.

More success came in the share of the S-76 helicopter (Fig. 1.8), which became the first helicopter of the «Sikorsky» company specially designed for civilian use: to serve the offshore oil fields, as well as the administrative and multipurpose transportation for 6-8 passengers.

Helicopter S-76 with two GTEs has the most perfect aerodynamics among all the Sikorsky helicopters having the aircraft type fuselage with a retractable landing gear. In developing the S-76 helicopter a number of improvements implemented in the final design thanks to operation of S-70 helicopters (UH-60) were used, which makes it a next-generation helicopter. The helicopter S-76 set a number of world speed record, including the absolute world speed record of 345.74 km/h at altitude of 500 km in 1982, had demonstrated aerobatics. Since 1978 more than 460 helicopters delivered, also multi-purpose military helicopter H-76 was designed; it was undergoing flight tests, but has not received the application.

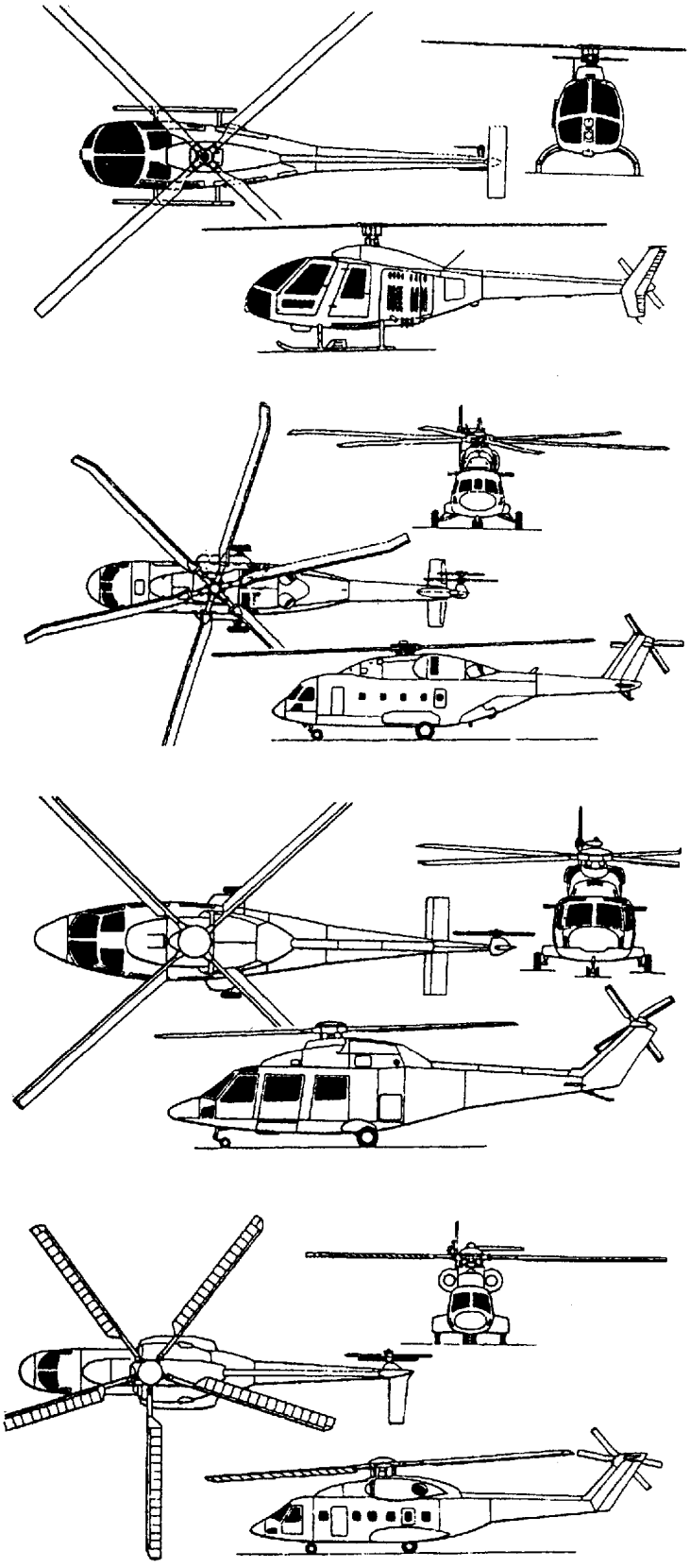


Fig. 1.6. Mi-34, Mi-38, Mi-54 and Mi-58.
New Helicopter Designs of Mil Design Office

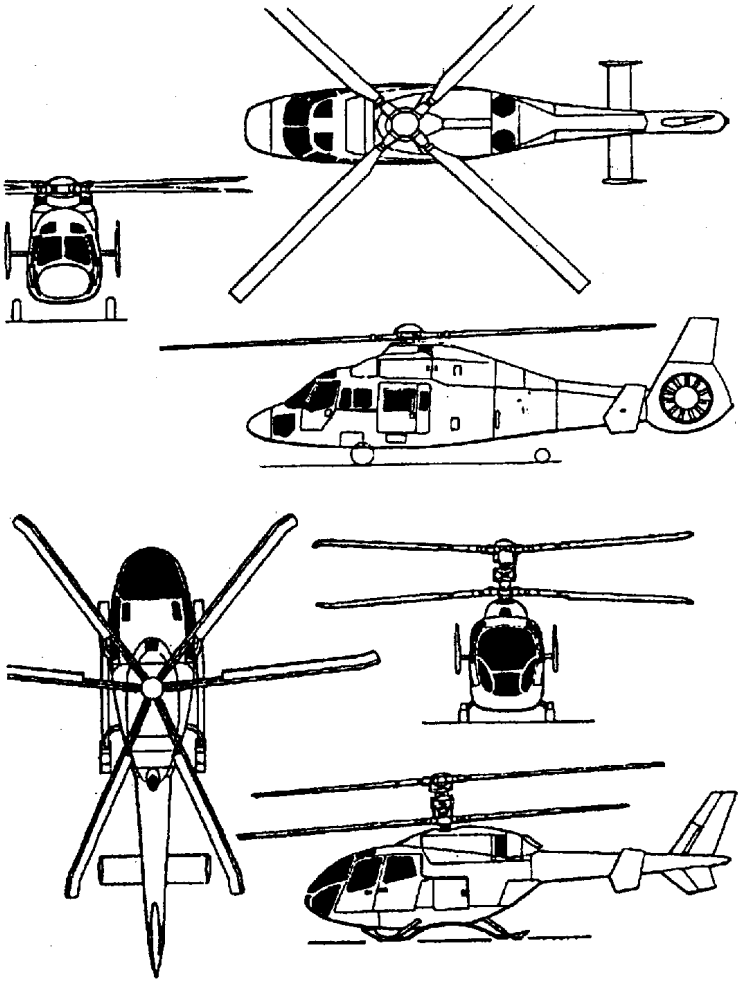


Fig. 1.7. Ka-62 and Ka-115 Helicopters of Kamov Design Office

An evolution of the S-76 helicopters in Japan has resulted in light multipurpose helicopter MH 2000, developed by the company «Mitsubishi Heavy Industries», with the use of carrier systems and assemblies of the S-76 helicopter. MH 2000 helicopter became the first civilian helicopter developed in Japan. It is equipped with two turbine engines, four-blade rotor, «fenestron» fan and ski landing gear, the cabin can accommodate up to 10 passengers. The helicopter made its first flight in July 1996, certified in Japan, mass-produced for use in the domestic market of Japan.

It is a new generation of multi-purpose medium and transport helicopter Sikorsky S-92 (Fig. 1.9), which will substitute the S-70 helicopter (UH-60), and to be competitive with the new European NH-90 helicopter. The S-92 helicopter started to be developed in 1990, in 1992 at an exhibition in Farnborough the layout of the helicopter was first shown and reported that it would be built in 1994. However, its development, for which 600 million dollars was spent, delayed and demanded attracting European and Asian companies.

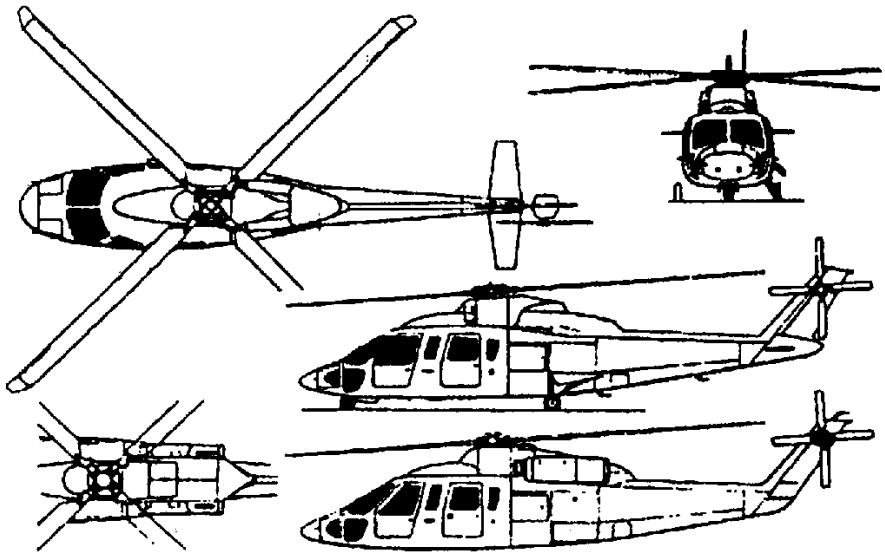


Fig. 1.8. Light Multi-Role Helicopter Sikorsky S-76

The company estimates the potential market for the S-92 helicopters in the 2000-2019 years as 5000 helicopters, helicopter estimated price – 12.5 million dollars. However, it should be noted that the S-70 helicopter (UH-60), which production was started in 1977, was built in series of 2000 helicopters and it is possible to produce several hundred of these helicopters. Therefore, the statement is too exaggerated figure. Finally, we must also take into account that in this class the most common helicopters are Mi-8 and Mi-17, which are produced in series of 11,000 helicopters and well-proven in operation in many countries.

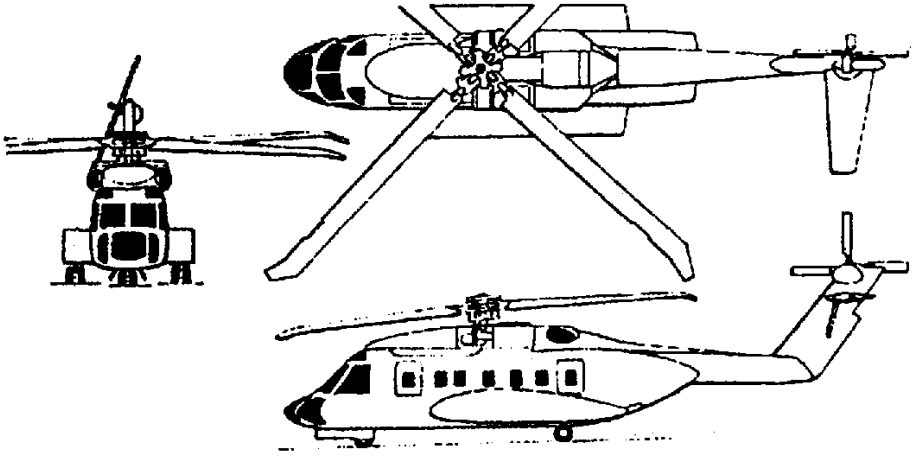


Fig. 1.9. Sikorsky S-92 Helicopter

The size of the S-92 is slightly larger than of the S-70; it can carry 19–22 passengers, in the military version it is equipped with cargo door at the rear of the fuselage. The helicopter features with wide use of composite materials and a number of improvements, approved by the company: the blade tips are swept and bent

down like of the UH-60L helicopter for the rotor higher efficiency; the tail rotor is placed up on the rather large vertical tail and deflected sideward to create thrust vertical component; GTE have an emergency power mode similar to the UH-60L helicopter.

Trying to simplify the design of single-rotor helicopter and not to use tail rotor, the «Hughes» company even in 1975 began to study NOTAR (No Tail Rotor) system, intended to balance the reactive torque and directional control. The «Hughes» company, which later became integrated with the «McDonnell Douglas» company considered that such a system would eliminate the drawbacks of the tail rotor: power consumption to the tail rotor drive, additional weight of the structure, high level of vibration and noise, complexity of maintenance and the risk of touching the obstacles by tail rotor and danger to maintenance staff.

In 1980, the «Hughes» company has developed an experimental helicopter with the NOTAR system based on OH-6A serial helicopter. The results of research and flight tests were used to develop a family of helicopters with the NOTAR system, being modifications of the OH-6A helicopter. Light multi-purpose five-seat helicopter MD 520N McDonnell Douglas with one GTE and NOTAR system made its first flight in May 1990, was certified in October 1991 and attracted a lot of interest in the market chock-full of light multi-purpose helicopters, not confirmed, however, by the demand; only 80 helicopters were built.

Later the new-generation MD 900N “Explorer” six-seat helicopter with the NOTAR system and two GTE was developed; 10 experimental helicopters were built; flight tests took place in 1992-1994. Mass production began in 1994, 45 helicopters were built for search and rescue services in Japan, South Korea and the UK police.

In Europe in the late 1970s - early 1980s also a series of new programs of the fourth generation of helicopters with advanced design and performance were began to develop. Even in 1975 the UK Ministry of Defense has started the study of a new helicopter to substitute the «Sea King» helicopters and in 1980 the «Westland» British company has teamed up with the «Augusta» Italian company and the «Fokker» Dutch company forming the «Europe Helicopter Industry» consortium (EHI) for joint development of a new three-engine helicopter EH-101 project on the basis of multi-purpose helicopter of the English WG.34 and Italian experimental three-engine transport helicopter A-101G in three versions: the sea deck for ASW, multipurpose and transport with rear cargo door and civil passenger (Fig. 1.10).

EH-101 helicopter has good aerodynamic shape, equipped with retractable landing gear, five-blade main rotor with hinged blades having elastomeric joints; improved blade design of the composite materials with variable relative thickness and twist lengthwise, and swept tips of complex configuration, designed by the «Westland» program BERP and used for the «Super Lynx» helicopter.

The potential market is estimated of 700 - 800 military and civil helicopters.

However, only 82 helicopters were ordered including 44 «Merlin» anti-submarine helicopters for the British Navy and 22 multi-purpose search-and-rescue helicopters for the British Aviation Forces and 16 anti-submarine and transport helicopters for the Italian Navy. Possible customers are Canada (which was going to order 50 helicopters) and Saudi Arabia. The civilian version of the «Gelyliner» helicopter was designed to deliver 30 passengers for servicing oil rigs at sea. It is supposed to use a wing on the helicopter to increase speed and range.

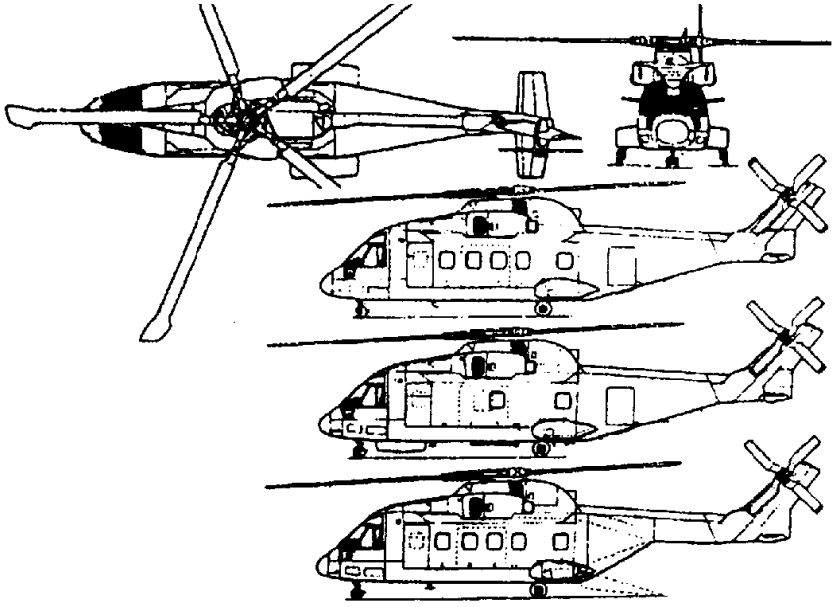


Fig. 1.10. EH-101 Helicopter

Somewhat later the EH-101 helicopter development according to new program was started in Europe – new multipurpose tactical transport helicopter for the Army and Navy of NATO in the 1990s, designated as NH-90 (NATO Helicopter). After long period of researching capabilities of the new helicopter began in 1983 the NHI consortium (NATO Helicopter Industry) was established in 1992, which includes the «Eurocopter France», «Eurocopter Deutschland» and «Augusta» companies.

The «Westland» firm also participated in the development program, but when had become the leader in the EH-101 program it refused to participate in the NH-90 program. During protracted development the program repeatedly revised, especially in terms of supply and purchase volume. It is assumed that Germany can buy 243 NH-90 helicopters, France – 160, and the Netherlands – 20 with the start of deliveries in 2003, i.e., totally 423 helicopters.

The NH-90 helicopter was seen as a rotor-craft version of the infantry fighting vehicle (IFV) for use in the TTH tactical transport version (Tactical Transport Helicopter); in the embodiment of shipborne NFH (NATO Fregat Helicopter) the helicopter to be used

for anti-submarine warfare and search and rescue operations (Fig. 1.11).

Design feature is the extensive use of composite materials as well as the implementation of the angular fuselage side surfaces formed by flat faces, as on infantry fighting vehicle, which should help to reduce the susceptibility by the land fire and reduce radar visibility. Landing gear is made retractable. Main rotor has four blades with hingeless attachment, titanium hub and elastomeric flexible elements. The helicopter can carry a variety of weapons.

According to carrying capacity (2000 kg) and take-off weight (10 tons) helicopter NH-90 occupies an intermediate position between the light helicopter, Sikorsky UH-60 «Black Hawk» and AS.332 «Super Puma» of the third generation ($m_{t\text{-}off}$ = 9 tons) and new heavy helicopter EH-101 ($m_{t\text{-}off}$ = 14.3 tons), so the helicopter NH-90 appears on the long-established market, filled with well-known helicopters.

To the fourth generation light multipurpose helicopter VK.117 may also be added, it is the deep development of the helicopter Bo. 105 with advanced design and systems. Development of the helicopter by the «LEV» and «Kawasaki» companies began in 1977 on the basis of projects of the Bo. 107 German helicopter and KH-7 Japanese helicopter.

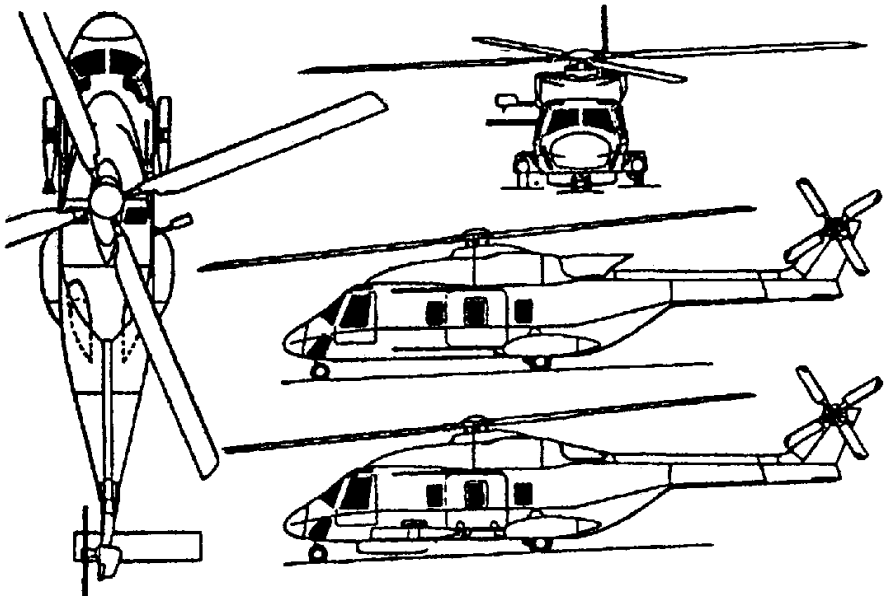


Fig. 1.11. NH-90 Helicopter for NATO Members

Initially the helicopters were developed as civilian to service rigs at sea, and then both as administrative and search-and-rescue experimental helicopters were built in 1979 in Germany and Japan, where their production was started since 1951, 255 helicopters were built in Germany and more than 100 were built in Japan. Having three-bladed rotor with rigid mounted blades similar to the Bo. 105 the BK.117 helicopter is equipped with two more powerful GTE and has a fuselage with the best aerodynamic shape, which cabin can accommodate 7-9 passengers.

ALH light multi-role helicopter is an evolution of the VK.117 helicopter– ALH light multipurpose helicopter, its development was conducted in India, with the participation of the «MVB» company since 1984, and the first flight took place in 1992. The helicopter ALH has the same carrying system as VK.117 helicopter with four-bladed rotor, but with hingeless mounted blades using elastomers made of CM, equipped with two more powerful turbine engines and has larger cabin for accommodation of 10-14 passengers. The helicopter is designed for use in India and adjacent countries; It plans to build more than 1,000 helicopters, including 200 civilians and 350 for the armed forces of India and 450 for export. The cost of developing the helicopter is estimated at 165 million dollars.

To substitute the Bo. 105 helicopter the «Eurocopter» European consortium in 1991 has started program to create light multi-purpose helicopter EC 135 of the new generation. The helicopter has cabin for 7-8 passengers, equipped with two more powerful turbine engines and improved four-bladed rotor with hingeless mounted blades of the CM. MR hub has a cowl, «fenestron» of the new generation with asymmetrically arranged blades to reduce noise is used instead of tail rotor. Compared with the Bo. 105 helicopter the fuselage parasitic drag was decreased by 30% due to good aerodynamic shapes. The first two test flights of helicopters were held in 1994, pre-production - in 1995. Serial production started in 1996, it is planned to produce 30 – 60 helicopters per year, and it is supposed to build 700-900 helicopters, helicopter price is of 4.4 million dollars.

The «Eurocopter» consortium together with the «Singapore Technology» company and the Harbin Aircraft Manufacturing Corporation has developed and in 1995 built new five-seat light multi-purpose helicopter of the EC 120 with one turbine engine, three-blade hinge-less rotor and «fenestron». 1600-2000 aircraft are expected to be produced for civilian and military applications. Configurations of VK.117, EC 135 and EC 120 helicopters are shown in Fig. 1.12.

Completing analysis of the new programs of the helicopter fourth and fifth generations it should be noted that their development is characterized by significantly increased duration and costs, this makes difficulties for financing and requires combined efforts of different companies and countries. Therefore development of new helicopters is carried out in a form of extensive use of international programs.

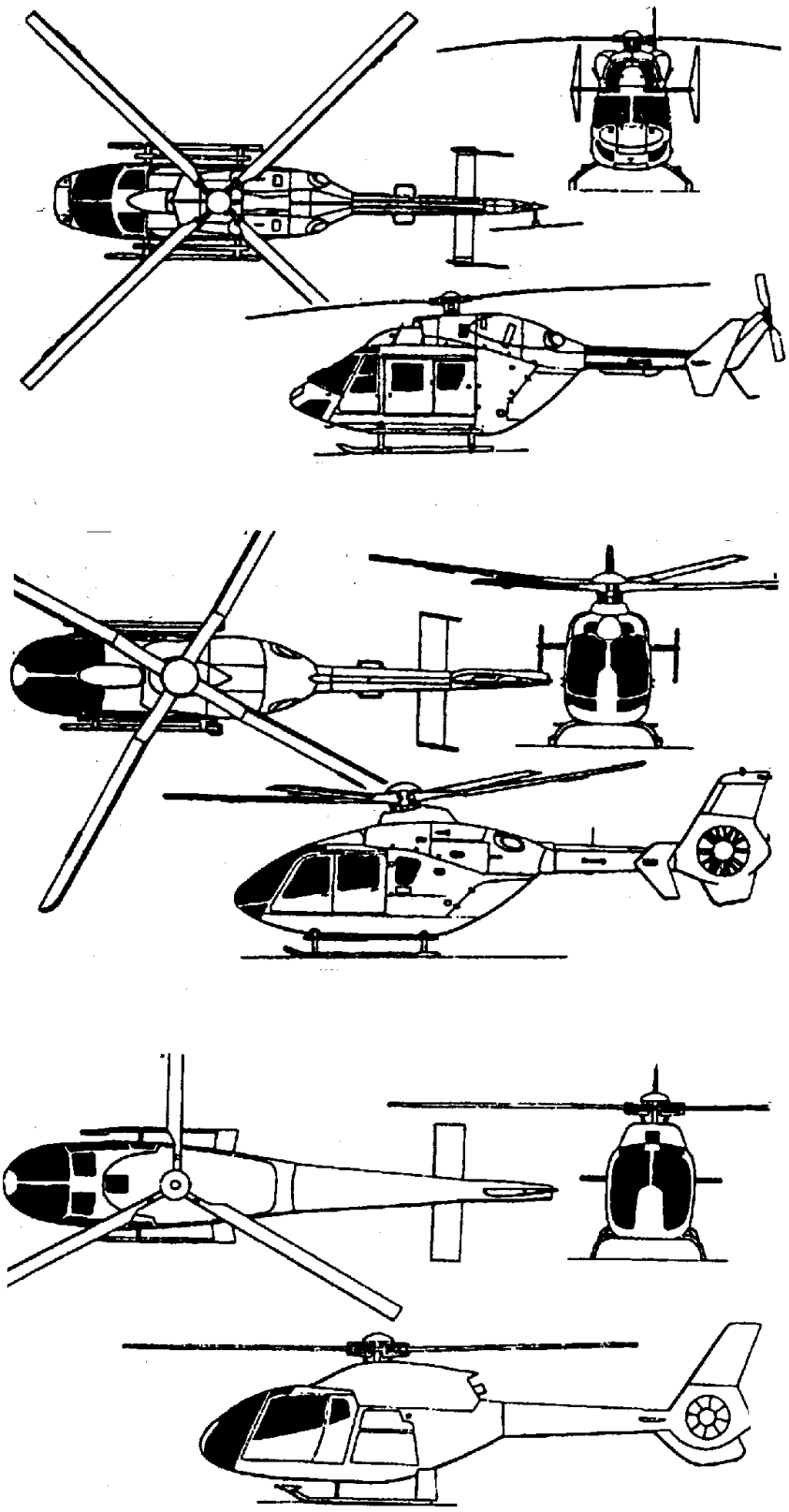


Fig. 1.12. BK 117, EC 135 and EC 120. Light Multipurpose Helicopters

1.3. COMBAT HELICOPTERS AND ROTORCRAFT

Successful application of light multi-purpose and transport helicopters in the hostilities in Korea has necessitated the creation of specialized military helicopters to escort them, as well as direct support to infantry units on the battlefield. Initially, for this purpose the attempts were made to adapt existing multi-purpose aircraft and helicopters by installation of weapons on them, but they could not provide the required combat effectiveness.

Therefore, first of all the United States since 1961 began intensive development of specialized military helicopters. The first among them was combat helicopter Bell AH-1 «Cobra Hugh», created in 1965 on the basis of multi-purpose helicopter UH-1 «Hugh» with the same carrying system and power plant, but with narrow streamlined fuselage having two-seat cabin for the pilot and gunner, and armed with a machine gun or gun on the ventral turrets, and also anti-tank guiding missiles & free-flight missiles on pylons under the wings. Helicopters AH-1 have passed accelerated test and since 1967 started to be widely used in military operations in Vietnam. For 10 years, it was delivered 1560 single-engine helicopters of various modifications: AH-1G, Q, S, E and F, used in the US Army, Japan and other countries. It was also delivered 350 twin-engine helicopters AH-1J «Sea Cobra» for the US Marine Corps and Air Force of Iran. The evolution of the AH-1J helicopter became enhanced helicopters AH-IT and AH-1W «Super Cobra». Four-bladed rotor modification of AH-1BW was developed.

The next successful combat helicopter has become the helicopter Hughes YAH-64 with upgraded main and tail rotors, called «Apache». Serial production of AH-64 helicopters was started in 1981 and completed in 1993. It was built 937 helicopters, which were the main US Army and other countries combat helicopters. In 1994, production under license of the helicopters «Apache» was began by «Westland» in UK.

Helicopters AH-64 have weapons consisting of caliber 30 mm cannon and 8-16 «Hellfire» anti-tank guided missiles with semi-active laser guidance system, equipped with a system of detection and identification purposes and fire control, including infrared night vision systems and laser designator. Helicopters were successfully used during the war in Iraq, and other military conflicts, mainly against armored aircraft, being considered the most effective means of its destruction. Total cost of development and production program of 811 helicopters AH-64 was 11.7 bn. dol., Including 1.23 billion dollars for R & D, and the average price of a helicopter - 14.3 million dollars.

Trying to keep the combat helicopters AH-64 «Apache» in service in the 2000s, the US Army started a large program of modernization by installation of the «Longbow» millimeter-wave radar in the cowling of the rotor hub. Upgraded helicopters AH-64D «Apache Longbow» (Fig. 1.13) can be effectively used without an escort helicopter OH-58D at any time of day or night in any weather. Development of the modernization

program of 218 helicopters began in 1993 and lasted until 2005, and its total cost will amount to 3.2 billion dollars at a price of one helicopter modernization of 14.1 million USD, which is comparable to the purchase price of the original AH-64A helicopters. The first modernized combat helicopter AH-64D made first flight on March 17, 1997.

Trying to combine the functions of reconnaissance and military helicopters as well as light multi-purpose helicopters, the US Army began in 1981 to study a new generation helicopter program LH (Light Helicopter), called «the most ambitious helicopter program ever developed in peace time» and corresponding to the doctrine of «Air Land Battle 2000» (joint air-land operations in 2000). It was originally supposed to order up to 4,500 LH-helicopters, to substitute multipurpose helicopters UH-1, reconnaissance OH-58 and OH-6 and combat AH-1 used in 10 different versions, and then the AH-64 with one type of helicopters that should not only simplify the operation of military helicopters, but also reduce its cost.

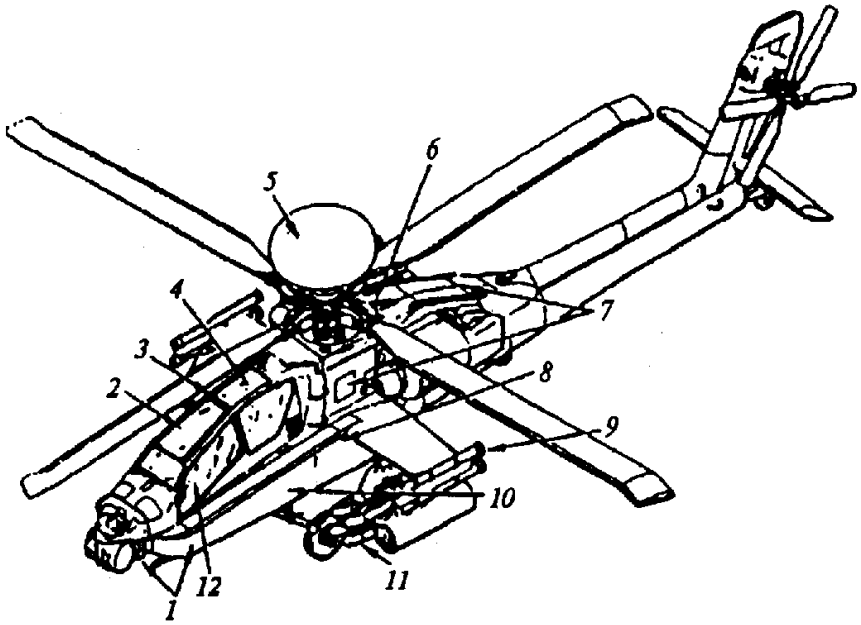


Fig. 1.13. AH-64D. Modernized Combat Helicopter

All combat helicopters, including new under development have a crew consisting of two members. Further development of the RAH helicopter was carried from the condition that its crew will include a pilot and the gun operator.

As the best the joint project of the «Boeing» and «Sikorsky» companies was recognized, which is based on the single-rotor configuration with «fantail» instead of the tail rotor, and got designation of RAH-66 «Comanche». RAH-66 helicopter is equipped with five-blade main rotor, hub with cowling and hingeless mounted blades with swept tips.

The design of the RAH-66 helicopter (Fig. 1.14) uses a number of original technical solutions, or «key technologies», such as the tilt of the «fantail» to create

thrust vertical component as well as the use of «stealth» technology with particular attention paid to the reduction of the visual, acoustic and infrared visibility, assisted by a special arrangement of the airframe with flat faces (Fig. 1.15), contributing to the dispersion of radar rays detecting a helicopter. The gas turbine engine stream flows through the narrow gaps mixing with the air flow pushed by the rotor; this not only reduces the temperature of the gases, but also improves the flow around the rear fuselage section. The helicopter is painted in standardized for the army green matt black polyurethane paint, contributing to reduce the visibility. With all these improvements the effective scattering surface of the RAH helicopter, which is used to estimate the radar visibility, is reported to be reduced by 1000 times compared with the AH-64 helicopter, also visual and acoustic visibility got significant decrease.

The RAH helicopter weapons feature with its conformal placement inside the compartments on the openable leaves, which also reduce the helicopter visibility. When the visibility requirements become non-obligatory then it is possible to install extra wing to house armament hardpoints. The helicopter will be equipped with 30 mm cannon on the turret under the fuselage in addition to the anti-tank guided missile, unguided air-to-surface missile and guided missile placed in the weapons bays.

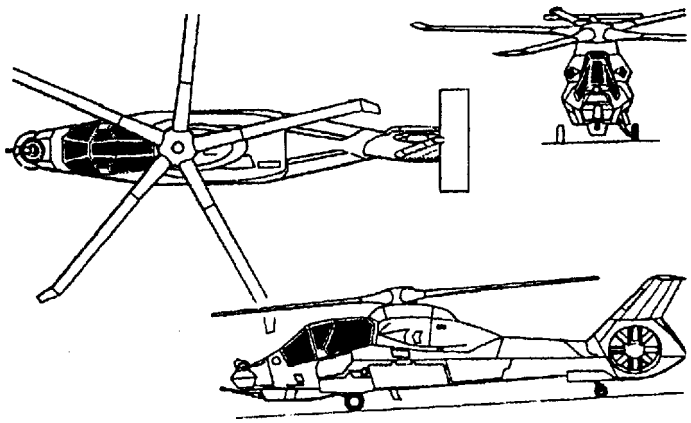


Fig. 1.14. RAH-66 «Comanche» Reconnaissance Combat Helicopter

The most important advantage of the RAH helicopter should be a set of advanced equipment to perform a variety of combat missions, having common elements with the equipment being developed for advanced F-22 fighter for the most expensive ATF program, including a system of «night piloting» with IR-view system and improved target designation system (third helicopter was supposed to be equipped with the «Longbow» targeting system with millimeter-wave radar).

The RAH-66 helicopter program remains of the top priority among the United States helicopter program, although its development is complicated by a number of problems. The first flight has revealed the transmission system resonance, which

required additional ground tests and detained the second flight for 7 months and the third flight for 7 months due to the need to change the LG design. Mass production will begin in 2005 and equipping the US Army with RAH-66 helicopter is scheduled for 2007, that is why the main hopes are laid on the modernization of the AH-64D and OH-58D army helicopters, and even to keep in operation modernized first-generation AH-1F and AH-1W combat helicopters.

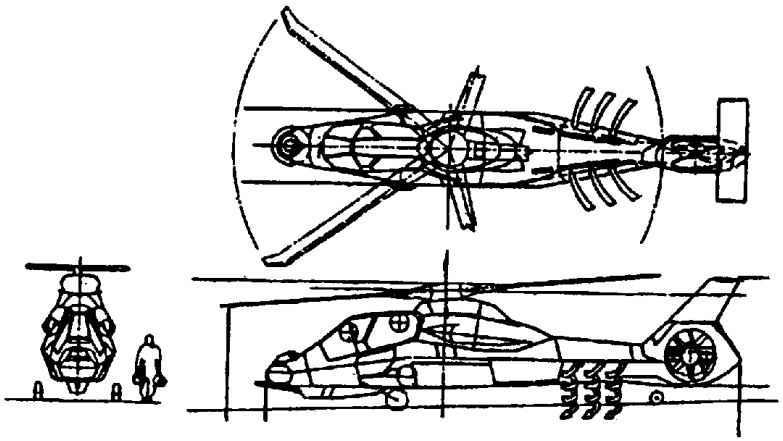


Fig. 1.15. System Contributing to Reduce IR Visibility of RAH-66 Helicopter

It is assumed that the RAH-66 «Comanche» helicopter appearing in operation only in 2007 for many years will be able to meet the needs of the armed forces, so in the near future it is not planned to develop a new program of the next generation attack helicopter following the «Comanche». The possibility of expanding the functions of the «Comanche» helicopter is considered such as aerial fighting and creation of new modifications in the future.

In Russia, development of military helicopters began with the installation of weapons to helicopters first appeared in use. For example the Mi-1 light multipurpose helicopter was used to start ATGM «Baby», managed by wire. Mi-8T helicopters were equipped with powerful weapons: on the truss pylons placed on each side of the fuselage, integrated holders were that accommodated 4 ATGM, up to 6 blocks UB-32 with 32 UGM C5, bombs caliber 100, 250 and 500 kg, or 4 gun containers with 23 mm guns, the nose part accommodated machine-gun installation НУВ-1 with 12.7 mm machine gun, and in the openings of sliding doors and windows 8 pivot units with 7.62 mm machine guns, making the Mi-8TV, according to the estimations given in foreign press «the most heavily armed helicopter in the world».

In contrast to the AH-1 light combat helicopters in the USSR it was started to develop transport-combat helicopter with weapons that could be used not only for direct support on the battlefield and fight with tanks, but also for tactical landings in the area of breakthrough, and to evacuate the wounded. It is having the cockpit, which could accommodate 8 paratroopers or 4 wounded on stretchers.

The V-24 first experimental helicopter made first flight on September 15, 1969, serial production of the Mi-24A was began in 1970, and since 1971 it have been started to put into operation. Since 1973 improved Mi-24A become available for countries of the Warsaw Pact. It was built of more than 3,300 helicopters for the countries of the Warsaw Pact and for export to 15 countries.

Similar to US helicopter AH-1 «Hugh Cobra» was developed on the basis of the UH-1 helicopter, the Mi-24 helicopter was created with the carrying system, transmission and engine of the Mi-8 helicopter, which not only speeds up the development, but also provides unification of units and systems. The helicopter had a streamlined fuselage with double armored cockpit for pilot and shooter, wing to reduce load applied to the main rotor and to suspend weapons and retractable landing gear (Fig. 1.16).

According to its armament and combat effectiveness the Mi-24 was superior to all foreign helicopters and got called «flying tank». In various embodiments, the helicopter was equipped with movable machine gun or conventional gun or stationary gun with 30 mm double-barreled gun. ATGM first with manual, and then with semi-automatic guidance system were hung on the launching blocks on the wing. Later they began to install automatic supersonic anti-tank gun, and even unguided “air-to-surface” missiles and guided missiles of «air-to-air» class.

Mi-24 helicopters were successfully used in military operations in Afghanistan and the Iran-Iraq war, and totally in 30 different military conflicts. During the Iran-Iraq war, Iraqi helicopter gunships Mi-24 successfully conducted aerial combat with Iranian combat helicopters AH-1 «Sea Cobra» and often shoot them down. During the hostilities mainly Mi-24, Mi-8 and «Gazelle» helicopters have shot down 53 Iranian helicopters AH-1 «Sea Cobra» and others. Encountering Iraqi Mi-24 superior in armament, an Iranian helicopter «Sea Cobra» summoned to help the fighters. In one dogfighting in 1982 involved the Mi-24 helicopter the Iranian fighter McDonnell-Douglas «Phantom», which is not only more powerful in weapon, but also in take-off weight is twice the Mi-24, was shot down by unguided missile on a collision course. This is the first and only case in the world, when combat helicopter has shot down the combat aircraft in air fight.

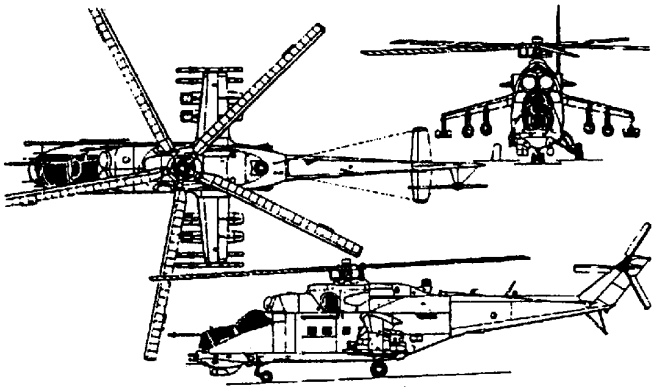


Fig. 1.16. Mi-24 Combat Helicopter

Accounting the fact that the Mi-24 helicopters have proven themselves in military operations and many of them are still in service in CIS and other countries, a deep modernization is offered for the existing fleet of Mi-24 helicopters. A modified export Mi-35M helicopter with advanced equipment and armament of the Mi-28 has been created.

New combat Mi-28 and Ka-50 helicopters represent a new generation of combat helicopters and possess greater combat effectiveness than the Mi-24 and the best performance characteristics, particularly a high maneuverability, allowing to perform complex aerobatic maneuvers.

Being different in design and executed in single-rotor configuration traditional for Mil design bureau (Fig. 1.17) and coaxial double-rotor configuration traditional for N.I. Kamov design bureau (Fig. 1.18), the helicopters are designed to solve similar problems, among which the most important are to search and to destroy armored vehicles and troops on the battlefield, as well as aerial targets in adverse weather and environmental conditions. Both helicopters are equipped with unique navigation and sighting system in conjunction with computer, but the Mi-28 crew consists of two members - a pilot and an operator of weapon systems, while the Ka-50 crew consists of only one person performing the functions both of the pilot and operator that requires a high level of automation for the first time in the world performed for combat helicopters. It should be noted that many of the problems for the Mi-28 and, especially, for the Ka-50 are solved for the first time in the world.

Both the Mi-28 and Ka-50 aircraft have strong armored protection for crew members and main systems and safely damaged design of major components that allows withstanding damage from the bullets and shells of 200 mm caliber. Landing gear design and crew seat securing members provide safe emergency landing, greatly reduce the impact of overloads. The Ka-50 helicopter was for the first time in practice equipped with ejection seat for the pilot bailout and preliminary abrupt removing of rotor blades.

Helicopters armament consists of 30 mm caliber automatic cannon, in addition, for fighting with armored vehicles helicopters can carry up to 16 supersonic anti-tank laser-guided missiles with a range of 8 km or 4 blocks of 20 not-guided missiles. Helicopters may also be armed with «air-to-air» missiles to fight with aircraft and helicopters, making them a versatile weapon system, which is not only competitive, but also superior to the best attack helicopters.

Mi-28 and Ka-50 were used as a basis for new enhanced modifications fitted with integrated avionics complex of the fifth generation for day and night use and all-weather flights with terrain following at a height of 5 ... 15 m in the automatic mode. The helicopter is equipped with over-hub millimeter-wave radar and thermal imaging night vision in the bow.

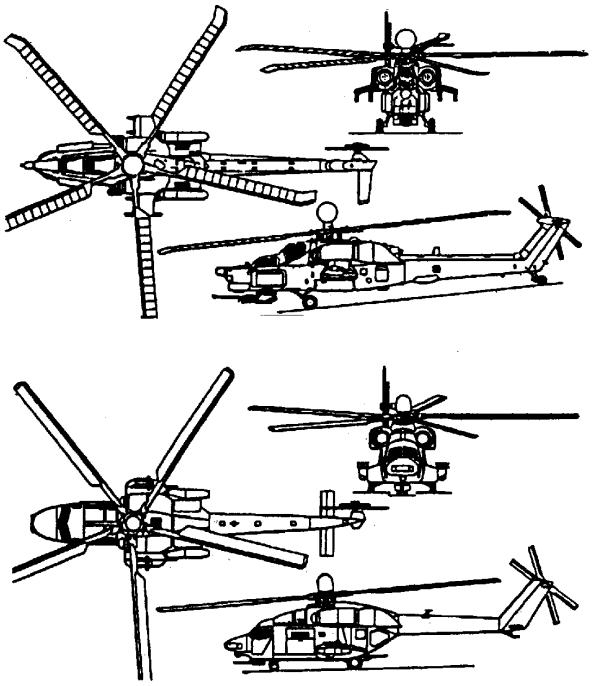


Fig. 1.17. Mi-28H and Mi-40
Combat Helicopters

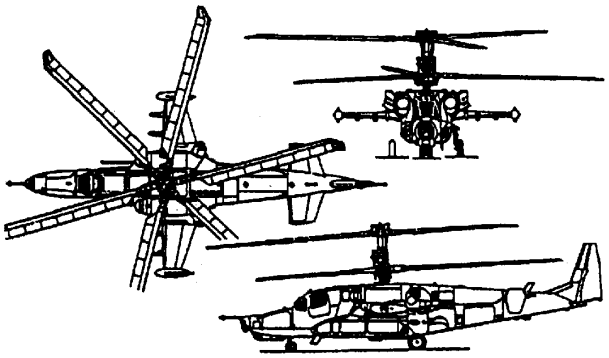


Fig. 1.18. Ka-50
Combat Helicopter

The Ka-52, as well as Mi-28N, are designed for all-weather day and night missions and able to solve wider range of tasks than the Ka-50 having 85% of common overall design with it.

An advanced Mi-40 was designed on the Mi-28 basis acting as infantry fighting vehicle and intended for airborne divisions of rapid deployment. Having carrying system, transmission and power plant of the Mi-28, the Mi-40 helicopter has larger fuselage, armored cockpits in addition to crew of two members, can accommodate up to 8 paratroopers, who can fire from the regular weapons through the hatches. Helicopter armament consists of 23 mm caliber gun on turret, as well as anti-tank guided missiles or unguided missiles blocks.

In European countries development of combat helicopters followed the same way as in the United States and Russia, but with some delay and smaller span. Originally «Scout» and «Links» European light multipurpose helicopters were used as combat aircraft in the UK, «Alouett» and «Gazelle» in France and Bo. 105 in Germany.

Development of new combat helicopter was started in Italy in 1982, on the basis of A. 109 multi-purpose helicopter. The «Mongoose» new combat helicopter A. 129 made its first flight in 1973. It had narrow fuselage, double cabin for the pilot and shooter. 4 containers for 52 unguided air-to-surface missiles or heliborne laser fire and forget missile «Hellfire» were suspended in containers under the wing. Also Installation of machine guns or guns is possible (Fig. 1.19).

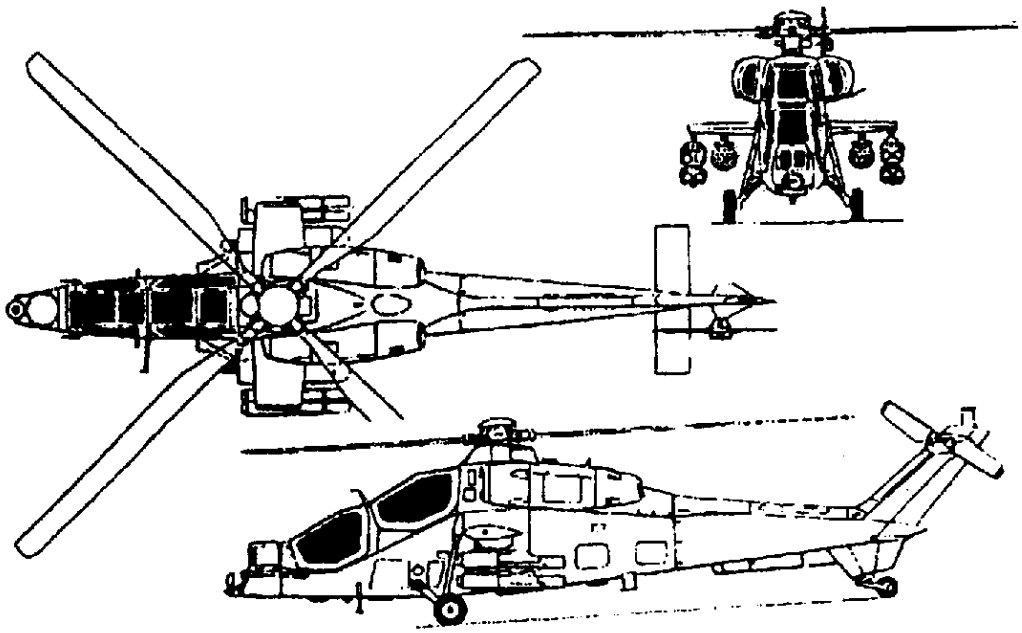


Fig. 1.19. A.129 «Mangoose»

Trying to create own European combat helicopter, superior to «Hugh Cobra» and not giving place to «Apache», Germany and France began in 1984 joint development of new generation military rotorcraft of «Tiger» antitank modification for Germany Army and of «Gerfaut» close-in ground support for French Army. The program was revised several times due to different requirements and high costs. 5 experimental helicopters were built. They made their first flights in 1991–1994. Initially it was supposed that Germany needs 212 antitank and 75 close-in ground support aircraft, and France needs 140 close-in ground support aircraft, which will be delivered up to 2010. At the moment this terms are shifted to 2020. Due to limitations of funding there provided first delivering of 80 helicopters to Germany and France with possible continuing procurement. Turkey has revealed an interest to buy «Tiger» helicopters.

The «Tiger» helicopter (Fig. 1.20) configuration is similar to «Apache». But it is smaller. Its engine has lower power and can carry lighter combat load. It is an ambitious project to essential degree putting an objective to create an own helicopter for European countries fitted with European equipment and armament with the aim to be independent from USA. Helicopter accounting price should be of 10…12 mln dol, that is close to averaged price of the AH-64A «Apache» vehicle. With that «Tiger» will be fitted with radar of millimeter wavelength range placed over the MR hub and «Flir» system placed in the nose, which are installed in the modified AH-64D «Apache Longbow» helicopter.

On opinion of France and Germany aerospace industry management the «Tiger» program should have the highest priority among European programs.

Great Britain has turned aside from the participation in the «Tiger» program preferred to produce an American combat helicopter «Apache» under license, and also

had bought the American military transport helicopter «Lockheed» C-130 having minimized own participation in the future European transport airplane FLA.

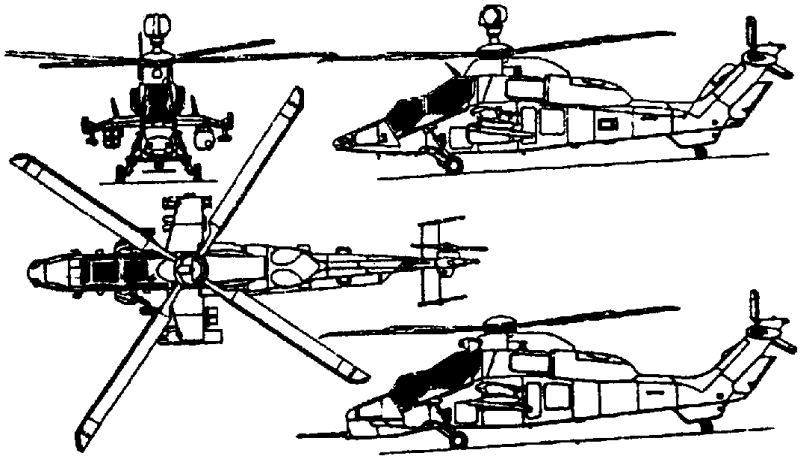


Fig. 1.20. «Tiger» Eurocopter European Combat Helicopter

The unexpected was new combat helicopter CSH-2 «Rooivalk» appeared on the market in 1990 developed in South Africa on the basis of carrier systems and units of the «Super Puma» French helicopter. By its layout and flight characteristics it is close to the «Apache» helicopter, with even more powerful GTE and some higher take-off weight and the same armament (Fig. 1.21). Consortium «Eurocopter» examines the helicopter «Roivolk» as a complement to the «Tiger» light helicopter, they are to be shared: the helicopter «Roivolk» as a combat helicopter and the «Tiger» for support. South African Air Force ordered 16 helicopters «Roivolk», which were successfully demonstrated at a number of aerospace exhibitions, competing with US and Russia combat helicopters.

Japan has shown the tendency to self-sufficiency, previously used in its Self-Defense Forces US helicopter gunships «Hugh Cobra» and «Super Cobra» and light reconnaissance and attack helicopters OH-6D. To replace these helicopters in 1992 the development program was initiated by the «Kawasaki» company of light reconnaissance and attack helicopter OH-X, which is the first helicopter fully developed by Japanese specialists (earlier Japan made helicopters under USA license only). Helicopter OH-X made its first flight in August 1996. It is designed according to single-rotor configuration with «fenestron» and has two Japanese GTE and Japanese equipment and weapons, characterized by the use of the latest technical solutions and the widespread use of the composite materials (Fig. 1.22). The helicopter is armed with close combat guided missiles «Air-Air» and is equipped with an optical system for surveillance, reconnaissance and targeting system and data far radar detection. Production is expected of 150-200 helicopters for Self-Defense Forces of Japan. The combat helicopter similar in layout and characteristics is developed in South Korea, where it will also be designed only for own armed forces.

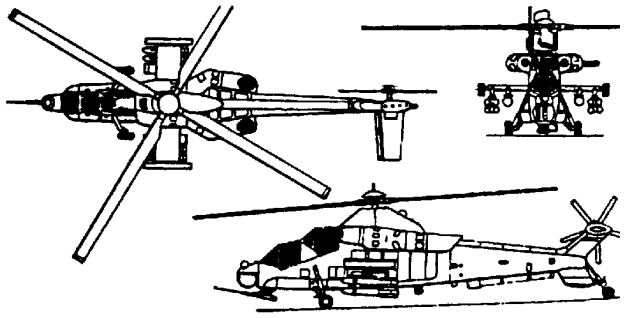


Fig. 1.21. CSH-2 «Rooivalk»
South Africa Helicopter

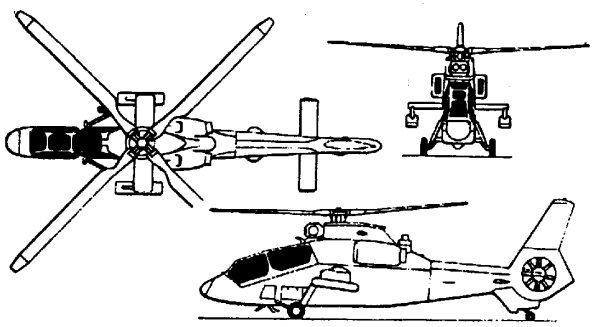


Fig. 1.22. Kawasaki OH-X
Japanese Combat helicopter

1.4. STUDYING HIGH-SPEED ROTORCRAFT

When creating the first experimental helicopters, the main purpose is to provide vertical takeoff and landing, and does not attempt to achieve high speeds. Therefore, these helicopters are distinguished by simplified structure, often truss-type without cladding, imperfect aerodynamic shape and high parasitic drag, usually estimated by area of equivalent harmful plate $\sum C_X S$. With the improvement of helicopters and creation of new generations it began to pay more attention to improving the aerodynamics and increase in cruise speed.

Fig. 1.23 shows graph of $\sum C_X S$ change, depending on the take-off weight for the helicopter types and for comparison also shows the values $\sum C_X S$ for aircraft with propellers. Assuming that presence of such helicopter construction elements as main and tail rotors, and taking into account the specific characteristics of their layout, assume that the value of parasitic drag $\sum C_X S$ should be not more than by 50 % exceeds the aircraft propellers $\sum C_X S$ (this relationship is shown in phantom). To better assess the perfection of external aerodynamics of helicopters it was suggested to divide value $\sum C_X S$ by the take-off weight $m_{t\text{-}off}$.

Fig. 1.23 represents also graphs of changing the value $\overline{\sum C_x S} = \frac{\sum C_x S}{m_{t-off}} \cdot 10^{-3}$ for different type helicopters depending on take-off weight.

In graphs it can be clearly seen that the first helicopters such as the Sikorsky R-4, or Bell 47 or Hiller OH-23, feature by great parasitic drag 3-4 times greater than drag of helicopters of this class with improved aerodynamics such as OH-6A or SA.342. Assumption made about limiting value of $\sum C_X S$ for helicopters, by 50 % higher than for propeller-driven airplanes, is well supported: the values of $\sum C_X S$ of the helicopters with the most advanced aerodynamics such as OH-6A, S-76, S-67, Mi-24 and Mi-26,

are close to the limit, shown by the dashed curve. The graphs also show that the single-engine helicopters feature less parasitic drag than those of the double-rotor tandem and coaxial configurations. We also see that all the helicopters, which set world records in speed (Mi-1, Mi-2, Mi-6 and Mi-24 and S-76, S-67, SA.342 etc.) feature good aerodynamic contours, and had low parasitic drag.

Improving the aerodynamics is exemplified by the development of single-rotor helicopter of the «Sikorsky» company. Fig. 1.24 shows at the same scale the side views and cross-sections of the family helicopters starting from R-4 and ending with S-76, and provides values of $\sum C_X S$ and $\overline{\sum C_x S}$ for them.

From the first helicopter R-4 with angular forms it was gradually transferred to the streamlined fuselage of airplane type with retractable landing gear and vertical tail surfaces, which enabled to reduce $\overline{\sum C_X S}$ for more than eight times and increase, respectively, equivalent lift-to-drag ratio $K_{eq} = \frac{Y}{X_{eq}} = \frac{m_{t-off} V}{75 N_{req}}$ from three of R-4 up to five of S-76, with that cruising speed corresponding to the maximum K_{eq}, increases for more than two times, from 120 km/h of the R-4 to 260 km/h of S-76.

Achieving of high speeds is possible, but it is associated with a reduction in lift-to-drag ratio and an increase in fuel consumption, so the maximum speed achieved in setting world records on the basis of 15 ... 25 km, is not indicative as, for example, World Record 408.75 km/h done by the «Super Lynx» helicopter with the engines, which created significant propulsive thrust. To assess the capabilities of helicopters more indicative rates are achieved in the absolute world records on the basis of 100 and 500 km, set by the helicopter Mi-6 with a wing (340.15 km/h in 1964) or modernized helicopter S-76 (345.74 km/h in 1982) and remain unsurpassed.

Increasing speed of helicopter was tried to achieve through the use of advanced supporting systems, such as the Dershmidta rotor with controlled movement of the blades in the plane of rotation or coaxial rotors using advanced blade concept ABC. However, these systems proved to be structurally very complex, and their operation, as shown by the test pilot of the helicopter Sikorsky S-69, has been associated with many problems, so their development has been stopped.

Trying to increase the speed of rotary-wing aircraft the intensive studies of high-speed rotary-wing aircraft of various types were conducted as shown in Fig. 1.25, which also shows variation of equivalent lift-to-drag ratio K_{eq} depending on flight speed. The simplest solution was to use a wing for reducing forces acting upon the rotor in high-speed flight. The wing is a necessary element for the helicopters of side-by-side configurations, such as B-11 and B-12, or is additionally installed on single-engine helicopters such as the Mi-6, Mi-24, Bell «Hugh Cobra» and others.

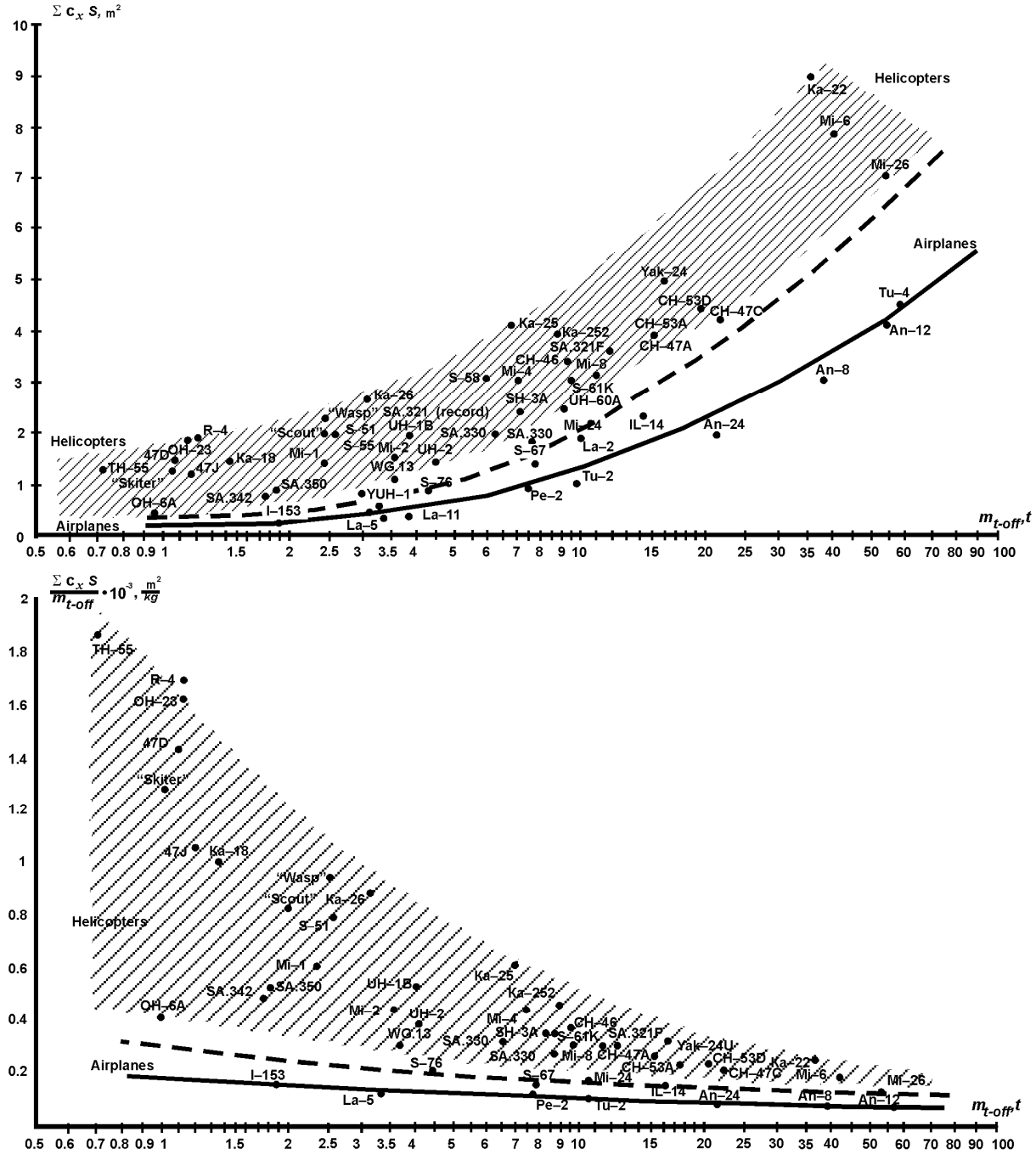


Fig. 1.23. Change of Equivalent Parasitic Drag Area $\sum C_x S$ and Ratio $\frac{\sum C_x S}{m_{t-off}} \cdot 10^{-3}$ for Helicopters of Different Types Depending on Take-off Weight

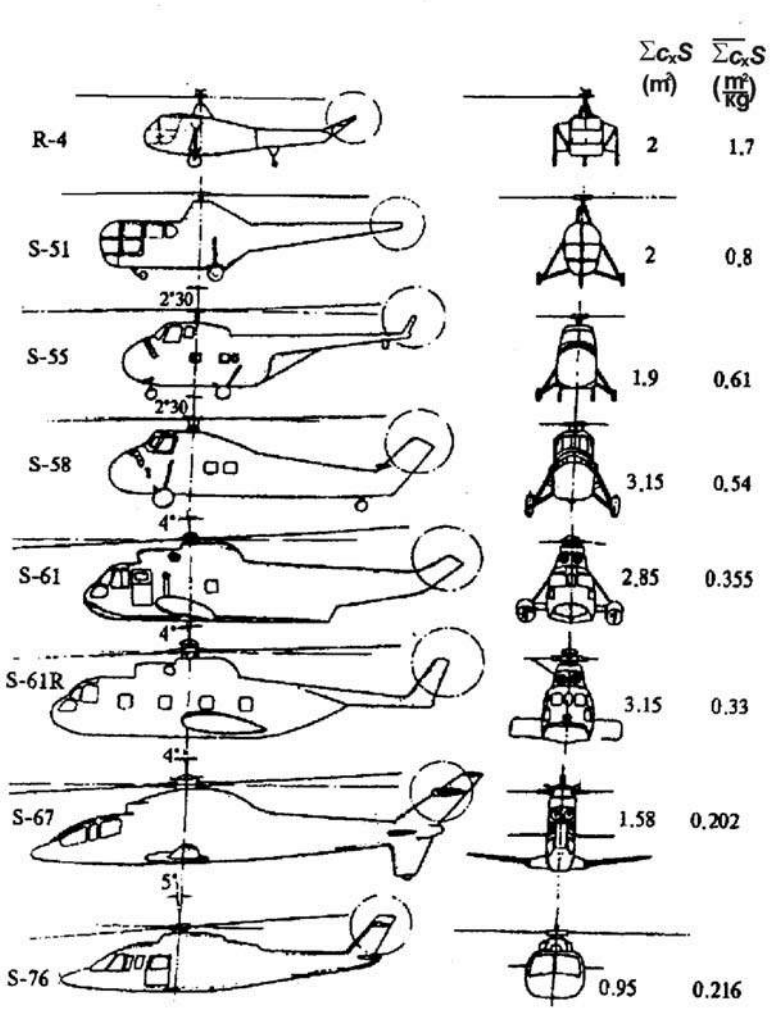


Fig. 1.24. Enhancing Aerodynamics of «Sikorsky» Helicopters

However, the wing, creating a lift to reduce load on the main rotor, also creates additional drag. So the next step was the use of the rotorcraft configuration, where in addition to the wing the additional thrusters were installed: pulling propellers, as on experimental rotorcraft TsAGI-11EA and Ka-22, or pusher propeller on rotary-wing Lockheed AH-56, or additional jet engines on experimental rotorcraft made by companies «Bell» and «Lockheed». At one of these rotorcraft Bell YUH-1 with a wing and two turbojet engines the maximum speed of 508.5 km/h was achieved in 1969, which was not registered as a world record, but it shown the possibility of stable operation of the rotor in such high flight speeds: tips of advancing blades were streamlined with supersonic speed (M = 1.01), and an equivalent lift-to-drag ratio was at most 1.5 due to increased drag of main rotor that means it is no sense to use main rotor at such speeds.

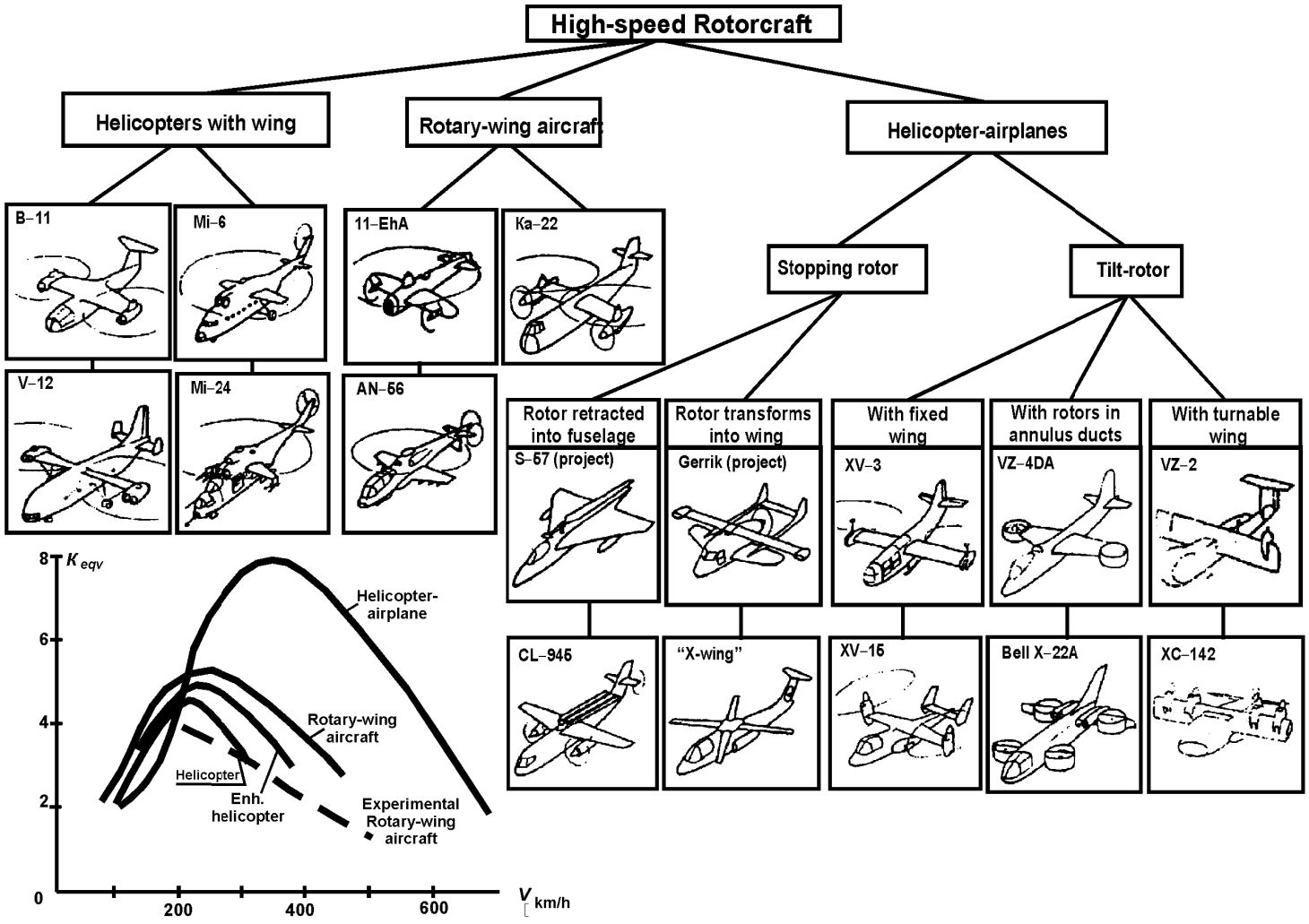


Fig. 1.25. Classification of High-Speed Rotorcraft and Variation of Their Equivalent Lift-to-Drag Ratio Depending on Flight Speed

That is why essential attention was paid to researching rotorcrafts or helicopter-airplanes, which can transform being airborne into airplane flying with high lift-to-drag ratio while taking off and landing as helicopters.

After successful demonstration of one XV-15 helicopter-airplane at aerospace exhibition in Paris in 1981 US Army decided to start JVX full-scale program (Joint Services Advanced Vertical Lift Aircraft) intended to develop an experimental multi-role airplane of vertical takeoff and landing (VTOL) for all arms, which will be used to carry paratroopers, for search and rescue, to be used as combat, reconnaissance and even antisubmarine with possible substitution of corresponding types of airplanes and helicopters. According to research results it was selected VTOL with fixed wing and tilted rotors of intermediate type between heavy loaded airplane propellers and light loaded hinged helicopter rotors that provides a compromise for hovering when helicopter rotor is optimal and for flight with high speed when propeller having blades with great twist is optimal.

US Ministry of Defense decided to entrust development of the JVX program accounting its complexity to the «Bell» and «Boeing-Vertol» companies in 1982 as these companies have maximum experience in developing rotorcraft with tilt rotors.

It was decided to develop VTOL common for all armies. It got name V-22 «Osprey» (Fig. 1.26). VTOL with tilted rotors having diameter of 11.6 m and two turboprop engines having power of 4585 hp; maximum take-off weight during vertical takeoff was set 21545 kg, for takeoff with short run – 24950 kg (with rotors tilted to 45°), flight range with vertical takeoff – 2225 km, for takeoff with short run – 3340 km, maximum payload transported in cabin – 9070 kg, on external sling – 6805 kg, VTOL MV-22 carries 24 paratroopers with speed of 460 km/h and range of 370 km. It can hover while landing of troops at altitude of 915 m and 33°C. Substitution of the CH-46E obsolete paratrooper carrying helicopters with 425 CV-22 VTOL helicopters will last till 2022. Total cost of the development and production program of 523 V-22 VTOL is estimated to be of 45 billion dollars, including R&D costs of 6,6 billion dollars, and average price of one VTOL – 86 billion dollars that makes the V-22 program the most expensive among rotorcraft programs. Nevertheless the V-22 program is one of the highest priority programs for the USA army and its funding will continue.

VTOL with tilted rotors many times were proposed for passenger transportation between cities using existing helipads in the city centers, and also to deliver passengers to the airdromes located out of the city.

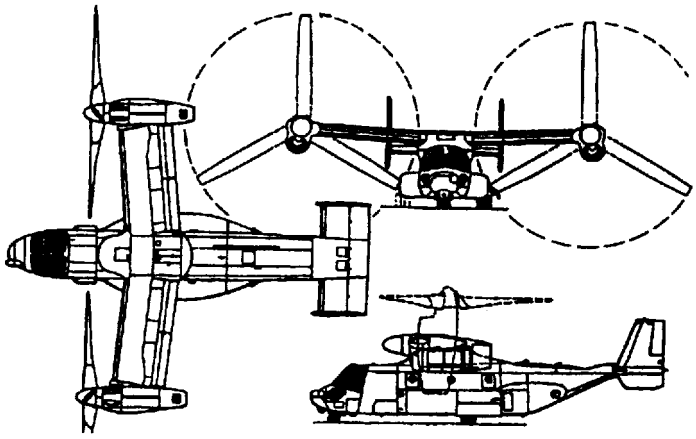


Fig. 1.26. «Osprey» Bell-Boeing VTOL V-22

And only now, after having started a series production of military V-22 VTOL it became possible to create a civilian tilt-rotor VTOL, which is not so large as V-22, for fear that it would not find a use, and lighter, like XV-15 for which it is possible to use existing helipads in the city centers and even roofs of buildings, adapted for operating small helicopters.

That is why the «Bell» company after careful consideration of possible VTOL applications has offered the project VTOL Bell-Boeing VV.609 developed jointly with the «Boeing» company for use as administrative, medical, search and rescue, maritime intelligence, police and training.

1.5. RESEARCH OF JET-DRIVEN ROTORCRAFT

Difficulties in creation of mechanical transmission for the first helicopters, especially in transmitting high power to main rotor for heavy helicopters forced designers to turn to a jet-driven rotor that does not require transmission and is not limited by the transmitted power. In Russia, USA, Britain, France and Germany, a number of experimental jet-powered helicopters with jet-driven rotor were built and original projects had been developed, which cannot be ignored when analyzing the development of the world helicopter.

Most of helicopters with jet-driven main rotor were light single-seat and two-seat and fitted with jet engines at the blade tips: YH-32 Hiller and Bratukhin's helicopters with ram engines, American HA-5 and XH-26 and Zherebtsov's helicopters with pulsating engines, liquid-reactive working on hydrogen peroxide Kellet KN-15 helicopters and rotor-craft RH-1 and turbojet on B-7 Mil helicopter.

Despite the simplicity of this drive type and a considerable simplification of the helicopter structure in a whole creation of helicopters with engines at the blade tips working in the field of high centrifugal forces and high noise level, having high fuel consumption, was revealed as not being a simple task, and they have not got evolution.

In later studies of promising power plants it was shown that for heavy helicopters

a mechanical drive rotor is more appropriate – a conclusion that defines the main directions of development of heavy helicopters.

1.6. FORECAST OF EVOLUTION OF HELICOPTER MARKET

An important part of researching helicopter evolution is creating forecasts of the global helicopter market, performed annually by special look-ahead organizations like «Forecast International» under the defense market analysis service (DMS (Defence Marketing Services)) in the USA for all aircraft types usually for 10 years, and leading aviation production companies they are interested in aircraft, for example, annually forecasts compiled by the «Allison» company of the light helicopters with GTE market for 10 years.

The most comprehensive DMS annual forecasts generalize statistical data on the production of civil and military helicopters, state of the development programs, major modifications, as well as characteristics. These statistics data, which are sufficiently accurate and agree well with the annual reports of helicopter manufacturing companies on production of their helicopters, supplemented with tables containing data of predicted production for 10 years, which correspond to the plans of production submitted by companies, usually overrated and, as a rule, not fully confirmed in the future. It is known that most of the programs vary considerably in the course of their development, changing volumes and the postponement of series production, which is reflected in the next forecast. Therefore these forecasts are advisory in nature and does not guarantee their performance.

According to the latest DMS forecast for development of the world helicopter market in 1997-2006 it is expected mass production of more than 4,600 military helicopters of total cost nearly 50 billion dollars and more than 8,000 civil helicopters of total cost about 20 billion dollars, that is totally for 10 years it should be produced 12,600 military and civilian helicopters, having total cost about 70 billion USD or an average of 1 260 helicopters per year, with their total cost of ~ 7 billion dollars.

Fig. 1.27 shows the DMS presented graphs of sales of military helicopters in 1987-1996 and the forecast of deliveries in 1997-2006 and their corresponding total cost in 1997-2006. The graphs show that delivery of military helicopters in 1987-1996 decreased by 2.5 times (Fig. 1.27, a) from 1254 in 1987 to 505 in 1996, however because of the considerable increase in price of new military helicopters their total cost decreased only by 25%, from 5.4 billion dollars in 1992 to 4.3 billion dollars in 1996 (Fig. 1.27, b).

For considered decade from 1997 to 2006 sales of military helicopters should stabilize at a level of 450 helicopters per year, with some increases to 490 helicopters in 2006, but total value of sold military helicopters to increase up to 6.3 billion dollars in 2006, that is by 60% compared to 1997

For civilian helicopters also observed a significant reduction in maximum sales –

1251 helicopters in 1990 to a minimum – 576 helicopters in 1993 (Fig. 1.28, a) for some increase in the total cost of 0.92 billion dollars in 1992 to 1.15 billion dollars in 1996 (Fig. 1.28, b). In the considered decade civil helicopter sales will increase slightly – from 714 helicopters in 1997 to 977 in 2005, with significant increase in their total cost of 1.3 billion dollars in 1997 to 2.9 billion dollars in 2005 that is 2.2 times.

Forecast of the company «Allison» for helicopters with GTE produced by this company is the closest to DMS forecast for 1997-2006. According to the forecast of the company «Allison» production of light military helicopters with GTE will increase from 270 in 1997 to 370 in 2002 and then will decrease again to 270 in 2006. Similarly, the production of light civil helicopters with GTE will increase from 500 in 1997 to 650 in 2001 and then once again will reduce to 600 in 2006. It should be noted that the forecast of the company «Allison» does not take into account production of medium-sized military and civilian helicopters, equipped with GTE of other firms, as well as the production of light helicopters with piston engines.

The DMS forecast graphs (Figs 1.27 and 1.28) also show distribution of produced and expected to be produced and sold civil and military helicopters with regard to type of propulsion system: with one piston engine, with one GTE and two or more turbine engines. Specifically, that military helicopters with two or more turbine engines dominated and will dominate in sales, and sale of helicopters with one GTE will be shortened essentially (ten times) and almost will be no sales of helicopters with piston engine (Fig. 1.27, a, b). For civilian helicopter production and sale of helicopters with piston engine will be the same as for helicopters with one turbine engine; for civil helicopters with two or more GTE production and projected sales will increase, and they are crucial for the total value (Fig. 1.28, a, b).

The analysis of sales by companies in 1992 - 1996 years is also of interest and sales forecast for the period of 1997-2006 with the evaluation of prices is shown in Fig. 1.29 a, b. During the period of 1992-1996 with regard to number of helicopters sold the first place is occupied by the «Bell» company producing light helicopters with its Canadian branch, which accounted for 22.3% of all sold helicopters, surprisingly followed on second place with the «Robinson» company, which produces light helicopters with piston engines and accounts for 15.1% of all sold helicopters, and only the third place is occupied by the «Sikorsky» company.

When adding to it the «McDonnell Douglas» company sold 8.5% of the helicopters, then the share of USA companies accounting for more than a half of the helicopters sold is of 58.1%, and for the rest of the countries – 41.9%, with the largest share of 11.5% introduced by the French branch of the «Eurocopter» consortium, 5.5% – for Mil helicopters and 5.1% - the «Augusta» company, and the remaining 19.8% – for the rest of the companies of other countries that produce own helicopters or helicopters under license of other countries (Fig. 1.29, a).

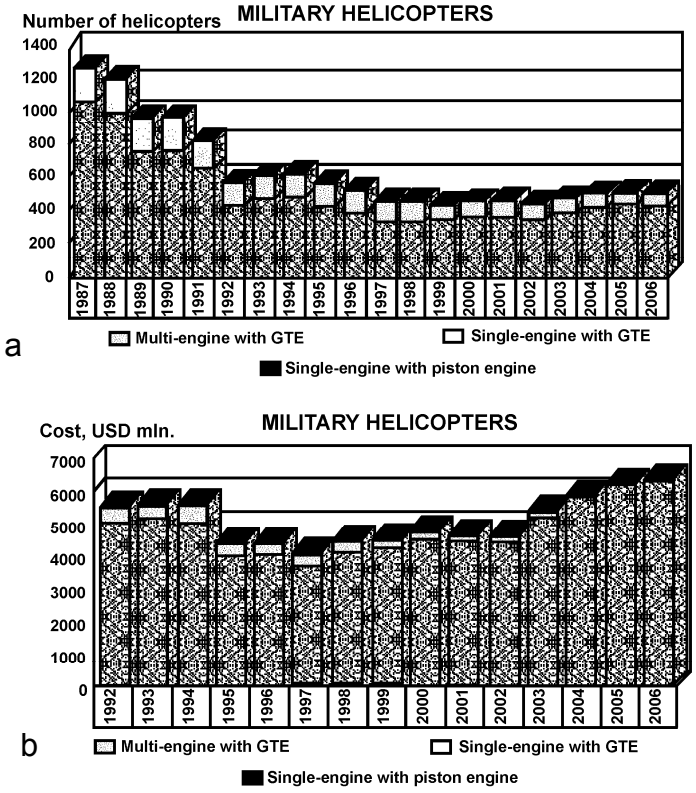


Fig. 1.27. Graphs: (a) Changes in Sales and Predicted Deliveries of Combat Helicopters in 1987 – 2006; (b) Their Corresponding Price in 1992 – 2006

During the period of 1997-200 the share of US companies in the total helicopter production will decrease to 42.7%, and then during 2002-2006 will increases to 50.7% due to wider participation of «Enstrom» and «Schweitzer» companies producing light helicopters. Consortium «Eurocopter» will keep production at the same level, and the Russian helicopters in the forecast for 2002-2006 are not stood out and included in all the others.

Such ratio will be kept in distribution of production and expected total sales (Fig. 1.29, b), where the estimated production and expected sale of Mil helicopters, which may increase by 6.3% in 1992-1996 to 8.9% in 1997-2001 and to 11.4% in 2002-2006. The share of the «Eurocopter» consortium in the total cost will remain stable at 7.5% and will be supplemented by sales of the «Europe helicopter Industry» and «NATO helicopters Industry» European consortia and accounts for the bulk of US firms, among which the «Sikorsky» company holds a leading position.

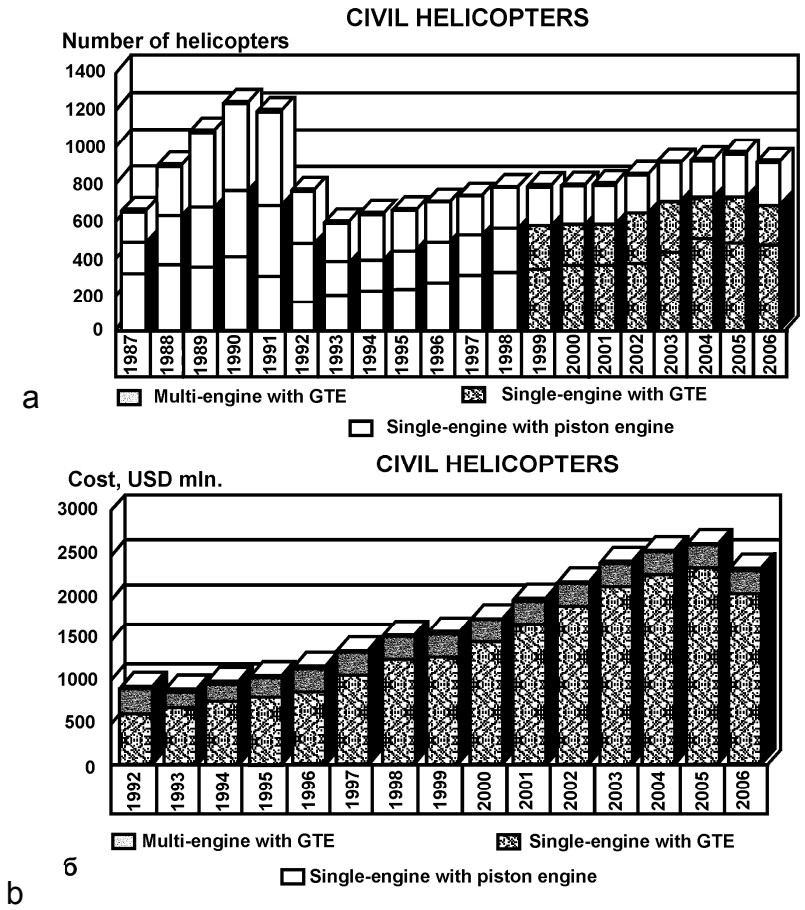


Fig. 1.28. Graphs: (a) Change in Production, Sale and Predicted Supplies of Civil Helicopters; (b) Their Corresponding Prices in 1992 – 2006

Note that all forecasts published abroad are based and reflect the views of the designer, stressing the advantage of his firm or country. DMS is a pro-American outlook and overestimates sales of US helicopters, with which, for example, the «Eurocopter» consortium does not agree considering that they have quite a few opportunities that will significantly expand the sales of European helicopters.

Thus in addition to the DMS forecast Table 1.2 provides information on all major development programs of new helicopters and VTOL with tilt rotors. This table shows the programs of helicopters and VTOLs under development, potential years of serial production and expected number of helicopters or VTOLs to be built under the program, and estimated total cost of the program. For the programs of new military helicopters and VTOL aircraft developed for a long time the specific details of planned purchases are given, such as the program of reconnaissance and attack helicopter RAH-66 or MV-22, CV 22 and HV-22 VTOLs, while for the program of multi-purpose transport helicopter S- 92 maximum possible number of helicopters, which can be ordered, is given.

Similarly, for European and other programs of new helicopters and VTOL: stated

certain figures of purchases of military helicopters are combined with a very approximate data for civilian helicopters and VTOL aircraft. Total number of helicopters and VTOL aircraft with tilt rotors, which may be purchased during the period of 1998-2025, exceeds 15,000, and Table. 1.2 takes into account only the major helicopter programs. Unfortunately, it is impossible to present such a table for Russian helicopters because of the continuing uncertainty of funding. However, as it is evident from the DMS forecast sales of Russian helicopters can take a part of the global market sales.

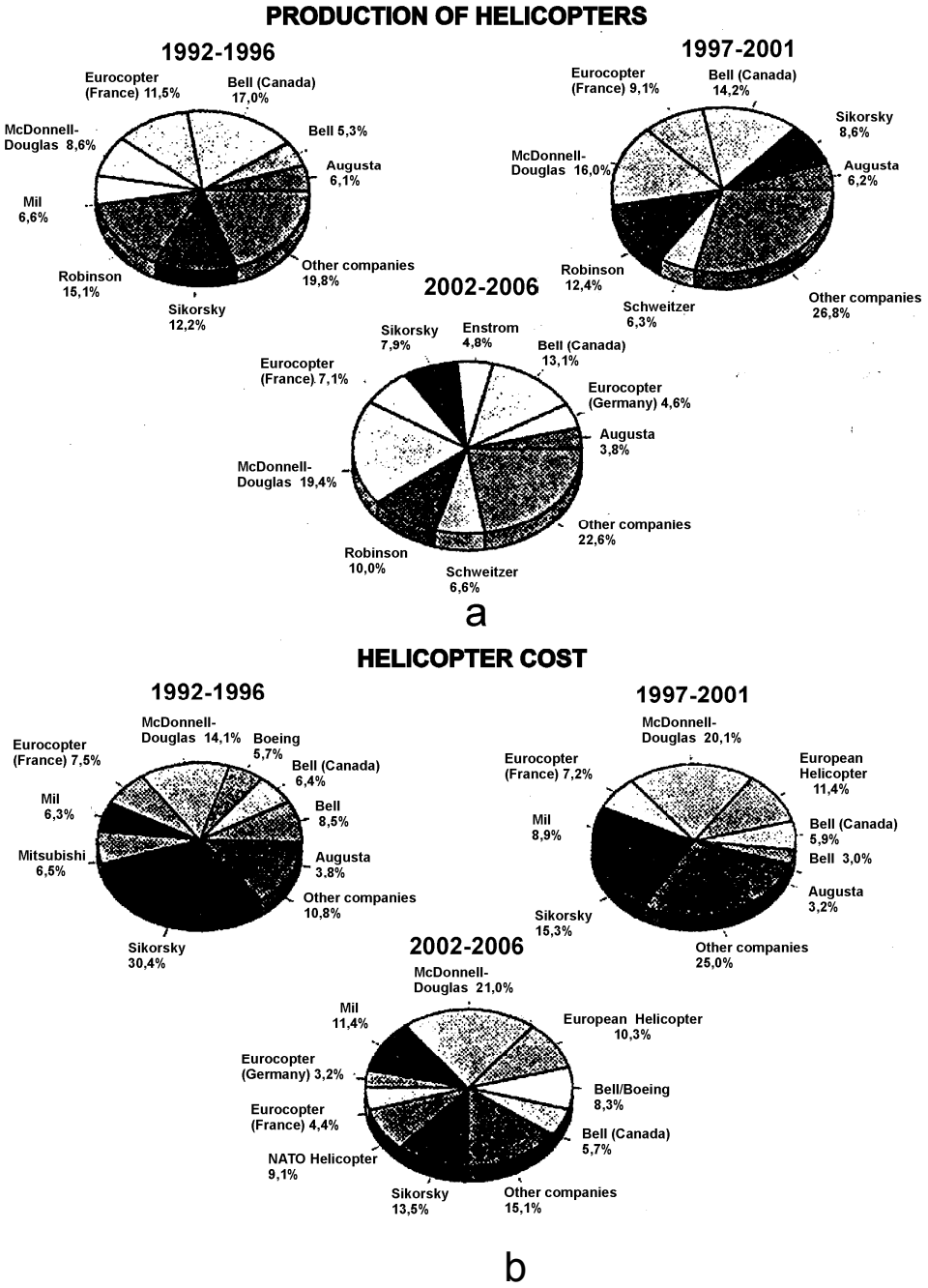


Fig. 1.29. (a) Distribution of Production and Sales of Combat and Civil Helicopters by Different Companies in 1992 – 1996;
(b) Predicted Sales in 1997 – 2006 and Corresponding Prices

Table 1.2

Basic Development Programs of New Helicopters Abroad

Program and Helicopter Purpose	Company	Beginning of Develop-ment	Serial Production, years	Number of Helicopters	Program Cost, billion dollars	Helicopter Price, billion dollars
USA						
Light reconnaissance and combat helicopter RAH-66	«Boeing» – «Sikorsky»	1981	From 2005	1292	35,4	27.4
Multi-role MV-2, aerial delivery CV-22 and search and rescue HV-22 VTOL	«Bell» – «Boeing»	1981	2001-2022	523	45	86
Multi-role transport and passenger S-92	«Sikorsky»	1990	2000-2019	2000-5000		12.5
Business multi-role VTOL BB.609	«Bell» – «Boeing»	1996	From 2001	1000 (ordered 50)		8-10
EUROPE						
Combat helicopter «Tiger»	«Eurocopter»	1984	2001-2025	427		11
Multi-role transport helicopter EH-101	«European Helicopter Industry»	1980	From 1998	700-800 (ordered 82)		
Tactical multi-role helicopter NH-90	«NATO Helicopter Industry»	1983	2003	400 + 600	12	18.7
Light multi-role helicopter EC.135	«Eurocopter»	1991	1996-2007	700-900		4.4
Light multi-role helicopter EC.120	«Eurocopter» «Singapour Technology»	1990	From 1997	1600-2000		0.8
Business multi-role VTOL	«Eurofar»	1987	2004	300		
INDIA						
Light multi-role helicopter ALH	«Hindustan Aeronautic»	1984	From 1998	1000	0.165	4.5

Program and Helicopter Purpose	Company	Beginning of Develop-ment	Serial Production, years	Number of Helicopters	Program Cost, billion dollars	Helicopter Price, billion dollars
JAPAN						
Reconnaissance combat helicopter OH-X	«Kawasaki»	1992	From 1998	150-200	6.85	4.4 (15.8)
Light multi-role helicopter MH-200	«Mitsubishi»	1995	From 2000	200		3.7

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Chapter 2. AIRWORTHINESS REQUIREMENTS FOR TRANSPORT ROTORCRAFT (AP-29) WITH REGARD TO ROTOR DRIVE SYSTEM

2.1. AIRCRAFT CERTIFICATION

Certification means activity intended to prove aircraft compliance with applicable requirements.

Aircraft (AT) means air vehicles and their components: aircraft engines, propellers and OEM components, machinery, instruments, equipment, used to fly the aircraft.

It must be borne in mind that the following legal entities are subject to an obligatory certification:

– Designers and manufacturers of airplanes and other aircraft;

– Aviation companies carrying out and servicing an aerial transportation and works;

– Aviation enterprises engaged in maintenance and repair of aircraft;

– Airfields, airports;

– Educational institutions that prepare the appropriate level of expertise in accordance with the list of posts of aviation personnel;

– Aircraft, aircraft engines, propellers, airborne and ground equipment and other facilities, which activities are directly related to the safety of aircraft and aviation safety.

Certification of aviation equipment is a part of the safety management system aimed at providing access to the operation of aircraft, the appropriate governmental requirements for airworthiness and environmental protection. Compliance with the requirements of the object of certification is certified by a document issued by a specially authorized body, with has responsibilities for the organization and conduct of the compulsory certification of civil aircraft.

Legal bases of certification are established by the Air Laws and Regulations (ALR).

Air Laws and Regulations contain the basic provisions of air law, according to which the laws apply to air laws, presidential decrees, Government decisions, rules for the use of airspace, aviation regulations and adopted in accordance with them other normative legal acts.

Development of aircraft airworthiness standards and certification rules, as well as the certification of aircraft type, its production and initial airworthiness approval of aircraft units with issuance of certificates and equivalent documents are entrusted to the Interstate Aviation Committee, in which these functions are performed by one of its structural units - Interstate Aviation Committee Aviation Register (IAC AR).

IAC is an institutional executive body authorized to execute practical implementation of the Intergovernmental Agreement on Civil Aviation and Use of

Airspace (the Agreement), adopted in December 1991 in Minsk. Under the Agreement, the signatory states (Armenia, Azerbaijan, Belarus, Georgia, Kazakhstan, Kyrgyzstan, Moldova, Russian Federation, Tajikistan, Turkmenistan, Ukraine, Uzbekistan) had established inclusively spheres of joint authority and control work in the field of safety.

The requirements to objects of certification and certification procedures are contained in the Aviation Regulations (AR), adopted in the order established by the States being parties of the Agreement.

Aviation rules adopted by the Interstate Aviation Committee and approved by the Council on aviation and airspace formed by the Interstate Agreement on Civil Aviation and Use of Airspace shall be implemented by relevant executive authorities.

2.2. AVIATION REGULATIONS USED FOR AIRCRAFT CERTIFICATION

Aviation Regulations is a set of requirements, procedures and standards, the implementation of which is a prerequisite for safety and environmental protection.

Airworthiness Requirements (Airworthiness Directives) – a part of the Aircraft Regulations, which contains requirements for the design, parameters, performance and flight characteristics of aircraft, to ensure safety.

Airworthiness Requirements to transport category helicopters that are the part of Aviation Regulations 29 (AP-29) were developed as further development of the first edition of "Airworthiness Directives for USSR Civil Helicopters" (1971) and the second edition of "Airworthiness Directives for USSR Civil Helicopters (NLGV-2)", entered into force in 1987. According to the structure and content AP-29 were harmonized with the US Airworthiness Codes FAR-29 for Rotorcrafts including amendments from 29-1 to 29-32.

Work on the harmonization of national NLGV with US FAR-29 Standards for structure and content requirements was started in 1990.

Numbering of AP-29 sections coincides with the numbering of the FAR-29 paragraphs.

AP-29 requirements, which have no analogues in FAR-29, are administered either in the form of paragraphs, which numbers after digital group have an additional symbol – a large letter of the Russian alphabet, which does not coincide in shape with Roman letters (e.g. 29.610-Б) or inside sections in the form of points, indicated in lowercase letters of the Russian alphabet, which do not coincide in shape with latin letters (e.g. b, c, d, e), or as additional sections.

Harmonizing Aviation Rules - convergence of the domestic system of the Aircraft Regulations with the same rules of the United States and united Europe. The objectives of harmonization are:

– To improve safety of aircraft operations;

– To promote export of domestic aircraft and ensure its competitiveness in the international market;

– Simplification of recognition of national standards abroad;

– Development of international cooperation opportunities in the field of standardization and certification of aircraft airworthiness.

Differences in NLGV and FAR requirements, and especially distinct difference in the nomenclature of the rules and their structure, have made great difficulties in carrying out works on certification of imported aircraft and was the biggest obstacle for the certification of the national aircraft by foreign aviation administrations for its exports.

Analysis of the aviation regulations relating to the aircraft certification and production acting in different countries shows that almost all of these rules are based on the respective US FAR Aviation Regulations (Federal Aviation Regulations) or the combined European JAR (Joint Aviation Requirements). FAR and JAR requirements are mainly harmonized with each other, as the USA and united Europe administrations had carried out and continue work to minimize differences in their aviation rules.

Comparing NLGV requirements with FAR shows that they contain requirements for the level of aircraft airworthiness, which are virtually equivalent.

Since 1990, work began on the harmonization of aviation regulations with US and united Europe ones. The harmonization is based on the following principles:

– System of Aviation Regulations covers all aspects related to aircraft certification and production, and the corresponding nomenclature and structure of the USA and united Europe Aviation Regulations;

– Requirements having the same content in the domestic norms and the FAR, set out in the harmonized aviation regulations in the FAR corresponding text version;

– In the harmonized aviation regulations the distinction of national standards from FAR are retained reflecting the national experience of safety.

Today the national system of aviation regulations is formed, harmonized as much as possible with the relevant USA and united Europe aviation rules. It includes:

– Certification procedures for aircraft equipment and its production;

– Airworthiness Requirements to aircraft, main engines, auxiliary engines and propellers;

– Emission standards (emissions) of pollutants for aircraft engines;

– Aircraft ground noise standards;

– Other aviation regulations to ensure aircraft certification and maintaining aircraft airworthiness.

List of aviation regulations and their designations are given in Table. 2.1.

Aviation Regulations, Part 29. Airworthiness Requirements for Transport Aircraft. AP-29 was developed as further evolution of the second edition of "Airworthiness Requirements for USSR Civil Helicopters" (NLGV-2). AP-29 structure and content are harmonized with FAR-29. This documentation contains airworthiness

requirements necessary to get type certificates and amendments to the certificates for transport rotorcraft.

Table 2.1

List of Aviation Regulations Used for Civil Aircraft Certification

Aviation Regulations	Code
Aviation Regulations, Part 21. Aircraft Certification Procedures	AP-21
Aviation Regulations, Part 23. Airworthiness Requirements for Light Civil Aircraft	AP-23
Aviation Regulations, Part 25. Airworthiness Requirements for Transport Aircraft	AP-25
Aviation Regulations, Part 27. Airworthiness Requirements for Normal Rotorcraft	AP-27
Aviation Regulations, Part 29. Airworthiness Requirements for Transport Rotorcraft	AP-29
Aviation Regulations, Part VLA. Airworthiness Requirements for Very Light Airplanes	AP-VLA
Aviation Regulations, Part 33. Airworthiness Requirements for Aircraft Engines	AP-33
Aviation Regulations, Part APU. Airworthiness Requirements For APU Engines	AP-APU
Aviation Regulations, Part 34. Environmental Protection. Aircraft Engine Emission Norms	AP-34
Aviation Regulations, Part 35. Airworthiness Requirements for Rotors	AP-35
Aviation Regulations, Part 36. Aircraft Noise Certification	AP-36
Aviation Regulations, Part 39. Airworthiness Directives	AP-39
Aviation Regulations, Part 145. Repair Services	AP-145
Aviation Regulations, Part 183. Aviation Register Representatives	AP-183

2.3. AP-29 STRUCTURE

AP-29 «Airworthiness Requirements for Transport Rotorcraft» contains the following sections:

Section A – GENERAL

Section B – FLIGHT

Section C – STRENGTH REQUIREMENTS

Section D – DESIGN AND STRUCTURE

Section E – POWER PLANT

Section F – EQUIPMENT

Section G – OPERATIONAL LIMITATIONS AND DATA

Section A shows the following classification of rotorcraft in terms of their certification categories.

Rotorcraft of transport category to be certified in accordance with the requirements of either Category **A** or Category **B.** Multi-engine rotorcraft can be certified at the same time and under category **A** and category **B**, provided meeting the relevant general and special operating limitations for each category.

Rotorcraft with maximum weight of more than 9080 kg and number of passenger seats of **10** or more must be certified on the type of category **A**.

Rotorcraft with a maximum weight of **more than 9080 kg** and number of passenger seats of **9 or less** can be certified on the type of category as rotary-wing aircraft provided to meet the requirements for Category **A**, contained in sections **C, D, E, F** of this Part.

Rotorcraft with a maximum weight of **9080 kg or less,** but number of passenger seats of **10 or more** can be certified on the type of category as rotary-wing aircraft provided to meet the requirements for Category **A**, contained in paragraphs 29.67 (a) (2), 29.79, 29.1517, as well as in sections **C, D, E** and **F** of this part.

Rotorcraft with a maximum weight of **9080 kg or less** and the number of passenger seats of **9 or less** can be certified on the type **B** of category as rotary-wing aircraft.

Requirements to helicopter transmission with regard to its certification are set forth in section E “POWERPLANT”, subsection “ROTOR DRIVE SYSTEM”.

Let’s consider these requirements.

29.917. Structure

(a) General. Rotor drive system includes all components needed to transmit power from engines to rotor hubs. They are reducers, shafts, cardans, joints, rotor brake units, transmission supports, any accessories or drives, any fan being a part of rotor drive system, adjacent or attached to the system.

(b) Arrangement. Rotor drive system arrangement should meet the following requirements:

(1) Each rotor drive system of multiengine rotorcraft must be arranged so that in case of any engine failure other engines continue working to drive each rotor, which work is needed to fly and to control;

(2) For single-engine rotorcraft each rotor drive system must be configured so that each rotor needed for control in autorotation, continues to be driven by rotors after disengaging the main and auxiliary rotors from the engine;

(3) Each rotor drive system must have a unit for each engine to disengage automatically from the main and auxiliary rotors in case of engine failure;

(4) If rotor drive system uses a device to limit torque, it should be placed so that it provides long-term control of rotorcraft during operation of the device;

(5) If you want to install rotors in phase in order to avoid their crossing, the structure of each drive system should provide constant and reliable phase ratio under any operating conditions;

(6) If special device is provided to install rotors in the respective phase, so the device must be provided for fixing rotors in the correct phase before operation.

29.921. Rotor brake

If there are means for braking the rotor, then management of the system should not depend on the engine, all the restrictions must be given on the use of these provisions and the brake control must be protected against accidental use.

29.923. Tests of rotor drive system and control mechanisms

(a) Long-term tests, general provisions. Each rotor drive system and each rotor control mechanism should be tested in accordance with items (b) - (n) and (p) of this paragraph, at least for 20 hours, plus the time required to meet the requirements of items (b)(2) and (k) of this paragraph. Tests should be carried out as follows:

(1) 10-hour cycle tests with the exception of those cases where the test cycle must be extended in order to conduct the test with one engine inoperative in accordance with items (b)(2) and (k) of this paragraph if such tests are required

(2) tests shall be conducted on the rotorcraft or approved stand;

(3) the amount of torque and speed when tested should:

(i) be determined based on the limits provided for power plant,

(ii) relate to the rotors, approved for use on rotorcraft.

(b) Long-term tests; testing the engine with takeoff power. Running engine with takeoff power must be carried out as follows:

(1) except as provided in item (b)(2) of this paragraph, the engine running with takeoff power must be carried out within an hour by means of alternating cycles of 5 minutes each one with a torque value corresponding to takeoff power and at maximum revolutions per minute, used on this mode, and 5-minute running at almost the lowest possible engine speed with idle power. During the first minutes of the running with idle power the rotor drive system must disengage from the engine and rotor braking system should be put in operation, if it is available or provided. During the remaining 4 minutes of the cycle with idling power a clutch must be engaged with the motor so that the motor drives the rotor movement with almost the minimum speed. The engine and rotor drive system should be accelerated with the maximum rate. After the engine switched off the engine speed should decrease with a rate sufficient to ensure operation of the overrunning clutch;

(2) for helicopters when requesting a certificate for use 2.5 minutes power, with one engine inoperative, engine running with takeoff power must be carried out according to the requirements of paragraph (b)(1) of this section, except for the 3rd and 6th running for which this paragraph provides the magnitude of torque under takeoff conditions and maximum speed in this mode. For this engine running there are stipulate the following requirements:

(i) each running has to include at least one period of 2.5 minutes under torque corresponding to takeoff and maximum speed in this mode;

(ii) each running must include for each engine at least one period in series, during which a failure of the engine is simulated and other engine operates under torque corresponding to 2.5-minutes power (with one engine inoperative), and the maximum rotor frequency corresponds to 2.5 minutes power;

(c) Long-term tests; test with the engine running at maximum continuous power. Continuous engine operation for 3 hours at maximum continuous torque and rotor speed must be performed with the following requirements:

(1) rotor controls to be moved at least 15 times per hour, wherein the rotor set angles correspond to the maximum vertical thrust component, maximum horizontal thrust component directed forward, backward, left and right; controls movement should not cause stress or flapping of the blades, loads in excess of the maximum values or motions, which occur in flight;

(2) directional controls to be moved at least 15 times per hour, alternately passing through the maximum torque limits of the right turn, the neutral point, which provides balancing of the rotor in this mode and the maximum torque of the left turn;

(3) at each position, corresponding to the maximum deflection, controls must be held for at least 10 seconds, and the speed of controls movement must be at least the same as during normal operation.

(d) Long-term test: running the engine with 90% of maximum continuous power. Continuous engine run should be performed for one hour at a torque corresponding to 90% of maximum continuous power, and maximum rotor speed corresponding to 90% to the maximum continuous torque.

(e) Long-term test: running the engine with 80% of maximum continuous power. Continuous engine run should be performed for one hour at a torque corresponding to 80% of maximum continuous power, and the minimum rotor speed corresponding to 80% to the maximum continuous torque.

(f) Long-term tests; running the engine with 60% of maximum continuous power. Continuous engine run should be completed within two hours, and helicopter for which permission to use 30-minute capacity is requested, - for an hour at a torque

corresponding to 60% of maximum continuous power, and the minimum rotor speed corresponding 60% of the maximum continuous torque.

(g) Long-term tests under conditions simulating engine failure. It is necessary to determine whether the failure of fuel or ignition systems, or running engines at different power will cause a dynamic effort that could lead to damage to the drive system. If this is confirmed, it is necessary to perform engine run over a reasonable number of hours under the conditions mentioned above, with one hour of such run should be included in every cycle, and distribute the remaining time over 20 cycles at the end. If there are no adverse conditions, it is necessary to conduct further engine run for an hour according to paragraph (b) of this section and in accordance with the run schedule items (b)(1) of this section without consideration of paragraph (b) (2) of this section.

(h) Long-term tests; running the engine with excess of permissible rotor speeds. Continuous engine run must be performed for one hour at a torque corresponding to maximum continuous power, and the maximum excess of the rotor permissible speed, expected in operation with normal functioning of means limiting the torque and rotor speed, if available.

(i) Long-term tests; main rotor controls motion. If during the test on a leash rotor controls do not move cyclically, you must move the rotor controls, using the procedure specified in paragraph (c) of this section, which provides a position corresponding to the maximum rotor thrust within the following time periods in percent relative to the total test time, in this case controls movement should not give rise to stresses or swinging motion of the blades in excess of the maximum values and traffic loads expected in flight.

(1) under total vertical thrust – 20%;

(2) thrust directed forward, – 50 %;

(3) thrust directed rightward, – 10%;

(4) thrust directed leftward, – 10%;

(5) thrust directed backward, – 10%.

(j) Long-term test enabling clutch and brake assembly. During engine run at a torque corresponding to the take-off power, and, if necessary, during each change of torque and rotor speed for all tests perform totally at least 400 insertions of clutch and brake assembly. Each time the clutch is inserted the shaft on the driven side of the clutch hate to begin rotation, from a state of complete rest. Insertion of the clutch should be carried out at the speed and by method stipulated by the applicant. During deceleration after each clutch insertion the engines must be stopped quickly enough to provide automatic disconnection of engine from rotors and rotor drive system. If rotor is fitted with brake assembly to stop, the clutch before insertion of the brake device must be disconnected when the rotor speed of rotation corresponds to 40% of maximum continuous speed,

after that the rotor brake is inserted. If the clutch design does not permit to stop rotors with engines running or if the clutch is not provided, then before each insertion of braking device stop the engine, and the rotors have been stopped immediately turn it on.

(k) Long-term tests; engine run at torque with failed engine.

(1) Engine run at a torque corresponding to higher power for 30 minutes after an engine failure. If the helicopter is requested permission to use the 30-minute power then engine run at torque and maximum rotation speed corresponding to 30 minute capacity, it should be carried out as follows: sequentially turned off one engine then continuously run the remaining ones for 30 minutes.

(2) Engine run with continuous power after an engine failure. If for helicopter a permission is requested to use continuous power in the event of engine failure, the engine run should be performed at torque and maximum rotational speed corresponding to the continuous power, thereby sequentially one by one off the engine, and then run the remaining engines for one hour.

(3) The number of cycles described in paragraphs (k1) and (k2) of this section shall not be less than the number of engines or at least two.

(m) Any elements that are exposed to wind gusts must be examined under the same flight conditions as the rotors, with their service life to be determined by fatigue tests or other suitable methods. In addition, safety must be ensured, the safety level should be the same as of rotors of:

(1) each element of the rotor drive system that failure would lead to uncontrolled landing;

(2) each element important for setting rotor phase position on the rotorcraft, or element, ensuring operation of the main control of the control linkage in autorotation;

(3) each element, which is common for two or more engines on multi-engine rotorcraft.

(n) Special tests. Each rotor drive system designed to operate with two or more gear ratios must be subjected to a special test for duration of work needed to confirm safety of the rotor drive system.

(o) Each item that has passed the tests specified in this paragraph, at the end of the test must be in the condition suitable for use. During the tests disassembly that could affect the test results is not allowed.

(p) Long-term tests: operating lubricants. To get approval for use lubricants in the rotor drive and control systems, the lubricants must comply brands of lubricants used in the tests prescribed in this paragraph. Additional or alternative lubricants can be assessed by testing or equivalent comparative analysis of brands and characteristics of the rotor drive and control systems. More over:

(1) at least three 10-hour cycles prescribed by this paragraph shall be performed with such temperatures of lubricant in transmission and gear box measured at points intended for measuring temperature, which is not lower than maximum operating temperature for which certification is requested.

(2) for pressure lubrication system at least three 10-hour cycles required by this paragraph shall be conducted under such lubrication pressure (measured at the designated place), which is less than minimum working pressure for which approval is requested.

(3) test conditions contained in paragraphs (p1) and (p) (2) of this section should be applied both to extend their applicability to operate with power for any one engine inoperative, for which approval is requested.

29.927. Additional tests

(a) to determine minimum safety of rotor drive system the dynamic long-term, performance and vibration studies should be performed.

(b) If the gas turbine engine output torque to the transmission can exceed the engine maximum limit and transmission torque, and the output is not controlled directly by the pilot under normal operating conditions (such as in situations where the primary engine power is controlled with the aircraft controls), the following tests should be carried out:

(1) under the operating conditions of all engines 200 starts should be done, each lasting 10 s from the torque equal to at least the lesser value among:

(i) maximum usable torque, corresponding to paragraph 29.923, plus 10%; or

(ii) maximum attainable torque value under possible operating conditions with active torque limiting device, if any, and its normal functioning.

(2) For a multi-engine rotorcraft to simulate cases for each engine failure in turns; remaining engines create maximum attainable torque at the transmission input under operating conditions, provided that the device limiting the torque, if any, functions properly. Each transmission input must be tested with a maximum torque for at least 15 minutes.

(c) Failure of the lubrication system. For lubrication systems required for proper operation of the rotor drive system, the following requirements are specified:

(1) Category A. Except for the circumstances, when failures are extremely unlikely, it is required to show by test that any failure leading to leakage of lubricating oil in any commonly used oil system will not interfere with the safe operation of long-term (though not necessarily without damage) with the values of torque and speed specified by the applicant for the conditions of continued flight. This is the time required for at least 30 minutes after the crew members have detected failure in oil system or oil leakage.

(2) Category B. Applicability of the requirements laid down for category A, except when the rotor drive system can be operated only in autorotation at least for 15 minutes.

(d) Tests in excess of the permissible speed. Rotor drive system shall be subjected to 50 test cycles, each lasting for 30 seconds (± 3), exceeding the permissible speed either not less than the highest speed, expected as a result of the engine control system failure, or 105% of maximum speed taking into account the transients expected in operation. When installing the unit limiting torque and speed, which does not depend on the normal engine operation controls and being a reliable tool, speed must not exceed the permissible limits. Test cycles should be carried out as follows:

(1) Cycles with exceeding the permissible speed must alternate with cycles of stable modes of operation lasting from 1 to 5 min with a speed ranging from 60 to 80% of the value corresponding to the maximum continuous power.

(2) The processes of increasing and decreasing the rotor speed should be carried out over a period of no more than 10 seconds (except in cases where the engine throttle response requires access to high speed for more than 10 s). With that time, specifically reserved for cycles at high speeds, can not be reduced by the amount of the specified time intervals.

(3) Tests in excess of the permissible speed must be carried out with the smallest (in absolute value) blade setting angle in order to achieve smooth flow.

(e) Tests specified in paragraphs (b) and (d) of this section shall be carried out on the rotorcraft and the torque should be taken by rotors, which are installed on the rotorcraft, except in cases where other means of ground and flight tests can be used for other corresponding torque absorption methods, if the conditions of mounting and vibration characteristics closely simulate test conditions on rotorcraft.

(f) Each test described in this section, shall be carried out without disassembling the parts and, except in the case of testing on the lubrication system failure, regulated in paragraph (c) of this section, each test part must be in working order at the end of the test.

29.931. Critical speed of transmission shafts

(a) Critical speed of shaft rotation of any transmission system must be determined by testing, however, in those cases where the analytical methods can be used apply the acceptable methods of analysis for any particular case.

(b) If any critical speed lies within the working range, or close to it in terms of the engine idle power and autorotation, the stresses occurring at that frequency should be within safe limits. This must be demonstrated by tests.

(c) If the analytical methods are used and they show that the critical frequency is not within the allowed operating range, then calculated critical speed must be out of range with a sufficient margin to allow for possible changes between the calculated and actual values.

29.935. Couplings of the transmission shaft

Each universal joint, and other transmission shaft sliding connection, which requires lubrication for operation must be provided with lubrication.

29.939. Engine performance

(a) Engine performance should be investigated in flight to determine that during its operation under normal conditions and in special situations within the operating limitations rotorcraft and engine there are no adverse effects to the engine (such as stall, surge, stall combustion, detonation, invalid parameter values, system malfunctions) in the dangerous degree of their manifestation.

(b) The engine intake unit and other parts of the rotorcraft must not be the reason of dangerous engine vibrations caused by uneven air flow under normal operating conditions.

(c) For engine drive regulator, it must be demonstrated there is no danger of the drive system rotational instability associated with critical combinations of power, rotor speed and movement controls.

Requirements for helicopter transmission lubrication system with its certification are set forth in section E - “POWERPLANT” subsection “OIL SYSTEM”. Let’s consider these requirements.

29.1027. Transmission and gearboxes: general

(a) The oil system for components of the rotor drive system that require continuous lubrication must be sufficiently independent from the engine lubrication systems and provide:

(1) operation with any engine inoperative and

(2) safe autorotation.

(b) Lubrication system of transmission and gearboxes operating under pressure must conform to the requirements set out in sections 29.1013, only paragraphs (c), (d) and (f), 29.1015, 29.1017, 29.1021, 29.1023. Furthermore, the system must have:

(1) The oil strainer or another filter through which the lubricant flows fully, and which should:

(i) be designed so as to remove any contaminants from lubricant that may damage drive components of the rotor transmission system or prevent lubrication process to an extent that can be dangerous;

(ii) be equipped with a pressure relief device, manufactured and installed in such a way that:

(A) lubricant was supplied at a normal pace through the remainder of the system with completely clogged filter;

(B) appropriately placing the bypass device minimizes possibility of contact with the accumulated dirt in the overflow pipeline;

(iii) be provided with means indicating accumulation of contaminants in the filter and the grid in the bypass opening or upstream this point.

(2) On the intake device for each oil tank or sump, through which the lubricant is supplied to the rotor drive systems or their elements, place the safety net to prevent falling into the lubrication system of any object that could obstruct the flow of lubricant from the intake device to the filter, it is required in accordance with item (b)(1). The requirements of item (b)(1) shall not be applied to protective grids installed on the fence device and oil tank sumps.

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Chapter 3. HELICOPTER EFFICIENCY CRITERIA

3.1. EFFICIENCY CRITERIA CLASSIFICATION

At any stage of the design process a developer completing the design and calculation of a new helicopter or its components, continuously determines the level of perfection in comparison with the prototype that is what combination of parameters and characteristics for the object design is appropriate. For such kind of assessment certain criteria are required.

A system of concepts and criteria used to assess the effectiveness of the projects under consideration has been compiled in process of work of design offices and organizations engaged in helicopter operation. A number of works devoted to the development and justification of methods of aircraft valuation mostly airplanes was published. These problems are considered in the most comprehensive way by V. M. Scheinin [4], A. A. Badyagin and E. A. Ovrutsky [1], A. V. Glichev [2], S. A. Sarkisian and E. S. Minaev [6]. Selection of criterion is a complex and challenging task, since it is not always possible to reduce all the variety of helicopter parameters and characteristics to one or more factors. Errors in the selection of criterion lead to unnecessary application of efforts and resources in creation and operation of the helicopter due to the incorrect assessment of the design object. In general, the evaluation criteria have to meet the following basic requirements [5]:

– criterion should be measured (countable) value, which calculation method is well known;

– the criteria should take into account the main purpose for which the object is created as well as the conditions and restrictions of use;

– criteria should include those parameters and characteristics of the object, the effect of which is required to be evaluated or which is need to be optimized;

– at every stage of the design the criteria have to be consistent;

– it is desirable that at all design stages the only criterion to be used.

The last requirement is expressed in the form of suggestions, because sometimes it is difficult to give preference to any single criterion, and solutions need to know several possible criteria.

Specific criterion is selected depending on the work performed. Thus, when calculating the aerodynamic as a criterion it is logical to take the rotor relative efficiency for hovering mode or the helicopter lift-to-drag ratio, when flying with forward speed. Both of these criteria, you can combine under the concept of aerodynamic perfection. In the development of a single unit an important indicator is its weight, referred to a specific characteristic parameter, and for evaluation of the helicopter as a whole – the

ratio of its structure weight to the take-off weight. These criteria may be combined under weight perfection concept.

The most universal and common criterion for the transport helicopter efficiency is the ratio between the useful work performed by it, and the total costs for its creation and operation.

However, such wide assessment of the newly created aircraft is not always possible, especially at the early design stages. Therefore, along with the more general terms of the project assessment the particular criteria and their combinations are used.

Note that comparison of the different aircraft using both general and particular criteria does not yet allow to make a definitive conclusion about the most advantageous aircraft as often such criteria do not estimate very important features, which have certain utility value. So, for transport operations when comparing airplanes with helicopters the latter usually lose.

Helicopters have a very important feature for consumers: the ability to take off vertically, perform a variety of works in hovering, and that makes them indispensable, despite the fact that the cost of transportation of 1 t·km of them is higher than that of the airplane. Therefore, the same criteria should be applied to objects having the same function and used in similar conditions.

Considering the basic criteria or methods of assessment let's divide them according to the generality into three levels.

At the *first level*, you can select criteria for functional, production and finally operational efficiency.

Functional efficiency characterizes completeness and level of technical (inclusive weighting) perfection this helicopter performs its main task. So, for the transport and passenger helicopters the transporting efficiency is accounted, for cranes helicopters – the efficiency of crane operations.

Production efficiency makes it possible to assess how perfect is the project from the point of view of technological design and other conditions associated with the production.

Operational efficiency allows evaluating various options for the project, or different vehicles in terms of their performance.

The *second level* includes criteria to evaluate integrally individual aircraft characteristics considering its function, manufacturing and operational efficiency. Such comprehensive assessment of the project can be done using economic methods.

Most often in practice the cost of transportation of 1 ton per km for transport vehicles, the cost of treatment of 1 hectare for agricultural aircraft and the cost of 1 hour operation for crane-helicopters are used as such criteria.

The *third level, the highest level criteria* include methods of systematic, generalized economic evaluation of the efficiency not of a single plane or helicopter, but

the entire set of vehicles of this type or several types, created to solve specific problems. The A.V. Glichev publication adequately set forth a systematic approach to solving these problems [2]. The problem of selecting the optimal options of aircraft systems based on a comprehensive economic evaluation of the work is represented in works of S.A. Sarkisian and E.S. Minaev [6].

3.2. FUNCTIONAL EFFICIENCY

The task of determining technical and economical characteristics of the transport helicopter with the help of clear and concise formula is always seemed to be very tempting. Therefore, solution of this problem has been given a lot of attention both in our country and abroad.

All variety of the functional efficiency criteria is usually reduced to some combination of three determining factors given below:

– cruising speed at which helicopter performs a job;

– weight perfection;

– energy perfection linking fuel consumption needed to perform the transport operation with amount of this work.

Each of these factors may be used separately as a particular criterion in solving certain problems. However, the most complete are the criteria that take into account all these factors.

Cruising speed. Features of the helicopter rotor operation cause fundamentally possible range of its flight cruising speeds. During one revolution a rotor blade passes through two zones: within the first zone the incident flow speed is added to the rotation speed, and within the second zone it is deducted. Therefore, with increasing flight speed in a zone in which the speed is added, there are areas where drag begins to increase rapidly due to approaching to the sound velocity, which causes increase in the required power. Reducing the rotational speeds in the second zone causes other restrictions associated with an increase in angle-of-attack and leads to disruption of flow. These circumstances determine helicopter maximum cruising speed.

During the entire period of helicopter development the flight cruising speed has gradually increased, which is explained as follows:

– increased available power-to-weight ratio;

– decreased drag of non-carrying elements;

– improvement of the rotor aerodynamic characteristics.

Currently cruising speed is of 160 km/h for ultra-light vehicles with piston engines and is up to 270... 290 km/h for heavy helicopters fitted with gas turbine engines.

Helicopter weighing perfection. The most commonly criterion to assess the weight perfection is weight ratio**,** which is widely used in practice. It is determined by dividing the difference between the helicopter take-off weight m_{t-off} and empty weight

m_{emp} by the helicopter take-off weight m_{t-off} or the ratio of full load weight m_{total} to take-off weight m_{t-off}

$$k_{we} = \frac{m_{t-off} - m_{emp}}{m_{t-off}};\ k_{we} = \frac{m_{total}}{m_{t-off}}. \qquad (3.1)$$

Increasing the weight ratio is an important means of increasing the static ceiling and reduced transport helicopter performance.

If we consider the weight ratio variation with time for helicopters (Fig. 3.1), we can see that this factor tends to some increase with time, indicating steady progress of the weight perfection. However, with time, rate of improving the weight ratio decreases.

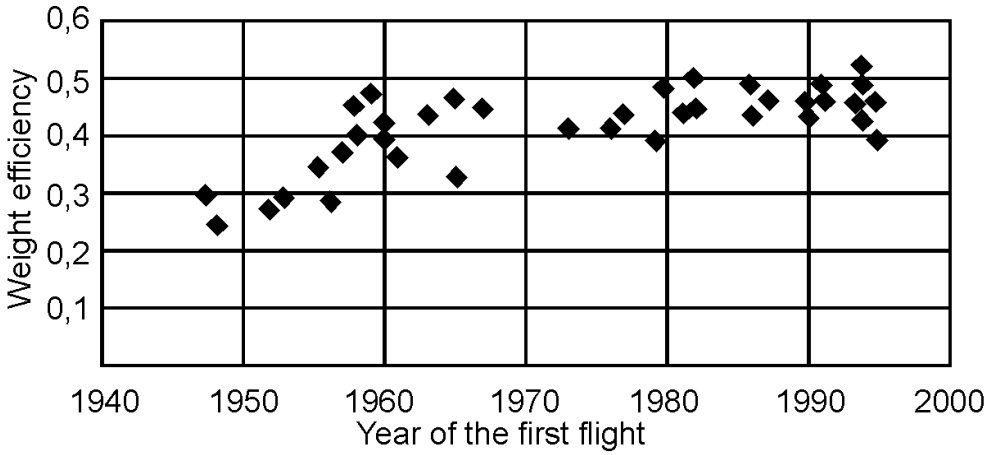


Fig. 3.1. Weight Ratio Variation with Time

It may be noted a number of reasons, contributing to increase the helicopter weight ratio:

– transition from piston to gas turbine engines, widespread use of new materials (titanium, composite materials, etc.);

– application of new technological processes (weldbonding, chemical milling, machining with CNC machines etc.);

– improving methods for assessing strength of structures and materials.

Currently main ways of increasing helicopter weight ratio are:

- Choice of rational load-carrying structures and layouts for helicopters and their components;
- Integrated optimization of helicopter and its units parameters in design process;
- Use of new materials;

- Application of new technological processes in the manufacture of components and assemblies;
- Creating designs with uniform strength and use of materials with high endurance limit;
- Reduction of acting loads in the construction of components and assemblies.

Thus, increase in helicopter weight perfection depends on solving the whole complex of main design tasks in engineering and manufacturing. Minimum helicopter empty weight, and therefore, high weight perfection is achieved only as a result of optimal solutions of development and construction problems listed above.

Note that design office opportunities involved in the helicopter design process, as a rule, are limited to the construction share corresponding to 50... 80% of empty helicopter weight, while the remaining 20... 50% of weight belong to vendor components. Nevertheless, despite the fact that increase of weight ratio has a limit it can be expected that some reduction in the helicopter empty weight will be reached soon.

Weight ratio is often used as a measure of the weight perfection of a particular helicopter, and for comparison of several vehicles, as it clearly characterizes the helicopter design weight perfection under consideration.

Naturally, while designing new aircraft somebody always seeks out the highest weight ratio. More over at stage of working draft of a particular design, which basic parameters have already been identified, the task of obtaining the minimum weight becomes one of the main ones.

To a large extent weight ratio is determined by weight characteristics of major components of power plant and complex equipment. Therefore, when choosing the helicopter parameters it is required to take into account factors that determine the weight of these systems.

However, application of weight ratio in order to optimize the weight of the transport helicopter can lead to error. For example, two helicopters equal in take-off weight, having equal values of weight ratio but not the same specific fuel consumption (due to differences in lift-to-drag ratio or power plant consumption) are unequal in terms of transport output rate. As a result, the weight ratio, as an absolute parameter that characterizes structural weight perfection, can not be used during the preliminary design stage as a synthesis criterion for the aircraft transport efficiency under design.

Helicopter power perfection. For further research it is necessary to link the range with the amount of fuel consumed to move over this distance.

Using helicopter lift-to-drag ratio concept and rotor propulsive factor [3], we obtain the following expression for the engine total power in kW needed for level flight at cruising speed:

$$N_{eng} = \frac{m_{fl} g v_{cr}}{3600 K \eta_{pr} \xi}, \quad (3.2)$$

where m_{fl} – weight, kg; v_{cr} – cruising speed, km/h; K – helicopter lift-to-drag ratio; η_{pr} – main rotor propulsive factor; ξ – engine output rate factor.

Multiplying the power by the specific fuel consumption, we get hourly fuel consumption, and dividing it by the speed of flight, - fuel consumption per 1 km, kg/km:

$$q = \frac{c_e}{3600 K \eta_{pr} \xi} m_{fl} g. \quad (3.3)$$

Having divided consumption q by weight we got specific consumption per 1 km:

$$\bar{q} = \frac{c_e g}{3600 K \eta_{pr} \xi} = \frac{1}{E}. \quad (3.4)$$

Specific consumption $\bar{q}$ or inverse value of E is suitable for use as a criterion for assessing the degree of fuel usage perfection of any vehicle. Flight range due to consumption of a certain fuel reserve is the most frequently used feature of such means. Therefore, knowing the flight weight you can easily determine this parameter.

Since the larger E the higher fuel efficiency, this value may be an index (coefficient) of the vehicle power perfection.

The physical meaning of the energy perfection index - this is the path that the vehicle can pass due to consumption of fuel weight equal to the flight weight, while moving horizontally at a constant cruising speed.

Energy perfection is very useful for comparing the amount of energy required in the performance of the same work by different transport vehicles.

Output rate. If you want to carry out a large transport operation and for which you need to make a lot of flights, then to determine the timing of this operation and the required number of helicopters it is necessary to know the work done by one helicopter at a time, i.e., this helicopter output rate.

To calculate the output rate of the transport helicopter you should be aware of the transport job A (the name is applied to a variable that is not a mechanical job in the strict sense of the term), determined as

$$A = \sum_i (m_{crg})_i \cdot L_i \quad (3.5)$$

or

$$A = (m_{crg})_{av} L_\Sigma,$$

where $(m_{crg})_i$ – weight of cargo transported in i-th flight; L_i – range of i-th flight; L_Σ – total distance of all flights; $(m_{crg})_{av}$ – average weight of the transported cargo.

Performance of the transport helicopter is assessed by transport job done in the unit of time:

$$W_{av} = A / T_\Sigma, \tag{3.6}$$

where T_Σ – total calendar time spent for the transport operation A.

The operation total calendar time is the sum of the flight time T_{fl}, time T_{l-ul} spent for loading and unloading, time T_{mt} ,required for maintenance, downtime T_{dt} due to limitations of various kinds and, in particular, to restrictions on the medical standards of flight time for flight-lifting crew, weather conditions, supply of fuel and lubricants, as well as elimination of detected faults:

$$T_\Sigma = T_{fl} + T_{l-ul} + T_{mt} + T_{dt}. \tag{3.7}$$

Obviously, downtime characterizes the work organizational level in the operating organization and reflects the helicopter reliability and dependability degree.

Time spent for maintenance determines the performance characteristics of the helicopter.

Let's analyze the flight output rate, only determined by the flight time T_{fl}. The flight output rate is the most important component of the total output rate depending on the helicopter performance and weight characteristics. More over it characterizes the total transport output rate of the helicopter, as the service life of all units and aircraft engines is calculated in flight hours.

Normally the flight output rate is calculated taking into account the loss of time spent for maneuvering during takeoff and landing, as well as the amount of time needed to climb and descent. This reduces the average flying speed, making it less than cruising speed. Flight performance

$$W_{av} = \frac{(m_{crg})_{av} L_\Sigma}{T_{fl}} = (m_{crg})_{av} V_{sch}, \tag{3.8}$$

where V_{sch} – scheduled speed.

Usually scheduled speed is related with the cruising speed by factor k_v $(V_{sch} = k_v V_{cr})$, that allows us to express the output rate as follows:

$$W_{fl} = k_v m_{crg} V_{cr}. \tag{3.9}$$

Due to the fact that the vast majority of the papers deals only with flight output rate, further we assume this parameter as an output rate. Maximum output rate of a given helicopter is a very important criterion for its transport perfection. It is obvious that

for calculation of the maximum performance you need to multiply maximum weight of cargo $m_{crg\,max}$ carried over the predetermined distance, by the scheduled speed V_{sch}:

$$W_{fl\,max} = m_{crg} V_{sch}. \tag{3.10}$$

Note that in addressing the specific design when the designer has a job with definite values of cruising speed and cargo weight to be transported to the predetermined range, output rate may not be a criterion for optimization, as all comparable options that meet specified conditions, will have almost the same output rate.

This difficulty can be overcome, if we consider the specific output rate, i.e., the ratio of output rate to the helicopter empty weight:

$$\overline{W}_{flc} = \frac{k_v m_{cargo} V_{cr}}{m_{flc}}. \tag{3.11}$$

The specific output rate can also be determined by dividing the output rate by the take-off weight m_{t-off}

$$\overline{W}_{flc} = \frac{k_v m_{cargo} V_{cr}}{m_{flc}}. \tag{3.12}$$

Thus, it is possible to obtain a criterion for assessing the project in terms of its weight perfection with output rate in mind. This is more generalized criterion than the weight ratio or cruising speed alone, as it takes into account influence of both these factors. However, energy perfection in this criterion is not reflected explicitly; it manifests itself through the influence of flight range upon weight of the maximum carried cargo for which the output rate is determined.

Weight ratio with regard to payload transported over predetermined distance. Quite often as one of the main requirements for newly designed helicopter the cargo weight is given that must be transported to the specified distance.

The criterion for appraising the ability of such a helicopter is a weight ratio with regard to payload m_{pl} at a fixed distance:

$$\overline{m}_{pl} = \frac{m_{cargo}}{m_{t-off}}. \tag{3.13}$$

The value $\overline{m}_{pl}$ can also be considered as a criterion that accounts the weight perfection and energy perfection of the projected helicopter. However, it does not account its output rate. Helicopters with equal loads and flight ranges, but different cruising speeds, will have different values of the output rate with the same weight ratio factors with regard to payload, carried over the same distance.

Reduced output rate. Above we have listed the main factors that should be accounted by the criterion of transport efficiency. Considering these factors, the generalized criterion requirements can be summarized as follows. Among the considered options and performed tasks of the designed transport or passenger helicopters choose one that has the best output rate and the best weight and power perfection.

As the parameter characterizing performance and weight perfection in flight over the predetermined range choose relative performance as determined by formula (3.12). Takeoff weight determines dimensions of the helicopter and its main units. Therefore, output rate, attributed to the take-off weight, is objective and independent of the flight weight criterion that characterizes the output rate of the transport or passenger helicopter per unit weight.

To evaluate and compare the projected helicopters with regard to fuel consumption use specific fuel consumption in accordance with formula (3.4).

Dividing specific output rate by specific fuel consumption, it is possible to obtain a criterion for assessing the effectiveness of the transport helicopters taking into account three factors mentioned above – ***reduced output rate*** (Scheinin criterion):

$$\tilde{W}_{fl} = \overline{W} / \overline{q}. \tag{3.14}$$

The helicopter reduced output rate with takeoff weight m_{t-off} carrying load m_{cargo} over distance L at speed $V_{sch} = k_v V_{cr}$ having specific fuel consumption $\overline{q}$ or power perfection factor E can be represented as follows:

$$\tilde{W}_{fl} = k_v \frac{m_{cargo} V_{cr} L}{m_{fuel}} = k_v \frac{m_{cargo} V_{cr}}{\overline{q}}; \tag{3.15}$$

or

$$\tilde{W}_{fl} = k_v^2 \frac{m_{cargo} V_{cr} E}{m_{t-off}}. \tag{3.16}$$

The physical meaning of the criterion (3.15) [7] shows what kind of output rate will have a conventional helicopter, having the same cruising speed, perfection by weight, fuel specific consumption and lift-to-drag ratio, but different in take-off weight. And take-off weight of the conventional helicopter should be such providing fuel consumption of 1 kg per 1 km.

Expression (3.15) is convenient to calculate the reduced output rate for already created or designed helicopter, while the expression (3.16) is more convenient to assess the impact of individual component parameters varied in the design process on $\tilde{W}_{fl}$.

Fig. 3.2 shows the function between reduced output rate $\tilde{W}_{fl}$ and flying range L of the Mi-6 helicopter. When the distance is less than L_{opt}, despite the increase in payload the scheduled speed reduction leads to performance degradation.

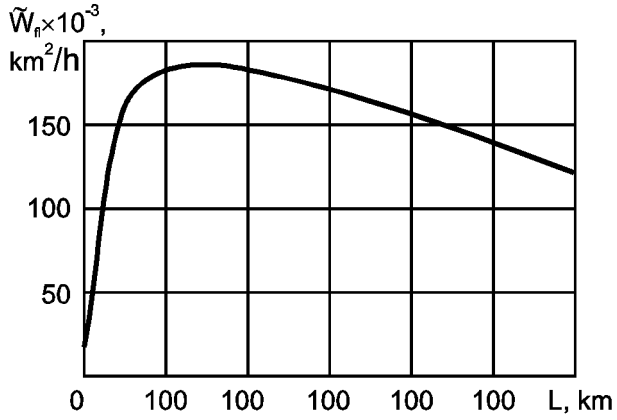


Fig. 3.2. Reduced Output Rate $\tilde{W}$
vs Flight Range L for Mi-6 Helicopter

For ranges longer than L_{opt} the output rate degradation is caused by decrease of payload.

Several values of the reduced output rate for serial domestic helicopters depending on the year of their production are shown in Fig. 3.3. Each new helicopter besides that it meets the requirements for carrying capacity, provides high reduced output rate.

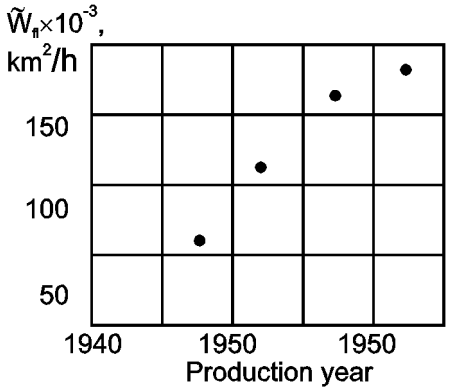


Fig. 3.3. Reduced Output Rate $\tilde{W}_{fl}$ of Domestic Helicopter Production

According to [7], the values of the criteria $\tilde{W} \cdot 10^{-3}$ for helicopters with flight range of 300 km are: Mi-4 – 35, Mi-26 – 167, Mi-8 – 157 km^2/h. For helicopters produced in recent years the value of $\tilde{W} \cdot 10^{-3}$ is of 300... 360 km^2/h.

Reduced output rate may be used as a criterion for evaluating output rate of the considered helicopter and its weight perfection taking into account the fuel consumption

efficiency. This criterion can be applied when considering the various options of the transport helicopter under design, evaluation and comparison of the selected option or passenger transport helicopters of various weight categories.

When using a transport helicopter for transportation of oversized cargo as crane helicopter its efficiency in such work can be estimated by expressions (3.15) or (3.16).

However, its use for erection or crane operations associated with work mainly in hovering with the conditions required to ensure duration of hovering, and work safety in case of failure of one engine. Based on the conditions of the job done by crane helicopter there are three variants of the helicopter take-off weight ratios and therefore the target load:

– operation using engine take-off power. In this case, the required erection work cycle must be performed within the time limits allocated for continuous operation at takeoff power. The weight of lifted load is the greatest;

– when using the nominal engine power the load weight will be less than in the first case;

– in case of one engine failure weight of the lifted load is determined by the operating conditions of the engine remaining at the highest possible power. In this case, it may be that the weight of the payload will be close to zero.

Therefore the criterion for evaluating the effectiveness of the crane helicopter should be focused on these kinds of works that are performed mainly in hovering.

As a criterion of optimizing the parameters in this case use the specific output rate $\tilde{W}_{crane}$, which for crane helicopter is expressed by the ratio of relative performance $\bar{W}_{crane} = K_{fuel} \cdot \frac{m_{c.n}}{m_0} = K_{fuel} \cdot \bar{m}_{c.n}$ to the hourly fuel consumption $\bar{q}_h = q_h / m_0$ [7].

Reduced output rate of the crane helicopter

$$\tilde{W}_{crane} = \frac{\bar{W}_{crane}}{q_h} = K_{fuel} \frac{m_{c.n}}{q_h}. \qquad (3.17)$$

Using the relationship between power and weight of the vehicle in the hovering mode (Velner formula), we have

$$\tilde{W}_{crane} = 37.5 \cdot \eta_o \cdot \xi \cdot K_T \frac{m_{c.n}\sqrt{\Delta}}{C_e \sqrt{p}}. \qquad (3.18)$$

Here, the coefficient $K_T = T_{erec} / T_{fl}$ takes into account the difference between time T_{erec} spent on erection, and flight time T_{fl}, including time spent for maneuvering before takeoff and after landing, as well as the time required for the approach to work

and returning back. When the specific conditions of the crane helicopter are unknown then K_T=0.5 can be taken on the basis of statistics.

3.3. PRODUCTION EFFICIENCY

The concept of technological design of a product includes production efficiency as an integral part.

Technological design of the product is defined as a set of properties of the product design, exhibited in the best possible cost of labor, tools, materials, and time with the technical preparation of production, manufacture, operation and maintenance as compared with those of similar designs. This definition includes qualities that characterize the operational efficiency of the project (at the pre-production and production) and operational efficiency (operation and maintenance).

When considering the production efficiency of a particular project must be in the general case to evaluate its labor intensity, capital intensity, material consumption, as well as the time spent on pre-production and production cycle.

At all stages of the aircraft design a designer should strive for the most economical use of materials and obtaining the highest rate of use.

The particular form of this problem solution depends on the stage of development. As a rule creation of serial samples is preceded with construction of experimental prototypes used to verify correctness of the design laid down in the decision and also to rework components and assemblies. Therefore, technical documentation, issued in early development, often designed for an experienced process technology, which is created as a result of only a few copies of the product. At this stage, it is not always possible to obtain high values of the material use coefficient. Meanwhile, the major costs associated with the construction of the planned helicopter, to be determined in the process of weight production. Therefore, it is extremely important during designing including the early stages carefully consider terms of the future series production and the most advanced and provide appropriate processes in the structure.

Using billets for structures obtained by precision casting, precision stamping and other technological methods and do not require further large machining, used in non-critical, non-power elements of plastic parts, and power components glass - and carbon fiber materials as well as many other solutions allow to obtain high material use coefficient.

In the process of designing an aircraft a designer constantly have to resolve the contradiction between the requirements to obtain a minimum weight and to ensure acceptable labor intensity in weight production and in operation. When analyzing the structure of various embodiments of an assembly or unit it is important to compare them in terms of the complexity of manufacturing.

3.4. OPERATIONAL EFFICIENCY

Duration of operation of any aircraft is much greater than the time of its design or construction. During the operation that useful work is realized for which each helicopter or aircraft was created. However, man-hours associated with operation are considerable. That is why in recent years improvement of operational conditions and methods has received an increased attention.

Each of these factors alone is widely used in practice and is well known. The main difficulty in the use of these values at the pre-design or working design stage is that at this point all the necessary details and the small details that are required for specific calculations are not fully known.

Obviously, if it would be possible to develop simplified quantitative criteria for evaluation of a project, it would be possible to get even at the early design stages a certain way of choosing the best option among the possible ones.

During operation, as well as during execution of maintenance works all sorts of deviations or damage to structural components of aggregates, components, systems and equipment are found and eliminated by adjustment, refinement or replacement of damaged items. Work on the elimination of these faults is called repair. The main feature of these works is that they can not be planned in advance, so we put effort on such work to the second labor group. Clearly, determination of the specific effort can be obtained only as a result of operation analysis. Therefore, the complexity of the specific repair in the design process can not be evaluated.

Now in practice of aircraft construction and helicopter engineering it starts to standardize work hours need to perform replacement of the basic units of the helicopter and power plant. It is considered as appropriate in the event of any serious malfunction or significant deviation of the system parameters not to deal with their repair in an aircraft, and replace the faulty aggregate in a whole, and then in a specialized enterprise to eliminate the defect in the removed part or unit.

In this regard, particular attention is paid to creation of the so-called modular construction, providing relatively simple breakdown of the unit into separate modules, followed by replacement of the damaged module.

Thus, in order to achieve the highest perfection of design in terms of ensuring minimum labor input of repair it is very important to perform the design so that the complexity of its replacement with individual units or modules is minimal.

The third group of works related to the operation includes an overhaul. If work on the replacement units can be regulated, the restoration work associated with the elimination of the part wear and similar is impossible to be pre-normalized.

When considering optimization of design decisions at early design stages it seems to be appropriate among a large number of criteria proposed by different authors, select

only those that the designer can use in work on the project to assess perfection in terms of performance of the newly created prototype (helicopter or a single unit).

Labor costs associated with maintenance, can be divided into three groups. The first group includes those running time which is planned in advance, the second includes those, the execution of which can not be planned, and, finally, the third includes labor costs, for which the term of execution is known, and the work content and the associated labor costs are known in advance only partially, and further refined in the course of the above works.

Current practice of operation and repair of aviation equipment provides for the implementation of well-defined work to ensure reliable operation of the aircraft. These include pre-flight, post-flight inspections, post-flight training, as well as periodic and seasonal maintenance work. The content of these works lies in the inspections, filling, lubricating and various regulated checks.

The distinctive feature of these works is that their complexity can be determined in advance, even in the design process. Moreover, it is to some extent determined by the design process.

Thus, the above operation, aimed at maintaining operability of a helicopter, defines maintenance content, which is the first group of the planned work during operation. Labor-intensiveness, i.e. number of man-hours required for maintenance can be defined even on the mock-up of the helicopter by the timing of certain operations to be carried out in accordance with the operational documentation.

During the development of the assembly or unit structure it is necessary to provide for special arrangements needed to perform the work during overhauls. These include the provision of margins in the sections for the possibility of applying the repair of elements with dimensions of more than nominal, providing conditions for the replacement of bearings and so on.

It should be emphasized that the labor costs associated with maintenance, repair and overhaul, largely depend on such seemingly small circumstances, as the convenience of location of inspection doors and methods of their closing, convenience and ease of monitoring of liquid levels, ease approaches to units, possibility to remove a single unit without dismantling the other, and so on.

Thus, to ensure high operational efficiency of the project it is necessary to attain the lowest values of specific maintenance labor intensity, to reduce labor costs for the replacement of major components and systems, as well as to reduce the complexity of overhauls.

3.5. ECONOMIC EFFICIENCY OF HELICOPTER OPERATION

The above criteria allow – when studying the helicopter under design or already built – to assess it from the point of view of the individual functional, manufacturing or

performance features. However, an integrated assessment of the project, taking into account on a single basis all features of the vehicle, can be obtained using only an economic assessment, determining the cost of its operation: a single result, assessing the costs of work carried out.

Cost of work performed by the helicopter, is a recognized criterion for the efficiency of its use. This approach is not new. Back in the 30s, at the time of formation of passenger traffic, a more or less accurate calculation of the cost was revealed to be quite cumbersome and hard, especially when it came to considering a range of options to optimize solutions.

Currently, the solution of this problem is greatly facilitated by the use of digital electronic computers for the engineering calculations. This allows for the use of direct methods for calculating the cost of works carried out by helicopters. The accuracy of these estimates is directly dependent on the correctness of the original data. And if in practice some discrepancy is found between values predicted in the design process and the resulting actual operating numbers, the reason for this discrepancy is the most often associated with imprecise service terms laid down in the calculation, as well as with changes with time in prices and wage rate, refining certain details of the maintenance process and repair time.

However, when it comes to optimizing the parameters of the newly designed aircraft, all the factors that cause the above error, are the same for the options under consideration and will not affect the results of the comparison.

The basis of the aircraft valuation is the cost of one work hour. It is accepted to share (or group) operating costs into two parts - both direct and indirect. According to E. A. Ovrutsky [1], this grouping is due to the fact that the level of certain costs directly relates to the performance of the vehicle data (its weight, engine power, load output rate, fuel consumption, technological design, and others.); other expenses only partly conditioned by the type of aircraft operated (such as depreciation and maintenance of runways), to a greater extent as they are dependent on the total volume of traffic, traffic density, and other factors that are not directly associated with aircraft performance data. Therefore, when designing a new project designer can affect substantially only the first group that is on the direct costs. Moreover, in most cases, the indirect costs are defined as a percentage of the fundamental, and therefore only the direct operating costs can be used to optimize the design.

Knowing the hourly operating costs, you can get the cost per unit of useful work performed by dividing the costs of one flying hour on helicopter performance:

$$C = C_{fl.h} / W, \tag{3.19}$$

where C - cost of work performed; $C_{fl.h}$ - cost of the flight hours of the helicopter; W - efficiency of the helicopter.

Paper [7] sets out the method for calculating the cost of operation of the helicopter. The development of this technique specifically for helicopter has been caused by significant differences in operating conditions.

3.6. SCOPE OF APPLICATION OF DIFFERENT LEVEL CRITERIA

To get optimum parameters of future helicopter, to assess the impact on its quality of any changes occurring in the project during engineering, construction and finishing phases it is necessary to monitor the aircraft basic parameters for all works and each phase needs its own criteria.

Selection of criteria level or one among the selected level criteria for the assessment of work perfection is largely defined by the project stage of development. Considering the period of development and construction of the aircraft, since the inception of the idea of such a vehicle, and ending with the completion of the flight tests, there are three main stages in the process of its creation.

The first stage includes development of requirements for new aircraft and justification of its basic characteristics. With regard to the helicopter at this time such important characteristics are formed and given as load-carrying capacity, flight range, cargo compartment dimensions, cruising and maximum speed, static and dynamic ceilings, number and type of engines, reliability, performance (for example, required lifetime of basic units), complexity of maintenance and frequency of major maintenance work. The first phase ends with creation of the project tactical and technical requirements to the newly created aircraft.

Since the set of requirements during this period is formed by organizations carrying out orders, purchasing and further operation of the helicopters, they primarily develop methods to optimize the tasks.

During the first phase of the aircraft development, when not only the idea of creating a vehicle having specific dimension appears, but also an element of future transport system is formed, taking into account total number of helicopters, the intended production by years, loading during operation and date of decommissioning, it is appropriate to use the third highest level of criteria, allowing for the maximum possible number of basic factors defining the efficiency of the system in the entire national economy.

The methods used for solving this class of problems are discussed in detail by S. A. Sargsyan & E. S. Minaev [6] and A. V. Glichev [2]. Method of complex economic assessment is the most complete and efficient when used for economic resources of the country in solving the problems. This takes into account:

- Prospects for the development of technology, its technical efficiency, output rate;
- Terms of development, implementation and operation of new products;

- Industry production capabilities and prospects for the development of machinery and production technology;
- Duration of cycles of creating new products and production complexity;
- Current level and trends of further changes in concentration, specialization, cooperation and location of production;
- Resources and prospects in production of construction materials;
- Rational use of material and human resources provided by industry;
- Level and prospects of development of technical equipment and operating conditions, and the use of new products;
- Rational use of material and human resources in the field of aircraft operation.

Such problems are not inclusive in the problems to be solved in the Experimental Design Office. They must be solved by the customer's research institutes, for example, when developing the aircraft optimal sizes, which belongs to the particular transport system, and then in the development of tactical and technical requirements for each type of aircraft, which is a part of the developed system.

The second stage begins with the study of tactical and technical requirements in the Experimental Design Office. This period is characterized by the works of a very close interaction between the design office and the customer. To provide that requirements are feasible and based on a specific elaboration, a preliminary draft or as it is called, a technical proposal is typically performed by the design office. Already in the process of this design study it lays the basic design, aerodynamic and weight parameters of the future helicopter, its components and systems. On the basis of this study some provisions of the tactical and technical requirements are clarified, taking into account actually industry opportunities existing at a given time. This period ends with the final generation of the tactical and technical requirements. Then the period of conceptual design and layout of the developed system comes. At this point, finally the helicopter basic parameters are determined, individual systems are refined, and the work ends with creation of a full-scale mockup, which allows for final definition of the designed helicopter. The mockup allows evaluating suitability of the cargo compartment to load and accommodate cargo planned to be carried, ease of placement of passenger seats, cockpit, arrangement of devices and controls, review and so on. The mockup is also extremely important for the future work on design solutions (e.g., for convenience of maintenance approaches), as well as the time required for the replacement of certain units can be noted and mutual arrangement and operating conditions of major components and systems are to be verified.

During the works that make up the content of the second stage, the bulk of the work associated with the future aircraft parameter optimization is carried out, which provides a set of TTR to the performance characteristics of future vehicles. As a criterion for the solution of this problem in the case of the transport helicopters the most

appropriate criteria are the pay-load for the given flight range and take-off weight and the weight ratio for a given range.

It is advisable to consider possible solutions, used for a transport vehicle, such as transport efficiency criterion proposed earlier. As a rule, when considering such different results obtained for the solutions placed on different sides from the extreme value with regard to useful weight ratio. Thus, the domain of feasible solutions is narrowed. For this narrowed area it is advisable to calculate the cost of work performed in order to obtain the final optimal solution.

In this manner it is advisable to carry out calculations in the development of technical proposals and conceptual design, as well as refinements to the design mockup.

As a result, works on the mockup done by the design office and the customer, form a document, which to the maximum extent fully defines the basic characteristics of the future helicopter. This document allows the designer to move to detailed design, beginning the third and final stage of work on the creation of the new aircraft.

The third stage is the detailed design, construction and flight testing of the new aircraft.

Detailed design should be combined with the results of verification tests of certain critical structural elements and systems and based on them. The practice of domestic and foreign helicopter construction confirms the need to create a variety of stands, which carry out tests of various types.

In the design process the designer, on the one hand, takes as a basis certain scientific and experimental groundwork done before starting the design, and, on the other hand, proposes new solution aims as soon as possible to obtain experimental confirmation of the correctness of those decisions. In the latter circumstance, as a rule, the first made samples and units are sent to the bench tests to check their performance and finding the weak points that need to be strengthened in the elements, intended for the first flying prototype of the helicopter.

In recent years, it is evident the necessity of creation of a sample of the whole helicopter to conduct fatigue tests, to provide the necessary lifetime of the airframe in a whole, not only of the units exposed to the highest dynamic loads.

On the basis of the results of extensive testing the final drawings are executed, which are used to build the first prototype of the helicopter for the flight test.

During flight tests further details are clarified that could not be foreseen in advance. It is necessary to clarify the loads on the individual units, to look for the solution of new problems related to aerodynamics, stability, operating conditions, and so on. All these circumstances again led to the need to clarify the drawings and the main data of the aircraft.

In carrying out the work at this stage, designers can no longer change the basic parameters of the developed helicopter. The main content of the works of this period is to ensure a planned key performance characteristics and, above all, the weights. During this period, a special system of weight control is developed, taking into account changes in weight, and so on so that the most important parameter of the aircraft would not come out of the previously specified limits.

During the third phase of works fulfillment of specified tactical and technical requirements or internal management decision of design office should ensure quantities characterizing the technological and operational effectiveness of the future helicopter.

And finally, because only at the end of the third phase of the aircraft development its technical characteristics are finally clarified, it is appropriate at this stage to carry out a number of additional calculations to refine the values of operating cost obtained previously in order to get the final result of all the previous work.

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Chapter 4. HELICOPTER AERODYNAMIC CONFIGURATIONS AND THEIR ANALYSIS

Let's analyze basic aerodynamic configurations of rotorcraft performing vertical take-off and landing by means of carrying rotors, such as: helicopters, rotary-wing aircraft, and convertiplanes. Many helicopters are equipped with mechanical drives of the main rotors (MR). In this case while creating thrust it is under action of reactive torque equal to rotor torque and tending to turn fuselage opposite to rotation. Compensation of reactive torque in a single-rotor helicopter is provided by using different steering gears, in double-rotor and multiple-rotor helicopter – by balancing counter-rotating rotors or due to tilt of their axes of rotation. For helicopter with jet-driven rotor a reactive torque is zero, because the blades drag forces are balanced by thrust of the jet engines or other devices mounted on the blade tips.

Selection of a configuration depends on the helicopter purpose and its flight performance and operational characteristics, level of technological evolution during the helicopter development, as well as on the design technical level of the design bureau as per helicopter particular configuration. Often it turns out that helicopters of various configurations developed by various companies under the same terms of reference have similar performance characteristics, and each configuration has advantages under certain flight conditions. It should be also borne in mind that in most cases the helicopter is a multi-mode and multi-purpose aircraft, so the choice of its configuration is a complex multi-criterion problem.

4.1. SINGLE-ROTOR CONFIGURATION WITH STEERING ROTOR

The greatest amount of helicopters in the world is built under this configuration due to its compactness, transmission simplicity and other advantages.

This classical configuration with tail rotor has been proposed by B. N. Yuriev in 1911. In this configuration (Table 4.1) reactive torque is balanced by the torque generated by the tail rotor (TR) mounted on the tail boom. The tail rotor drive consumes 8 ... 15% of power consumed for MR rotation.

The following main types of steering devices can be distinguished:

using thrust of steering rotor mounted in the longitudinal plane, or of propellers mounted on the wings;

creating an aerodynamic force in an inductive flow thrown by the main rotor;

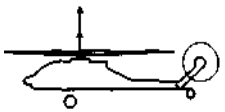
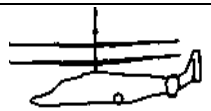
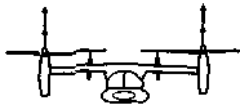
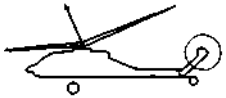
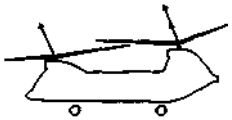
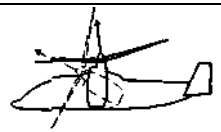
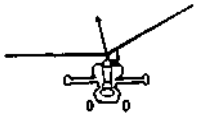
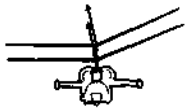
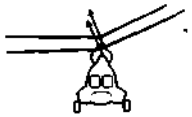
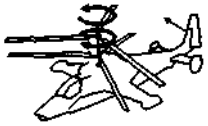
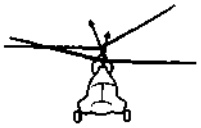
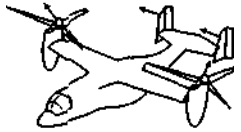
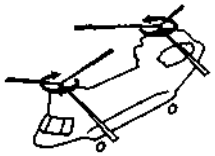
with baffles, deflecting flow from the propeller mounted in the tail section of the fuselage;

creating a reaction force of jet blown sideways through the nozzles at the end of the tail boom.

The *tail rotor* usually is made as pushing relative to the fin to reduce thrust losses for flow with engine operating, when power is applied to MR shaft. In autorotation when MR develops small power TR thrust changes sign. To reduce the length of the helicopter tail boom and overall size the load on the TR disc area is generally higher almost twice compared to the main rotor.

Table 4.1

Principles of Control and Balance for Helicopters and Convertiplanes

Type of control and balance	HELICOPTER CONFIGURATION				Convertiplane with tilt rotors
	Single-rotor with tail rotor	Double-rotor coaxial	Double-rotor tandem	Double-rotor side-by-side	
Altitude control					
Longitudinal control					
Lateral control					
Directional control					
Principal of balancing reactive torque					

Tail rotor also provides directional control of a helicopter and directional stability. In level flight the reactive torque can be balanced by an asymmetrical fin having specific angle of setting and end plates put at the ends of the stabilizer to be installed at the specific setting angle relative to the longitudinal axis of the helicopter. Fin and end plates provide helicopter directional stability in horizontal flight. Installation of fin and end plates will offload the tail rotor, increase its service life and reduce fuel consumption in flight due to decrease of drag.

Rotating TR blades closely spaced from the ground surface represent an increased danger to personnel and can cause accidents and damage of blades when flying close to the obstacles and operating in areas of limited size.

This disadvantage is absent in *fenestron* steering device consisting of a multi-blade rotor located in a shaped annular channel in the fin (Fig. 4.1). The fin protects rotor blades against damage. Lateral force counterbalancing the reaction torque generated by thrusts of rotor and ring due to rarefaction that occurs on the surface, as well as by the asymmetrical lateral force of the fin appearing in horizontal flight.

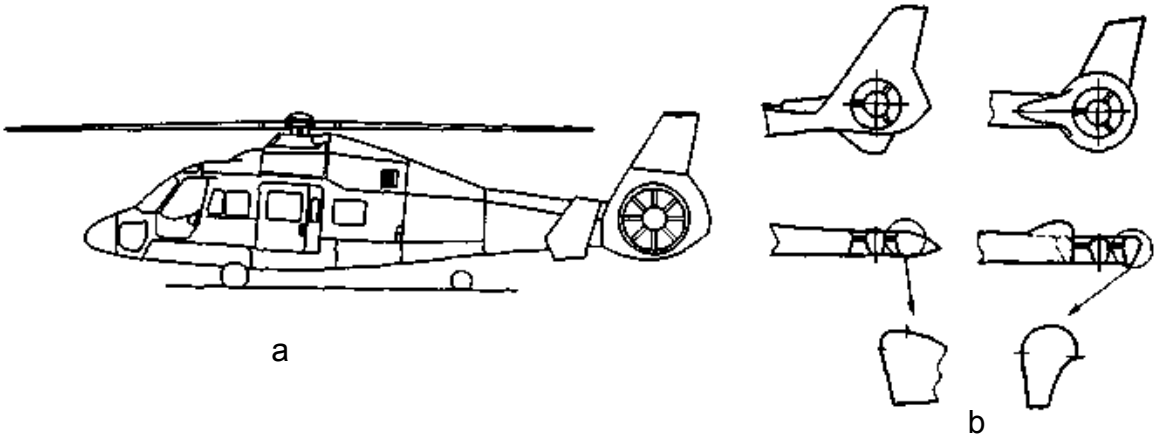


Fig. 4.1. Ka-60 Helicopter and Different Shapes of Fenestron Inlet Collector

Fenestron diameter is smaller than diameter of the traditional tail rotor, thereby reducing gear ratio of the tail reducer, and, consequently, the transmission weight. This eliminates the need for an intermediate angular gear unit, which also reduces the weight.

In hovering with decreasing diameter of the rotor the fenestron drive requires more power than conventional tail rotor. In horizontal flight power consumption may be a few percent smaller or the same as that of the tail rotor. Rather large area of fenestron fin allows to execute horizontal flight and landing of the helicopter with a small slip angle in case of failure of the fenestron rotor or tail transmission.

For the first time fenestron was applied at the SA-341 "Gazelle" helicopter of the Aerospatiale company (France), and then at the helicopters SA-360 and SA-365. Currently fenestron is installed at the helicopters of the Eurocopter company, the

helicopter RAH-66 "Comanche" made as co-production of "Sikorsky" and "Boeing" companies as well as on the Ka-60 Russian helicopter of the "Kamov" company (see Fig. 4.1, a).

Fenestron rotor have greater number of blades K_{bl} = 9 ... 13 than tail rotors, which reduces vibrations transmitted to the tail boom and increases "vitality" of the rotor when the blade is damaged.

Fenestron aerodynamic configuration is selected so that the fin upper surface made asymmetrical and set at an angle creates (in level flight) the lateral force necessary for full compensation of the reactive torque at cruising flight mode. In this case the fenestron virtually creates thrust. The lower part of the fin acts as a safety device when hitting the ground.

The shape of the air intake collector can be made flush with the fin surface (see Fig. 4.1, b) in order to reduce its drag in horizontal flight, or to act beyond the fin surface to implement more traction force developed by the annular duct in hovering. In the latter case, in horizontal flight the fenestron drag will be greater because of the stall in the tail part. In some helicopters (Bell-400) fenestron is a toroidal protector of the tail rotor with the fin: the unit partially retains fenestron functions, as well as ease and benefits of the tail rotor, which has lower power consumption in hover because of the larger diameter.

Fenestron is mounted on the helicopters having take-off weight up to 6 tons. For helicopters with greater takeoff weight fenestron application becomes impractical from an energy point of view.

Steering device of NOTAR type (No Tail Rotor) provides balancing of the reactive torque due to lateral aerodynamic force arising on the tail boom, and the reactive force of air blown through the side nozzles located at the end of the tail boom (Figure 4.2.).

In such a system, the air discharged into the tail boom by a fan is blown through one or two slotted nozzles disposed along the tail boom, and through the side nozzle at the end of the tail boom. At the tail boom blown with flow from MR the aerodynamic force Y_{tb} originates, which together with the reaction force of the side nozzle R_c balances the reaction torque on the MR shaft. Directional control is performed by varying the air flow through the side nozzles. Track stability is ensured by the fin surfaces.

The NOTAR system advantages are: lack of masses spaced apart at the end of the tail boom, decreasing the moment of inertia with respect to the rotation axis and MR transverse axis, simplifying the structure, increasing the load ratio, decreasing the drag in flight, as well as improving safety.

However, such a system requires higher power consumption for balancing the reactive torque, especially in hovering (17%). First NOTAR system was tested by Hughes company on Hughes-500 helicopter and subsequently used on MD-520N

helicopter of the McDonnell Douglas Company, commercially available since 1991, as well as helicopters MD-600N and MD-900.

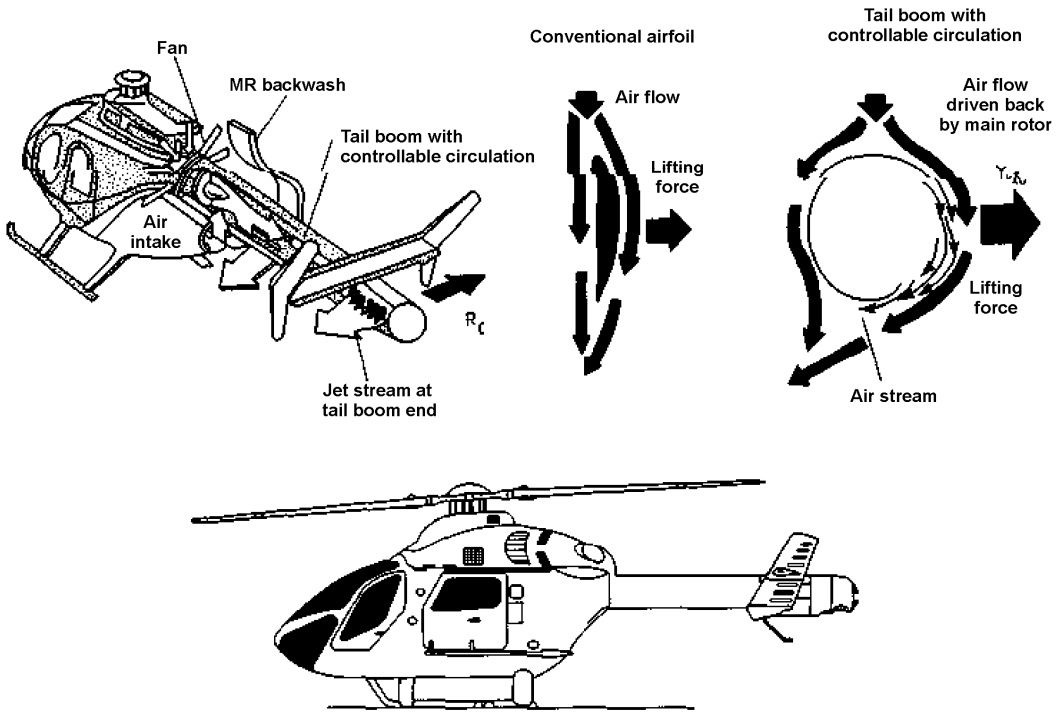


Fig. 4.2. Principal Diagram of NOTAR Steering Device Operation and McDonnell Douglas MD 900 Helicopter with NOTAR System

Balancing reactive moment only by the reactive force of the blown air requires large power consumption (up to 20 ... 35% of the demand for MR operation).

Single-rotor helicopter transmission design is relatively simple and includes an intermediate reducer, shaft that transmits power to the main gearbox, main gearbox and tail shaft that transmits power to the steering unit of various types. In today's helicopter turboshaft engines are commonly used, and piston engines are installed on light helicopters. Gear ratio of main reducer increases with the helicopter weight and its MR radius.

In most cases main reducers of medium and heavy helicopters have planetary stages. Tail shaft rotational speed is selected as high as possible to reduce its weight.

Single-rotor helicopter control, with the exception for the track control, is provided by main rotor swash plate assembly, that allows you to change the blade setting angle of all blades at the same time – so called collective pitch control (COM), or to change the angle cyclically by the first harmonic – so called cyclic pitch control (CYC). Using cyclic pitch control allows to tilt the cone described by MR blades and MR thrust in the necessary direction, and this creates a control moment relative to the center of weight of

the helicopter. In addition, a moment of the same sign is created proportional to spacing of the drag hinge axes on the sleeve.

Altitude control (along vertical) is executed by using COM (see Table 4.1) by the collective pitch lever simultaneously allowing change of engine power. Longitudinal control (by angle of pitch) is performed by tilting thrust in the longitudinal plane, and lateral control (by the roll angle) is executed in transverse plane with the lever associated with cyclic pitch. Directional control is provided by varying TR thrust or thrust of other steering unit using the pedals.

Helicopter control features with presence of cross-linking when the impact on one control requires simultaneous impact on others. For example, when changing MR thrust by means of collective pitch lever it simultaneously changes the reactive torque. In order to balance it you need to move pedals to change the tail rotor collective pitch and its thrust, which, in turn, should be balanced by moving the cyclic pitch lever sideways and tilting MR thrust.

Balancing helicopter is carried out by cyclic pitch using swash plate assembly, which angle of tilt has structural limitations. Therefore, the range of helicopter CG position change in single-rotor design is limited. To expand the center of gravity position range on some helicopters TR plane of rotation is inclined relative to the longitudinal axis of the helicopter, so there is an upward component of the tail rotor thrust. Piloting a helicopter in this case is made using an autopilot. Single-rotor helicopter control system is simpler in comparison with other helicopter configurations.

Wings may be mounted on a single-rotor helicopter to reduce forces acting upon MR in horizontal flight, which on combat helicopters are also used to suspend armament. In hover and when flying with low speed the wing streamlined by MR flow causes loss of thrust.

Helicopters of single-rotor design have less aerodynamic drag compared with other helicopter configurations due to possibility to implement a streamlined fuselage, hub low drag, application of cowling for swash plate assembly and MR shaft. MR operates in a free-flow, as it takes place in double-rotor configurations. Thrust losses for fuselage streamlining in hover compared with double-rotor helicopters of tandem and side-by-side configurations are lower as fuselage locates in the flow area at small radii from MR axis where inductive speed of air thrown by main rotor increases with growing radius and is yet small.

Loads caused by the payload located in the fuselage are transferred to MR over the shortest path, which ensures high rigidity and less exposure to various types of self-oscillations. Level of vibration transmitted from MR to fuselage depends on the number of blades and decreases with increase in their number.

Single-rotor configuration is widely used in helicopters of different weight categories: from ultra light to heavy. Currently, the Mi-26 helicopter can carry the heaviest cargo in the world among serial lifting helicopter (Fig. 4.3). It capable of

carrying loads of 20 tones. The Mi-26 holds the world record in lifting a weight of 25,111 kg to a height of 2,000 meters, set in 1982, the World speed record belongs to the single-engine helicopter "Lynx" of the Westland Company (UK) and is of 400.87 km/h. Helicopters of single-rotor design are used for very different purposes, they are used in both civil and military fields.

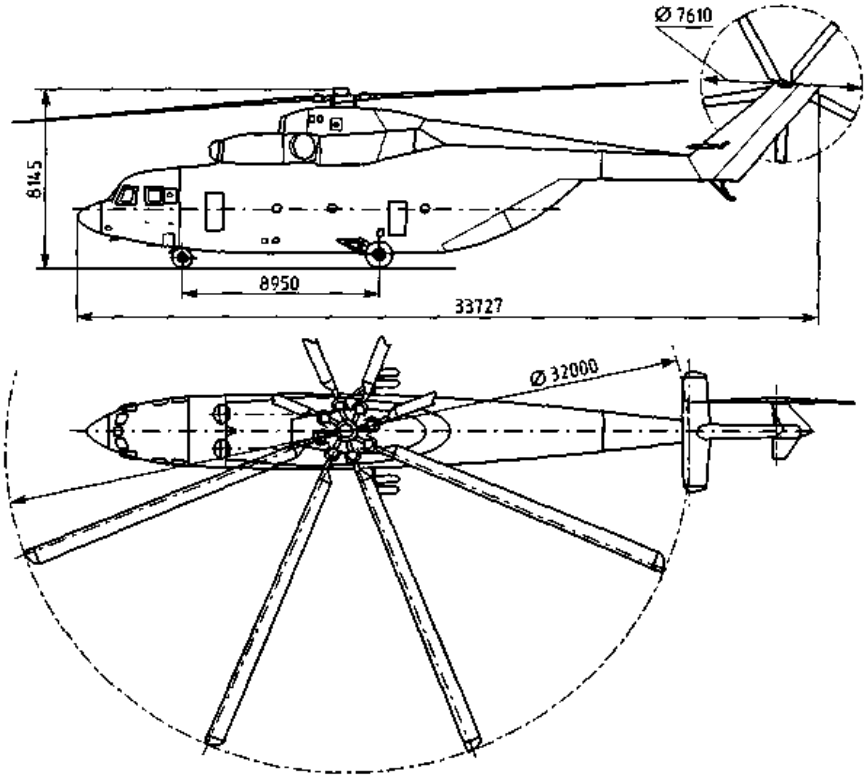


Fig. 4.3. Mi-26 Helicopter

4.2. DOUBLE-ROTOR CONFIGURATIONS

Double-Rotor Coaxial Configuration. In this arrangement thrust is produced by two main rotors rotating in opposite directions, so that reactive torques of both rotors are balanced (see Table 4.1). Therefore, a helicopter of the coaxial configuration, unlike helicopters of a single-rotor design has *no loss of power on steering unit drive*.

Transmission of coaxial helicopter has the smallest number of elements as compared with other configurations and is more compact. Power from one or two engines is transmitted through the intermediate shafts and main gearbox to two synchronously rotating rotors. The upper rotor shaft passes inside the lower rotor shaft. To prevent the blades crossing the top rotor usually is staggered relatively to the lower rotor by the value

$$\Delta y = 0.2R ,$$

where R – MR radius.

Coaxial helicopter control features by smaller moments of inertia about the rotation axis and transverse axis passing through the helicopter center of gravity compared to a single-rotor helicopter due to absence of tail rotor and gearbox located at the end of the tail boom, as well as of other elements. The coaxial arrangement is an aerodynamically symmetrical. Therefore control system contains less number of cross-links than single rotor configuration. Control is provided by a swash plate assembly on each of the rotor, units matching their work, as well as rudders, located on the fin or end plates.

Altitude is controlled by simultaneous change in thrust of the two rotors using a collective pitch (see Table 4.1). The longitudinal and lateral control is also carried out by simultaneous tilt of thrusts of upper and lower rotors using cyclic pitch respectively in the longitudinal and transverse plane. The effectiveness of the longitudinal and lateral control compared to the single-rotor arrangement is higher due to stagger of the upper rotor.

Directional control is provided by differential pitch, that is on one rotor collective pitch is decreasing, and on the other it is increasing. With that total thrust of rotors remains unchanged, and there is a difference reactive torque under the influence of which a helicopter turns in the right direction. Simultaneously rudders located on the fin or end plates are deflected and create moments of the same direction as the rotor in the case of power transmission from the engine to rotors. In autorotation, when the engines do not work, the rotors are rotated by the energy of the incoming flow and low power is taken from them. In this case, the control moment on the rotors changes the sign and directional control is provided only by deflection of rudders.

Spacing of rotors heightwise in hover improves the coaxial rotor characteristics compared with the single rotor having double solidity ratio (i.e. having the same number of blades as the coaxial aircraft), as large weight of air is engaged in creation of a coaxial rotor thrust. This is due to the fact that the air stream from the top rotor, reaching the low rotor plane has a preload, and the air is sucked further by the lower rotor through the side surface.

This effect reduces the inductive component of the power expended on rotation of coaxial rotor. This property combined with the absence of power consumption for balancing reactive torque enables coaxial helicopter compared with single-rotor one to have a greater excess power in hover or vertical flight and, consequently, a larger static ceiling, and hence the vertical rate of climb.

In combat coaxial helicopters as well as on single-engine helicopters small wings are set that reduce forces acting upon MR in flight and are used to suspend arms.

The helicopter of the coaxial configuration has reduced level of vibration transmitted from the propellers to the fuselage, as the amplitude of the migratory harmonics is inversely proportional to the number of blades, and at most number of coaxial helicopter blades is six.

Coaxial helicopter, as well as single-rotor, is compact, has rigid design, less probability of occurrence of resonant vibrations, since the loads of the masses located in the fuselage are transferred to MR over short path.

In horizontal flight rotor stagger has the effect of biplane box wings when the negative inductive effect of the upper rotor upon the lower one and vise versa, of the lower rotor upon the upper one is reduced, resulting in lower power losses. Coaxial helicopter fuselage provides low drag, which contributes to the lack of a tail rotor. However, control elements of the system located in the flow (which are often cylindrical bodies) and the upper shaft of the rotor have large drag. Nosing of these elements in order to reduce drag is difficult.

The coaxial configuration compared to others has small dimensions of the fuselage due to the relatively short tail boom, on which the vertical tail with rudders and horizontal tail in the form of stabilizer non-deflected in flight are located.

Small size of coaxial helicopter led to its use as ship-based, requiring small area on the deck and in the hold to land and accommodation. If blades are foldable they are arranged along the tail boom, which has no need to be folded as on single-engine helicopters.

Coaxial helicopters, which are produced in series only by the "Kamov" Company have both civilian and military applications. Notable benefits of coaxial helicopter are seen when it works as a crane. In Russia single-seat coaxial combat helicopter Ka-50 (Fig. 4.4), having takeoff weight of 10.8 tons and maximum speed V_{max} = 350 km/h is used for military purposes. Modifications of this helicopter were developed.

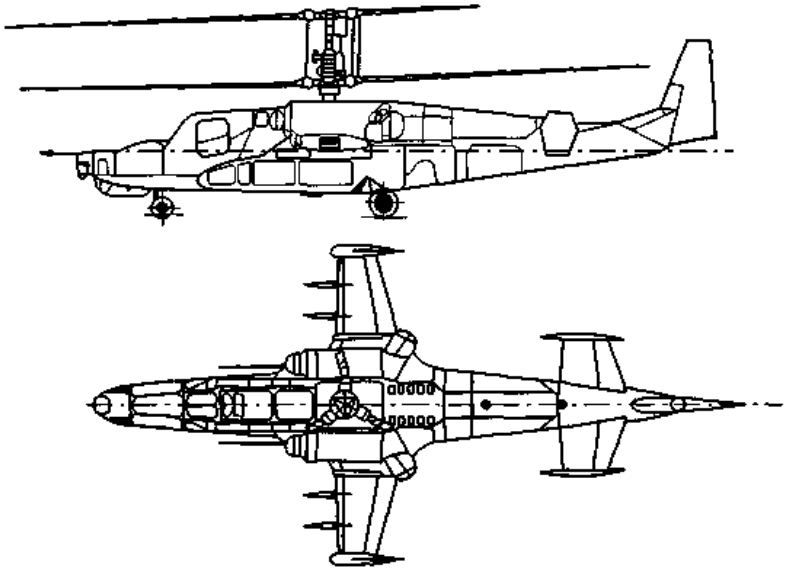


Fig. 4.4. Helicopter Ka-50

Remotely piloted helicopters are built coaxial thanks to compactness and other benefits of this configuration.

The firm "Sikorsky" (USA) has built and tested an experimental coaxial helicopter S-69 according to ABC (advancing blade) concept. Only advancing blades create thrust, i.e., blades operating in zone of speeds adding. The blades are rigidly attached to the hub. Roll moments arising in flight on each rotor are balanced. Rotors are not fitted with drag and flapping hinges. The virtual absence of flapping removes restrictions on the speed of flight caused by stall on hinged retreating blades. The helicopter S-69 with additional thrusters develops speed V_{max} = 450 km/h.

Double-rotor tandem configuration. In this configuration the reactive torques are balanced on two MR rotating in opposite directions and located on the longitudinal axis of the fuselage, one after another (see Table 4.1).

In tandem configuration rotors may rotate in one direction and the reactive moments are balanced by tilting axes of rotation of rotors and arising thrust with the projection on the plane of the rotors rotation.

Rotors are connected by means of a synchronizing shaft, which has different loading depending on the engine layout. In case of "modular" rotor installation when each module consists of MR, gearbox and engine the synchronizing shaft transmits power difference given by the rotors. When placing the engines in the rear fuselage the synchronizing shaft permanently transmits power to the front rotor.

Helicopter control in altitude is provided by means of simultaneous change in thrust of two rotors.

The longitudinal control can be provided similar to single-rotor configuration with the help of cyclic pitch by tilting thrusts of rotors or differential change in collective pitch of the front and rear rotors, which is more efficient, because the rotor stagger in horizontal plane provides large control torque. When providing longitudinal control with differential collective pitch this expands the range of variation of the longitudinal CG (center of gravity position of the helicopter).

Lateral control is performed using cyclic pitch of two rotors in simultaneous tilting thrusts and also track control uses cyclic pitch, but differential, when the rotor thrusts are tilted in different directions, and the horizontal projections of the thrusts on the shoulder, equal to the distance between the rotors create a control moment.

Tandem configuration is aerodynamically asymmetrical. In level flight the vortex wake from the rotor passes close to the rear rotor plane and blades of the latter operate in the field of the additional inductive velocity directed downwards, which increases their drag and hence power consumption of the rear rotor.

To reduce the adverse impact of the front main rotor upon the rear rotor the latter is staggered with regard to the front one by $y \approx 0.2R$. In addition, to make equal torques in horizontal flight the rotor axes are specially tilted: the axis of rotation of the front rotor is tilted forward and axis of the back rotor is tilted backward.

In sliding conditions the helicopter vortex trail of the front MR extends from the rear MR plane aside and negative inductive effect of the front MR upon the rear rotor reduces and consequently rear rotor torque reduces and difference reactive torque appears that returns the helicopter to its original position or turning it to greater slip angle, i.e., causes directional instability.

Essential interference of tandem helicopter rotors, presence of cross-links in the control system requires the use of an automatic control system that is implemented on modern helicopters.

To reduce length of the fuselage and increase helicopter weight efficiency the rotors have an overlap, i.e. rotor disc planes overlap (see Fig. 4.5). The relative degree of overlapping is selected so that no collision of blades rotating in opposite directions has occurred. With an increasing number of blades overlapping degree decreases. In hover the rotors overlap leads to thrust loss, which for a three-blade rotor is about 4%. Loss of thrust for streamlining fuselage of tandem helicopter is slightly larger than for the single-rotor and coaxial helicopters, since the fuselage is in the zone of high inductive velocities.

Tandem configuration helicopter transports cargo located in the middle of the elastic fuselage, and loads from it are transmitted to the rotors over the longer path than that of the single-rotor and coaxial helicopters. Therefore, tandem helicopter is more susceptible to resonant vibrations, especially as the level of vibration transmitted to the fuselage from the rotor with fewer number of blades, is higher than that of single-rotor and coaxial helicopters. To prevent vibrations on the tandem configuration helicopter special dampers are provided as well as damping suspension of pilot seats and other provisions.

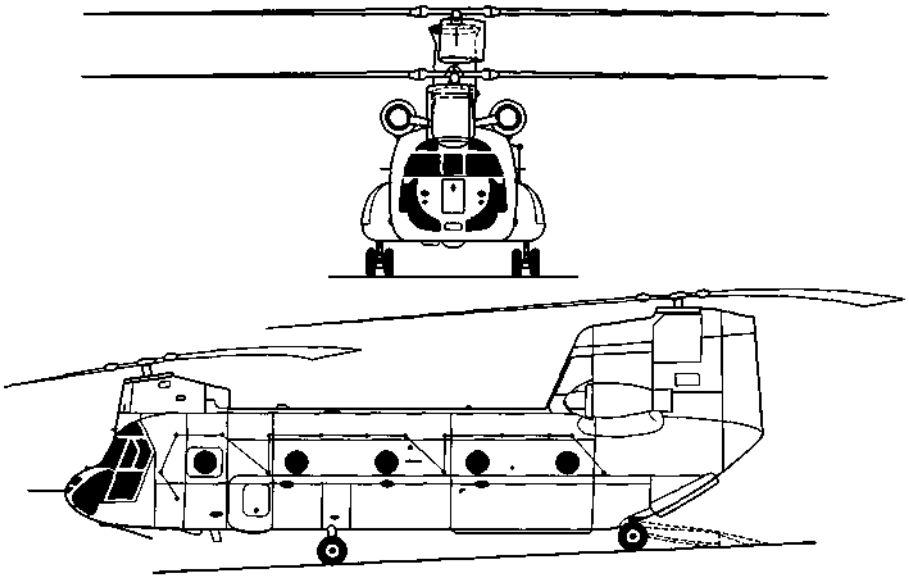


Fig. 4.5. Helicopter CH-47 "Chinook"

Tandem configuration helicopter is mainly used as a transport one because of its voluminous fuselage allowing to place various bulky loads. Boeing Company produced helicopter CH-47 “Chinook” (Fig. 4.5) of various modifications since 1962 with take-off weight of 15 ... 22.5 t, capable of carrying loads in the cabin and on an external sling. Helicopter “Chinook” is one of the main transport helicopter of the US army, Canada army and armies of other countries NATO-members.

Double-rotor side-by-side configuration. Side-by-side helicopter configuration, which includes main rotors arranged on transverse axis relative to the fuselage, is similar to aircraft in arrangement. It has no loss of power for compensation of reactive torques as rotors rotating in opposite directions, are automatically balanced (see Table 4.1). The configuration features with aerodynamic symmetry and positive interference of rotors. According to this configuration helicopters were produced in Germany and in Soviet Union (Bratukhin design bureau, Mil design bureau) were built.

Helicopter transmission consists of two main gears and shaft, synchronizing rotation of main rotors.

Altitude is controlled by changing collective pitch of two rotors simultaneously (see Table 4.1). The longitudinal control is provided in the same way as in the single-engine helicopters, using cyclic pitch by tilting MR thrusts. Lateral control is provided by differential change in the rotors collective pitch, when one MR thrust decreases and the other MR thrust increases by the same amount and pair of forces occurring on the arm, equal to the distance between the rotors, create controlling roll moment. Such roll control method is more efficient than using cyclic pitch of two rotors when their thrusts are tilted.

Directional control is provided by tilting one MR thrust forward, and the other - backward using differential cyclic pitch and deflecting rudders as on an airplane (see Table 4.1.), which effectiveness is determined by the horizontal flight speed.

In level flight each rotor creates additional upflow in the neighboring rotor plane, resulting in a reduction of power consumed by the rotors. Rotor rotation direction has practically no influence on this effect, but it has a significant impact on the control effectiveness, which is higher in the type of "breaststroke" direction of rotation.

Overlapping rotors which, as in tandem helicopter, reduce the distance between the rotors and weight of the helicopter structure that also leads to slight increase in power consumed for rotation of rotors, as induced velocity increases in the overlap area. Side-by-side helicopters with the wing, which is used for attachment of the rotors have the greatest losses in hover for flow about compared with other configurations as streamlined wing is flown with stall and has high drag.

To reduce losses for flow about on some helicopters (Mi-12) wing had the inverse taper, i.e., at the point where the inductive speed increasing over radius reaches maximum the wing chord and consequently streamlined area decrease (Fig. 4.6).

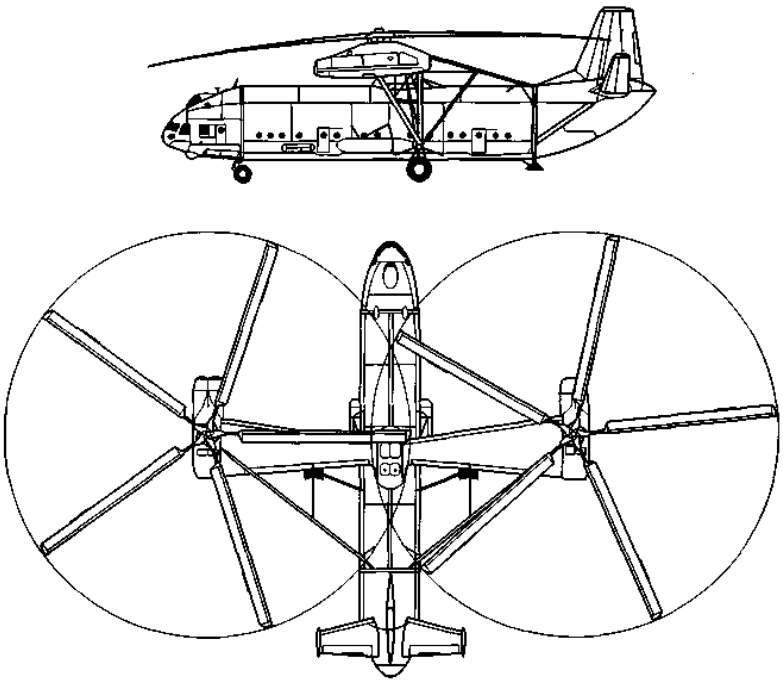


Fig. 4.6. Mi-12 Helicopter

The use of truss structures for fastening rotors instead of wings also reduces losses for streamline, but in horizontal flight, trusses produce great drag. Wing, as a rule, is also supported by struts, increasing drag, so the side-by-side helicopter compared to other has higher drag. In cruising flight wing creating lift offloads rotors.

The design of the side-by-side helicopter compared with single-rotor and coaxial helicopters has less rigidity as the loads caused by cargos placed in the fuselage are transferred to MR along lengthen path including fuselage, truss (or wing with braces). Rotors are located at the ends of the enough elastic long truss or beam structures undergoing flexural deformation in two directions (horizontal and vertical), as well as torsion. Therefore, side-by-side helicopter is subject to resonant vibrations, that requires to provide necessary hardness of MR mount, use of special dampers.

The world's first serial helicopter of side-by-side configuration Fa-223 (Germany) was produced in 1941. At the same time Bratukhin design bureau (USSR) built helicopter "Omega", which was used as a basis to create a number of modifications "Omega-2", G-3, G-4, which implement a modular design principle. Each module consistes of MR, gearbox and power plant. Helicopter T-3 was the first production helicopter in the USSR. In 1967, the Mil design bureau built the most lifting helicopter in the world Mi-12 (Fig. 4.6) with carrying capacity up to 40 tons and takeoff weight of 97 tons. It has modular design like "Omega" consisting of rotor assemblies of the Mi-6 helicopter.

Double-rotor configuration with crossing rotors. Such a configuration also referred to as intermeshing rotors is the ultimate variant of the side-by-side configuration

when the distance between the rotors coming close to the minimum possible distance. The axes of the two counter-rotating MRs have a camber angle of about 24°. Reactive MR moment is balanced, but due to tilt of MR axes unbalanced pitching moment arises, which is balanced by longitudinal tilt of MR thrusts via cyclic pitch and by stabilizer. In order to avoid crossing the number of blades is usually not more than two. This configuration, as well as coaxial, is compact, features with small size of the fuselage, simplicity and small number of transmission elements, which has no long shafts and engine torque is transmitted to two MR shafts. To prevent crossing of blades in some configurations additionally a synchronizing shaft is used directly between the shafts of rotors.

Altitude is controlled by changing the collective pitch of two rotors. The longitudinal and lateral control is provided by simultaneous tilt of two MR thrusts via cyclic pitch; directional control as in the coaxial configuration is provided by differential change of collective pitch of the rotors and rudder deflection.

Despite the existence of cross-linking, the helicopters have good controllability. At large angles of slip and yaw due to rotors interference they become unbalanced when reactive moments are not balanced, resulting in lack of directional stability. To eliminate this phenomenon choose a sufficiently large area of the vertical tail, as well as introduce automatic stabilization system.

Compactness, absence of long tail boom and small fuselage size provide high weight efficiency of the helicopter with crossed rotors. The slope of the rotors results in a smaller distance between the blades and the ground surface, which represents an essential danger to the crew and maintenance personnel. For safety reasons the rotors are installed on pylons that lead to an increase in drag and reduction of weight efficiency, as well as in possible helicopter turn-over when landing on a sloping surface.

The first production helicopters with crossed rotor of A. Flettner design were produced in Germany and was used as a deck-based, coastal anti-submarine, rescue and reconnaissance helicopters until 1945. Experimental helicopters were built in the United States, France and Czechoslovakia.

The greatest success in the production of intermeshing rotors was achieved by the Kaman company, created in 1953 on the base of Kellet company, which has developed several experimental helicopters. The company has produced a series of Kaman K-600 helicopters (HH-43) and their modifications. Distinctive features of the helicopter are compactness, small size, lack of power consumption for torque balancing, high weight efficiency, led to their use as naval helicopters, and helicopter-cranes when flying at a relatively low speed.

Design features, laid in a limited (two maximum) number of blades in order to avoid their crossing, and tilted axes of rotors, whereby the blades are close to the ground, limit the increase in weight of such helicopter up to 5 tons. From 1991 the

Kaman Company constructs the K-MAX helicopter having take-off weight of 5215 kg (Fig. 4.7). The helicopter is designed to carry cargo on external sling and to perform erection works.

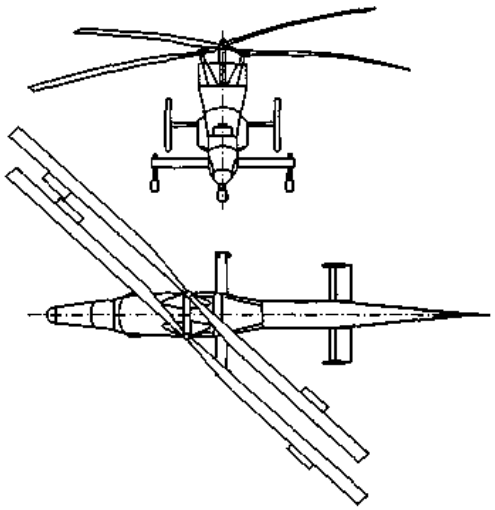


Fig. 4.7. K-MAX Double-rotor Helicopter with Crossed Rotors (Intermeshing Rotors) of Kaman Company (United States)

4.3. MULTIPLE-ROTOR CONFIGURATIONS

As with increasing the diameter of the rotor its weight and masses of the main gearbox and other helicopter units rapidly grow, then to increase the helicopter carrying capacity for a given level of technical perfection of its units it is advisable to apply several rotors.

Multiple-rotor helicopter with a number of rotors equal to 3, 4, 6, 8, and 20 were developed and tested at early stage of helicopter development. Designers were attracted by the possibility of using similar design elements to create a heavy-lift helicopter. Reactive moments of the multiple-rotor configurations can be balanced due to different direction of rotation of the rotors or by small tilt of the axes of rotation, which creates horizontal component of thrust providing necessary moment in the plane of rotation. In the latter case, all MRs can be rotated in one direction.

Main rotors of multiple-rotor helicopters are joined by shafts synchronizing their rotation; degree of loading, as in the double-rotor configurations with rotors set apart, depends on the engines location. If the rotor-engine unit is a single unit consisting of MR, gearbox and engine, then synchronizing shafts transmit only difference torques.

Control of multiple-rotor helicopters is enough effective, because the spacing of MRs provides large arms to create control moments. Due to this fact range of permissible center of gravity positions can be expanded. Structure of swash plate assembly may be simplified, because the longitudinal and transverse control is provided

by differential MRs collective pitch and directional control is provided by cyclic pitch of two or more rotors.

In multiple-rotor helicopters MRs experience more adverse inductive interference, which leads to increase in the power required and causes cross-links in control and requires significant control margins. Drag in horizontal flight of multiple-rotor helicopters compared with single-rotor and double-rotor is higher due to additional drag of load-bearing members of truss or beam structure connecting MRs with fuselage or interconnecting them.

Multiple-rotor helicopter structure as well as double-rotor helicopter of tandem and side-by-side configurations is susceptible to resonant vibrations as main rotors, which are the source of vibration, are interconnected by sufficiently extended resilient truss members or other supporting devices. Availability of long synchronizing shafts connecting MRs, structural elements providing mounting of spaced rotors and extended control linkage increase the empty helicopter weight and reduce its weight efficiency.

An example of the multiple-rotor helicopter passed flight test cycle in 1948-1950 is an experimental three-rotor helicopter W-11 "Flying Horse" of "Cierva" company (Fig. 4.8). The helicopter had a takeoff weight of 7940 kg (single Rolls-Royce piston engine, power of 1620 h.p.) and was at that time the largest helicopter in the world. 15 m diameter rotors rotate in the same direction, and their axes have a slight tilt from the vertical to compensate reactive torque. Because of this features change in MRs collective pitch while controlling with respect to altitude, roll and pitch caused jaw imbalance. The test results showed efficiency of such a configuration with all its inherent shortcomings. The helicopter was well-controlled. Work on the W-11 helicopter and its modifications have stopped after the accident, which occurred in 1950.

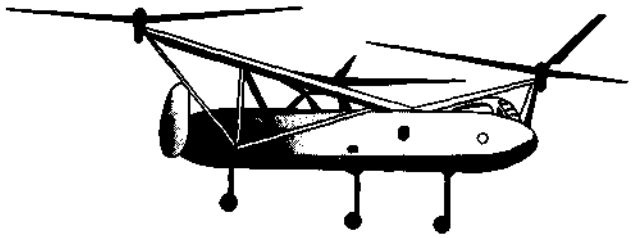


Fig. 4.8. W-11 Tree-Rotor Helicopter of Cierva Company (Great Britain)

The issue of creating three-rotor or four-rotor helicopter is periodically reviewed by the design bureaus, if necessary to carry heavy indivisible loads. Thus the Mil Design Bureau (USSR) in 1977-1982 have developed the Mi-32 super-heavy helicopter-crane of three-rotor configuration (Fig. 4.9) with carrying capacity of 55 tons using carrying systems and power plants of Mi-26 serial helicopter. MRs located at the vertices of an equilateral triangle formed by the circular beams connecting the rotor-engine units. Cargo should be carried on an external sling. The project confirmed possibility of creating a heavy-duty helicopter using multiple well-known engine-rotor units.

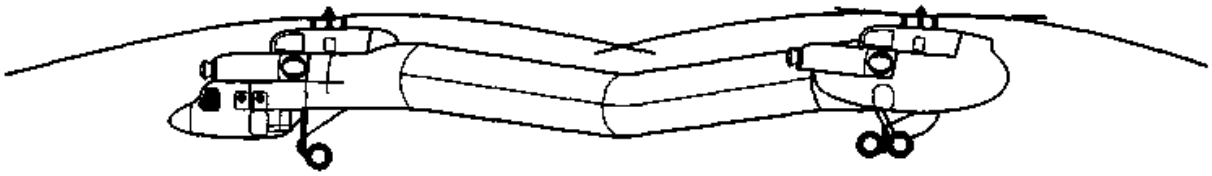


Fig. 4.9. Project Version of Mi-32 Double-Rotor Helicopter-Crane

4.4. HELICOPTERS WITH JET-DRIVEN ROTOR

In this configuration, the moment created by drag force of MR rotating blades is balanced by the moment created by thrust of the jet nozzles or engines mounted at the blade tips. Therefore, unlike of configurations with MR mechanical drive in such a construction there is no reactive torque transmitted to the fuselage. When using a jet drive on a single-rotor helicopter this eliminates the need for devices that consume a significant portion of the power to balance this moment, such as tail rotor, fenestron and so on. These devices are necessary only for directional control and to balance small moment of friction in the bearings, about which rotor revolves.

The positive feature of the helicopter with a jet drive is the absence of the main gearbox and loaded transmission, and consequently, lower empty weight of the helicopter structure and high weight efficiency. There are distinguished two main types of jet drives:

Jet compressor drive, where compressed air from a compressor installed in the fuselage, or hot air from the gas-turbine engine is supplied through the hub and blades to the nozzles located at the blade tips;

With jet engines mounted at the blade tips - ramjet, pulsjet, liquid-propellant jet engine, turbojet.

The compressor drive depending on the air temperature (gas) is divided into three systems:

1) "cold cycle", when air is supplied from the compressor;

2) "hot cycle", when gas having temperature up to 1200°C is supplied from a turbine engine exhaust installed in the fuselage;

3) "warm cycle", when air from the compressor is mixed with gas from turbine engine.

With increasing temperature of air supplied to the nozzles, the compressor drive efficiency increases, but also there are problems with providing strength and insulation of the hub bearing elements and blades. To increase the compressor drive power with "cold cycle" an afterburning system is used to burn an additional fuel supplied to the nozzles.

The main disadvantage of the compressor drive lies in essential "hydraulic" losses in long ducts, design complexity of the hub and blades, deterioration of the rotor characteristics due to the need to accommodate bulk ducts inside the blades.

Afterburning system reburning fuel in the burners working in the field of centrifugal force has low thermodynamic efficiency, low degree of combustion and generates large noise.

As compared with a mechanical drive, ceteris paribus jet drive system fuel consumption is several times higher.

The only helicopter with jet compressor drive operated in "cold cycle", weight-produced (1953), was SO-1221 "Gene" helicopter of the Sud Avias Company (Fig. 4.10).

Configuration of the compressor drive with afterburning has been implemented in a number of helicopters (from light single-seat to heavy ones), as well as in the single-rotor rotary-wing aircraft "Rotodayn" of the Fairey Company in 1957, which had a takeoff weight of 14969 kg.

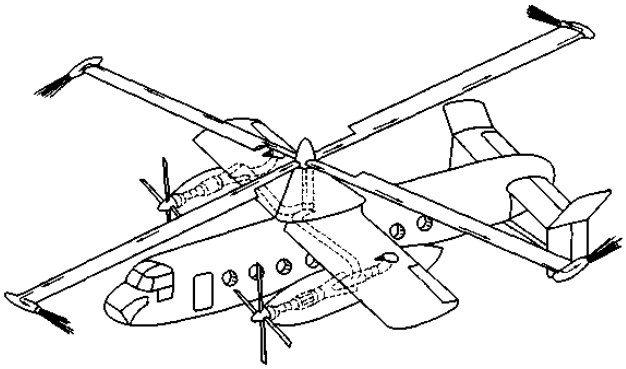


Fig. 4.10. Rotorcraft with Main Rotor Jet Compressor Drive (Great Britain)

Compared to the compressor drive the jet drive with engines on the blade tips is more simple in construction, because it has no engine, located in the fuselage and duct system.

The results of experimental studies on experimental helicopters with engines at blade tips shown that compared with the mechanically driven a fuel specific consumption is higher about 10 times in case of ramjet installation at the blade tips, and in case of pulsejet engines it is higher about 3-5 times. If liquid-propellant jet engines are installed then specific fuel consumption is even higher. The lowest fuel consumption is featured by rotors with turbojets. Rotors with ramjet engines that create thrust in the presence of the incoming flow require pre-rotation by means of a special device.

Thrusters located at the blade tips have large circumferential speed and create considerable drag, which deteriorates the aerodynamic performance of rotors. Big centrifugal forces generated by engines load blades and hub that complicates their design and increases weights.

Rotating parts of turbojets under action of high gyroscopic moments directly proportional to their moment of inertia, rotor angular speeds and engine component rotating speeds. Gyroscopic moments are transferred to blades and cause their twist.

Gyroscopic moments are balanced by means of the engine opposite rotation that is of compressor or turbine or by installation of two turbojets at the blade tip rotating opposite.

Jet-driven helicopters with ramjets, pulsjets, and liquid-propellant jets were constructed as super light (single-seat and two-seater) due to great fuel consumption have short flight range.

An example of two-seater helicopter with ramjets at blade tips is HY-1 made by Hiller Company (Fig. 4.11). Helicopter was not widely distributed due to short flight duration about 20 minutes.

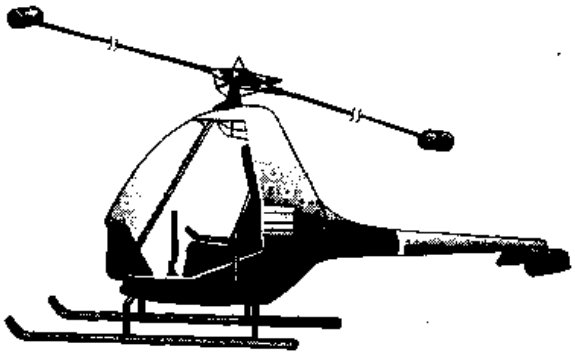


Fig. 4.11. HY-1 Single-rotor Helicopter of Hiller Company (USA) with Jet-Driven Rotor Having Ramjets at Blade Tips

PH-1 single-seat helicopter with liquid-propellant jets at blade tips was built in 1954 by the Rotorcraft Company. Its engine used concentrated mixture of hydrogen peroxide as fuel.

B-7 experimental four-seat helicopter (Fig. 4.12) with turbojets at blade tips was built in 1959 to solve problems of such configuration while developing project of super-heavy helicopter-crane in the Mil design bureau (USSR).

The larger MR diameter the lower its angular speed and gyroscopic moments acting upon turbojet rotating elements and also g-factors caused by centrifugal forces in places of the engines installation. That is why jet drive with turbojets is perspective for development of super-heavy helicopters.

Directional control of jet-driven helicopters is provided by installation of controllable surfaces in the inductive flow, and also steering rotors rotated by compressed air (pneumatic drive), application of jet gas rudders or mechanical transmission.

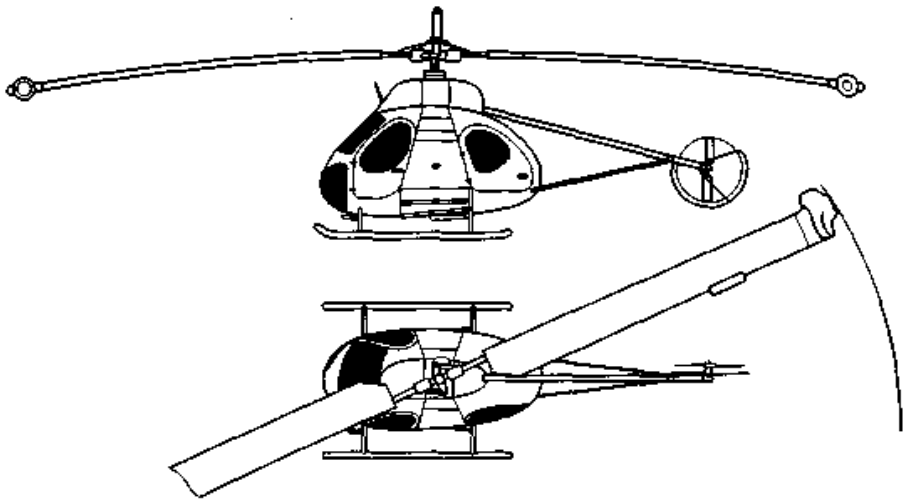


Fig. 4.12. B-7 Single-rotor Jet Driven Helicopter (USSR)
with Turbojets at Blade Tips

4.5. ROTARY-WING AIRCRAFT

Rotary-wing aircraft means combined aircraft, which develop lifting force by rotors (as of the helicopter) during vertical take-off, landing and flight with low speed, and by the wing mainly (as of the airplane) in cruising flight. With that main rotors are partially or fully relieved. To create propulsive force counterbalancing the aircraft drag in cruising flight it is fitted with propellers, turbojets and other propulsion devices.

Creation of rotary-wing aircraft is caused by intention to increase flight speed and transport efficiency compared to conventional helicopter in the way of enhancing the aircraft characteristics, increasing critical speed caused by stall at retreating blade of hinged MR and wave crisis on advancing blade.

Better lift-to-drag ratio in level flight is achieved so that while creating propulsive force by an additional propulsion device tilt angles of MR and fuselage in a whole are reduced that results in lower drag. It also accounts that wing lift-to-drag ratio is higher than of MR.

Rotary-wing aircraft were built according to single-rotor and double-rotor side-by-side configurations. In single-rotor configuration usage of jet-driven rotor has certain advantages related with absence of need to balance reactive torque, lower weight and simpler structure, and also possibility to change MR number of revolutions.

Built experimental rotary-wing aircraft had better speed characteristics in comparison with helicopters but were not produced in series due to structure complexity, lower weight efficiency and high losses of power for vertical flow of the wing, and also less reliability. In fact rotary-wing aircraft has all functional components both of helicopter and airplane. Weight of structure increases (compared to helicopter) due to

additional weight of wing, engines and their transmission systems, and also two control systems.

The first experimental rotary-wing aircraft was two-seater single-rotor aircraft TsAGI 11-EA, constructed in USSR in 1936 (Fig. 4.13). The aircraft was equipped with MR mechanical drive, and reactive torque was balanced by thrust of two steering rotors of variable pitch, placed at the wing tips and playing role of propellers in level flight.

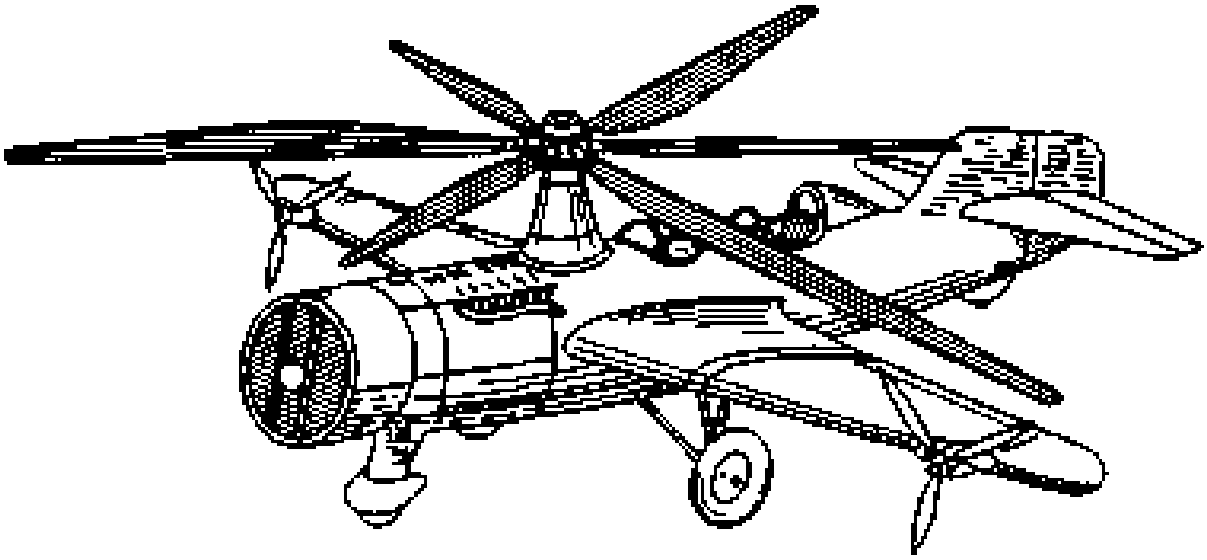


Fig. 4.13. 11-EA Rotary-Wing Aircraft Made by TsAGI

Rotary-Wing Aircraft, which can carry the heaviest cargo, was “Rotodyne” of Firy Company fitted with compressor drive.

Ka-22 double-rotor rotary-wing aircraft was built by side-by-side configuration in Kamov design bureau (USSR) in 1959 (Fig. 4.14). Helicopter take-off weight was about 42500 kg, weight of cargo transported in cabin was about 16500 kg. Rotary-wing aircraft developed maximum speed of 345 km/h, practical flight range was of 450 km. To reduce losses for flow about wing it had flaps deflected by 90°. At high speeds 80...90 % of lifting force was developed by the wing.

The example of high-speed combat rotary-wing helicopter is AH-56A "Shayen" two-seater aircraft built in 1972 by the Lockheed Company (USA). Rotary-wing aircraft was fitted with MR having rigidly attached blades, wing, tail rotor and pusher installed at the tail boom end. Maximum flight speed of the aircraft was of 407 km/h. Weight efficiency of the rotary-wing aircraft (31 %) was essentially lower than of the helicopters existed for that moment of time (about 40 %).

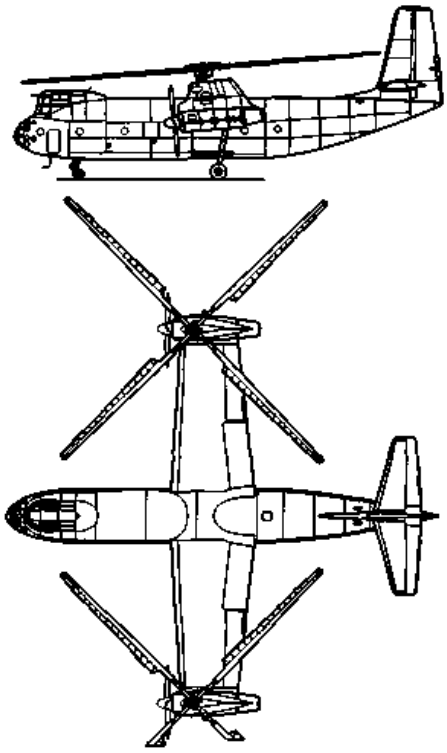


Fig. 4.14. Ka-22 Experimental Rotary-wing Aircraft

4.6. CONVERTIBLE HELICOPTERS

Convertible helicopters (convertiplanes) are rotary-wing aircraft, able to perform vertical take-off and landing like a helicopter, and long high-speed flight as a plane, and acting in these two flight modes using the transformed carrying-propulsive system. By improving lift-to-drag ratio and removal of speed restrictions inherent to helicopter rotor in horizontal flight (stall, wave crisis) convertiplanes have flight velocities close to the velocity of subsonic aircraft, and significantly greater range of flight and transport performance compared to the helicopters. Therefore, use of convertiplanes with costs exceeding those of the helicopters becomes profitable in the performing a range of tasks.

Main configurations of convertiplanes are:

With rotary body which during take-off and landing is arranged vertically, and in level flight – horizontally;

With tilted rotors and fixed wing;

With rotary wings and tilted rotors;

With deflecting flow from rotors;

With rotors stopped in flight that can be transformed into the wing.

On convertiplanes with tilted rotors during take-off and landing rotors hold the upright position and work as a helicopter rotor, and in cruise flight rotors are placed horizontally and work as propellers on the aircraft.

Rotor operating conditions in these modes and their developed thrusts significantly differ, so their characteristics are chosen as a compromise between the characteristics of the carrying propellers. Rotor thrust during take-off is higher than that in horizontal flight approximately by an order of magnitude (by the value of the convertiplane lift-to-drag ration). Therefore in horizontal flight speed is less by 20 ... 25% than in hover.

Configuration with rotary body was applied in US for construction of experimental deck vertically taking off fighter XFV-1 "Pogo" of Convers Company and XFV-1 "Salmon" of Lockheed company. The most suitable (Fig. 4.15) for the operation and passed the full test cycle from takeoff and landing modes to horizontal flight the XFV-1 "Pogo" aircraft was recognized.

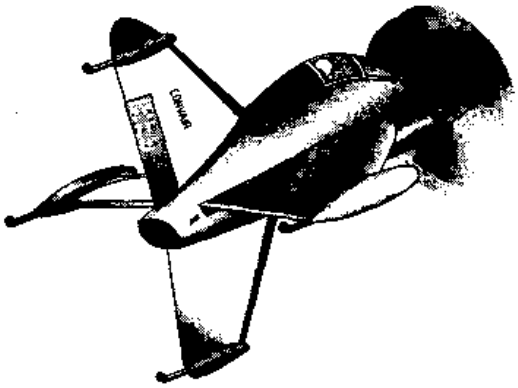


Fig. 4.15. Convertiplane-Fighter with Rotary Body
XFV-1 "Pogo" Conver Company (USA)

The aircraft had the coaxial rotor developing thrust of 90 kN during take-off with take-off weight of 6800 kg. Maximum design speed is 885 km/h. Control was provided by means of elevons and rudders streamlined with flow from rotor and incoming flow. Therefore, the helicopter cross-contours are subject to resonant vibrations, and to prevent resonance they require to provide necessary hardness of MR mount, use of special dampers.

Pilot seat thanks to cardan suspension remain in the same position at different evolutions of the aircraft. Despite this, the piloting of the aircraft, especially in transient conditions and during landing, causes certain difficulties associated with the orientation of the pilot in space, making it difficult for flight operation and was one of the reasons for the cessation of work on the final design of aircraft of this configuration.

Tilt-rotor and fixed wing aircraft were built using two-rotor and four-rotor configurations. They were fitted with carrying tractor rotors of various types:

– With low specific load over the disc area close to the load of helicopter main rotor and having the same structure as that of MR;

– With heavy load close to aircraft propeller load;

– Highly loaded and placed in ring profiled ducts.

The specific load substantially defines power transmitted to the rotor to create a traction force, as well as inductive flow rate ejected by rotor in take-off and landing modes. The more specific load, the higher the flow speed from the rotor and power consumed by it, and consequently, fuel consumption to provide hovering.

In double-rotor convertiplanes with tilt rotors, which are disposed at the wing tips, the rotors have small load on the disc area; they are structurally close to the helicopter main rotor and fitted with swash plate assemblies. During vertical takeoff and landing the aircraft is similar to the double-rotor helicopter of side-by-side configuration (has the same control system), and in horizontal flight it is similar to an aircraft control (see Table 4.1).

To transit from hover to cruise flight the rotors slowly turn forward by 90° for 11 ... 20 seconds. In the event of engine failure, landing can be accomplished in the autorotation (like a helicopter), as well as aircraft with gliding and partial tilt of rotors. To reduce loss of thrust from a vertical wing stream it has deflected flaps, as a result the relative loss of traction can be brought to 6%.

Bell Company made detailed studies of experimental convertiplanes XV-3 (takeoff weight of 2180 kg, maximum speed of 280 km/h) and XV-15 (take-off weight of 5900 kg, maximum speed of 615 km/h). Bell and Boeing Companies in 1982 developed multi-purpose convertiplane V 22 "Osprey" (Fig 4.16.), which first had flown in 1989, and the first production aircraft was put into operation in 1999. The take-off weight of the aircraft with vertical take-off is of 21545 kg, maximum weight of payload transported in the cabin is of 9070 kg, maximum cruising speed is of 638 km/h, range with vertical take-off with payload of 5445 kg was 953 km.

Under the tilt-rotor configuration in annular ducts double-rotor and four-rotor experimental aircraft were built. Setting rotor in a profiled duct can increase the relative efficiency of the rotors in hover, and increases the operating safety of the aircraft.

Disadvantages of aircraft with tilt rotors having greater load on the disc area, are laid in the large power consumed by them in hovering, high flow speed created by rotors and, as a consequence, impossibility of carrying out rescue operations, and making an emergency landing in autorotation in case of engine failure. These shortcomings, together with complex design of the aircraft, low weight efficiency, less reliability than those of helicopters and high cost were the reasons that the convertiplanes did not come out of the stage of experimental research.

These shortcomings are inherent also to convertiplanes with rotary wings. Carrying tractor rotors of these aircraft are similar in features and design to airplane propellers. Under this configuration the experimental apparatus with two or four rotors were designed and built in Canada, United States and other countries. Fig. 4.17 shows prototype of four-rotor aircraft XC-142A made by Hiller, Ling-Temko-Vought and Ryan firms (USA).

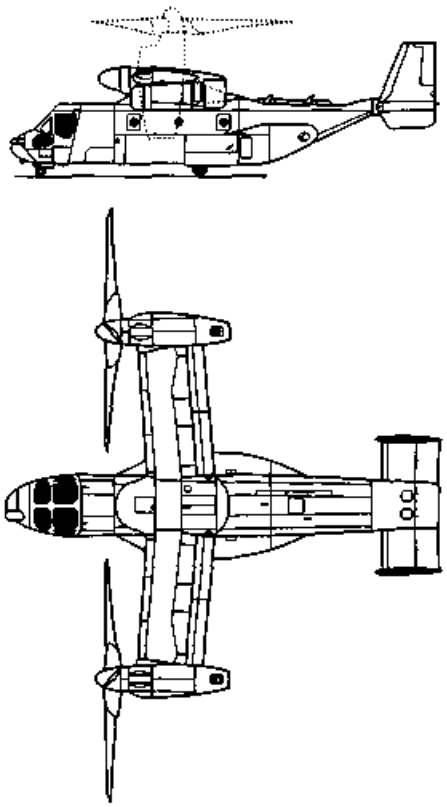


Fig. 4.16. V-22 "Osprey" Convertiplane with Tilt Rotors and Fixed Wing of Bell and Boeing Companies (USA)

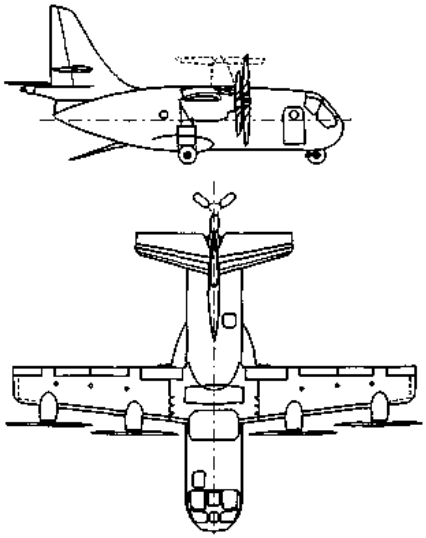


Fig. 4.17. XC-14ZA Convertiplane with Turnable Wings and Rotors

During takeoff and landing jet torques are counterbalanced by different direction of rotation of the left and right rotors. Lateral control is provided by the collective pitch differential control of rotors and longitudinal control by the additional tail rotor mounted in the rear of the aircraft. In the transition to the horizontal flight, when the wing is turned

together with rotors by 90°, at a certain flight speed control is transferred to ordinary aircraft control surfaces.

This configuration has no loss of thrust caused by flow about wing inherent to configurations with tilt rotors and fixed wing. Partial disruption of flow on the wing is possible in transient conditions.

Convertiplane XC-142A having takeoff weight of 17000 kg, maximum payload of 5445 kg at altitude of 6100 m has developed maximum speed equal to 693 km/h.

On aircraft with flow deviation from the rotors the vertical takeoff and landing was carried out thanks to the rotation of the stream ejected by the rotors by 60°. Wherein the aircraft itself must have a pitch angle of about 30°. Studies in experimental convertiplanes showed the fundamental possibility of their creation, but lower efficiency when creating vertical thrust due to the additional losses needed to turn the jet.

The principle of operation of convertiplanes with stopped MR is that in hovering MR creates thrust balancing force of gravity and MR in horizontal flight decelerates rotation and stops becoming a wing, creating necessary lift. One of the most developed project is the Lockheed company aircraft with X-shaped convertible four-blade rotor X-wing (Fig. 4.18). On this aircraft when the rotor stops the blade leading edge becomes trailing edge, i.e., changes the direction of flow on the blade profile (wing). Therefore, in the blade section a symmetric profile is used with boundary layer control system (BLCS) when the air blowing can be made from any slotted nozzles from both the leading and trailing edges, depending on the flow direction.

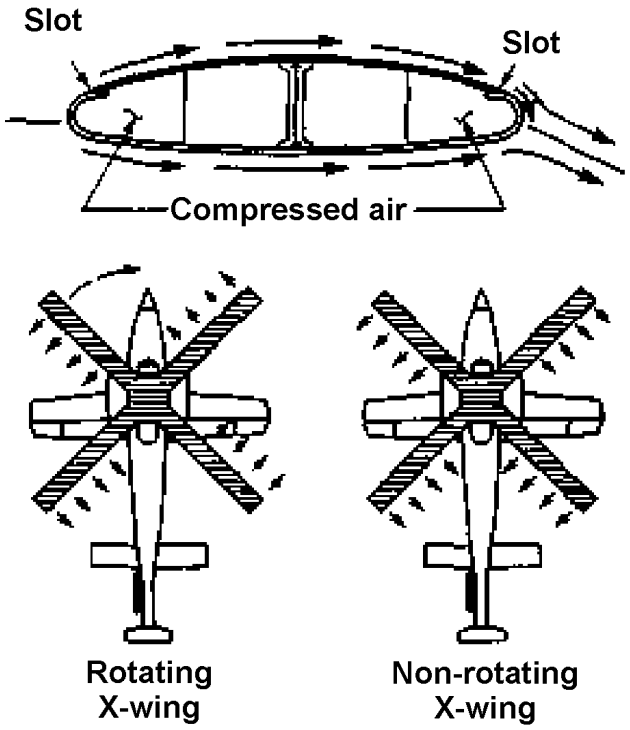


Fig. 4.18. Principal Diagram of Convertiplane with X-wing Stopped in Flight

With the system that provides air flow change in the blade azimuthal position it is possible to provide the control moments required for balancing the unit in transients. The aircraft is equipped with a steering device for directional control in hover and thrusters providing flight with horizontal speed. It is assumed that the aircraft with such a system will consume fuel in hovering in amount, which is several times smaller than convertiplanes with tilt-rotors and high load on disc area. The aircraft having take-off weight of 14 000 kg should reach speeds of up to 1000 km/h.

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Chapter 5. HELICOPTER BASIC PARAMETERS AND RULES OF THEIR SELECTION

5.1. BASIC DATA AND LIMITATIONS FOR HELICOPTER DESIGNING PROCESS

The main objective of the helicopter engineering in addition to selection of a configuration is to determine the most advantageous combination of key parameters to ensure fulfillment of the design specifications given in the Terms of reference, among which an essential part is taken by operating conditions and flight characteristics of a future aircraft. Basic data (besides the performance requirements) should also include the ideas and solutions of the main designer, achieved level of helicopter and its units technical perfection and the following limitations:

Common tactical and technical requirements for civil aviation helicopters, Rotary Wing Transport Category Craft Airworthiness Requirements AP-29;

Operating conditions and characteristics of application;

Possibility of application of up-to-date construction materials, advanced production processes, new designs, availability and technical equipment level, etc.

Among performance characteristics specified in performance requirements there are the following: helicopter load capacity, range of flight, and maximum cruising speed, static and dynamic ceilings, climb rate, etc.

With that helicopter flight performance data are usually interrelated and depend mainly on its take-off weight and power available. An example of this relationship is a diagram of the helicopter transport capabilities showing the dependence between transported cargo and flight range for the different helicopter weight.

Helicopter take-off weight at the same time is defined by the required static ceiling. A helicopter having maximum takeoff weight has not static ceiling and takes off with a run, a helicopter having normal take-off weight has a predetermined static ceiling, and having lower take-off weight has increased static ceiling. With any take-off weight the helicopter carries maximum load, defined by the cargo compartment volume or the strength of fuselage load-bearing members over certain (estimated) distance. For a given takeoff weight a flight for distance of longer range is performed with reduced load due to corresponding increase in fuel reserve.

As example of the existing design restrictions the requirements for flight data and flight characteristics of helicopters of different categories, contained in the Aviation Regulations AP-29 may be given.

Restrictions defined by operating conditions and application features are reduced primarily to the altitude-and-climatic conditions of flight provisions and limits established by the flight manual (AFM) of rotary-wing aircraft while performing different missions.

Helicopter performance characteristics in the Terms of reference are given under International Standard Atmosphere (ISA). The conditions given in the Terms of reference, possible deviations from the standard atmospheric parameters (pressure, density, temperature, humidity) and the condition of flight operations, including characteristics of base airfields (barometric altitude, strength of soil, slope) and flight parameters (altitude range, operating speed, helicopter category, flight safety conditions in case of one engine failure) are set.

Climatic conditions of helicopter operation are the most diverse - from the desert climate, the nature of which is determined by the intense solar radiation, dust, sand, relative humidity of about 5%, and temperature fluctuations from -10°C during night to +60°C during a day, to the Arctic having low temperature (-25 ...- 70°C) and high relative humidity (100%) and high content of ice dust in the air.

Limitations caused by the features of the helicopter operation in addition to the limitations imposed by the Flight Manual should also include limits defined by the helicopter purpose. Since all helicopters to some extent operate on hover near the ground, then average specific load p on the rotor disc area, which defines the flow rate thrown by rotor should be strictly limited.

For transport helicopter load $p \le 70 \ldots 80$ dPa, as the flow of air thrown by main rotor leads to erosion of the surface of the landing pad. For helicopter-cranes, used for erection and other works, wherein specially trained and equipped staff is located under helicopter in hover $p \le 50 \ldots 60$ dPa accordingly. For helicopters, which are used for the works requiring presence of people in the area hit by the MR thrown air (search and rescue helicopters), $p \le 30 \ldots 35$ dPa.

The group of technical restrictions is varied and determined by aerodynamic, strength, technology, design and other considerations. Therefore, most of these limitations are usually considered in the design of helicopter individual units.

However, there are some limitations common to all helicopters, without exception. These include the limit on stall. It is known that under certain flight conditions (flight at high speeds near the ground, at dynamic ceiling, climb, maneuvers with high overload, etc.) stall occurs at the main rotor blades. Approaching to stall at blades is manifested primarily in increase of hinge moments on the blades, alternating stresses therein and in booster control system, increase in helicopter vibration, its unbalance and worsening its controllability.

To exclude flights at such regimes it is accepted on helicopters not to exceed certain values of MR thrust divided by its solidity ratio $(C_т/\sigma)_{cr}$. Value of maximum allowable (critical) values of $(t_y)_{perm} = (C_т/\sigma)_{cr}\cos\alpha_e$, where α_e – angle between MR thrust and its lift in relation to relative flight speed $\overline{v} = v/(\omega R)$ at $\omega R = 220$ m/s for single-rotor helicopter is shown in Fig. 5.1 [3]. Here recommended values of $(t_{y0})_{max}$ are

represented in relation to dynamic ceiling H_{dyn} at $\omega R = 220$ m/s. At lower values of ωR the allowable values of $(t_{y0})_{perm}$ may be rather higher.

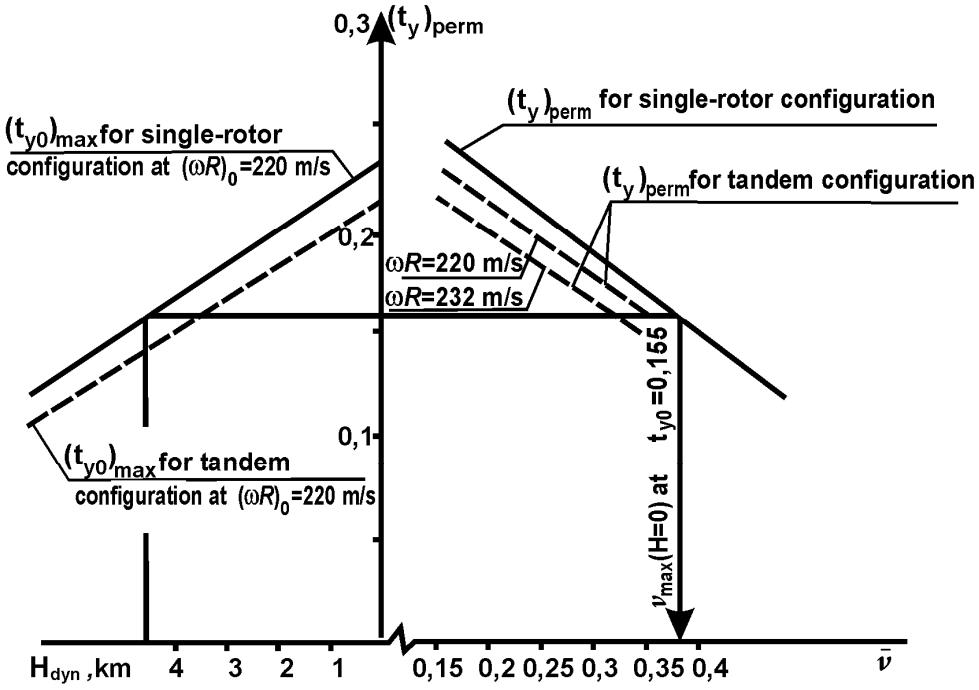


Fig. 5.1. Maximum Allowable Thrust Coefficients $(t_y)_{perm}$ by Characteristic of MR Operation Mode $\bar{v}$, Limited by Condition of Growing Alternating Loads in Control System Due to Approach to Stall, and Recommended Values of t_{y0} in Relation to Specified Dynamic Ceiling

For tandem helicopters values of $(t_{y0})_{perm}$ and $(t_{y0})_{max}$ must be rather lower due to more severe operation conditions of main rotors in level flights.

The important technical limitation influencing the parameter and configuration selection for helicopters with high load-carrying capacity is the power limit transmitted by the conical gear pairs of the helicopter main reducer. Maximum power [3], to be transmitted by a conical gear pair on strength condition can not exceed 5800 kW on the basis of the achieved technical level. That is why even application of multiengine transmission in the main reducer for super heavy helicopters can result in necessity not to use single-rotor configuration.

5.2. SELECTION OF HELICOPTER BASIC PARAMETERS

There are the following basic parameters of helicopter used in design process:

Takeoff weight m_0, kg;

Main rotor diameter D, m;

Disc area specific load p, dPa;

Engine power N, kW, or helicopter power-to-weight ratio N, kW/dN;

Main rotor solidity ratio σ;

Blade tip speed (ωR), m/s, or main rotor frequency of rotation n, min^{-1}.

Selection of helicopter basic parameters is performed first of all on condition of providing specified performance requirements;

Load-carrying capacity m_{load}, kg;

Flight range L, km;

Static ceiling H_{stat}, m;

Dynamic ceiling H_{dyn}, m;

Maximum flight speed v_{max}, km/h;

Cruising flight speed v_{cruis}, km/h;

Rate of climb v_y, m/s.

With that relative to specified basic data the direct and inverse design problems are distinguished.

For direct problem the following parameters are given: payload (transported cargo), flight range, dynamic and static ceilings, maximum and cruising speeds, flight, take-off and landing conditions and so on. To be determined: takeoff weight, power-to-weight ratio, disc area specific load and so on.

For inverse problem, occurring as a rule when helicopter is developed to be operated with particular engine or when modifying existing helicopter; and its takeoff weight is considered as given as it is explicitly defined by the engine power at given static and dynamic ceilings and selected load acting upon main rotor disc area. Weight of payload for given flight range or vice versa flight range for given cargo weight are the sought characteristics depending on parameters to be selected.

Using well-known Velner formula for main rotor thrust

$$T = \left(1.39\eta_0 \xi N D \sqrt{\Delta}^{2/3} \right), \tag{5.1}$$

It can be obtained the following expression describing relation between power and takeoff weight in hovering:

$$N = \frac{(m_0 g)\sqrt{p}\overline{T}^{3/2}}{0.5\eta_0 \xi \sqrt{\Delta}}, \tag{5.2}$$

where η_0 – main rotor relative efficiency; ξ – power usage factor; Δ – air relative density; $\overline{T}$ – main rotor relative thrust $\overline{T} = T/(m_0 g)$; N – engine power, W; m_0 – helicopter takeoff weight, kg.

In hovering main rotor relative thrust $\overline{T}$ is determined as

$$\overline{T} = 1 + \Delta\overline{T},$$

where $\Delta\overline{T} = \Delta\overline{T}_w + \Delta\overline{T}_f + \Delta\overline{T}_{HT}$ – relative increment of main rotor thrust required to overcome drag of wing, fuselage and horizontal tail being placed in flow created by rotor.

For single-rotor and coaxial configurations with rotors fitted with rectangular blades linearly twisted at an angle of 5...7°, it can be accepted [1]

$$\Delta\overline{T}_w = 0.375\overline{S}_w\overline{L}_w; \quad \Delta\overline{T}_f = 0.238\overline{S}_f \quad \Delta\overline{T}_{HT} = 1.38\overline{S}_{HT},$$

where S_w – wing relative area; $\overline{S}_w = S_w \Big/ \left(\pi R^2\right)$; $\overline{l}_w$ – wing relative span; $\overline{l}_w = \overline{l}_w / R$ $\overline{S}_F$ – fuselage horizontal projection relative area; $\overline{S}_f = S_f / \pi R^2$ $\overline{S}_{HT}$ – horizontal tail relative area; $\overline{S}_{HT} = S_{HT} / \pi R^2$.

For double-rotor (tandem and side-by-side) configurations it should be accounted the losses for main rotors to overlap

$$\Delta\overline{T}_{over} = 0.0667\left(2 - \overline{x}_{over}\right) \text{ at } \overline{x}_{over} = \left(x_{over}/R\right) \le 2,$$

where x_{over} (or z_{over} for side-by-side configuration) – distance between main rotor axes.

Power usage factor ξ estimates engine power losses for friction in transmission ΔN_{tr}, cooling ΔN_{cool}, hydraulic resistance of intake units $\Delta N_{hyd.r}$ and tail rotor drive system ΔN_{TR}:

$$\xi = 1 - \Delta\overline{N}_{tr} - \Delta\overline{N}_{cool} - \Delta\overline{N}_{hyd.r} - \Delta\overline{N}_{TR}.$$

As a rule relative power consumed for friction in transmission is assumed as $\Delta\overline{N}_{tr} = \Delta N_{tr} / N_{eng} = 0.02 \div 0.04$, relative power consumption for cooling $\Delta\overline{N}_{cool} = \Delta N_{cool} / N_{eng} = 0.01 \div 0.02$, relative power consumption for hydraulic resistance of intake units $\Delta\overline{N}_{hyd.r} = \Delta N_{hyd.r} / N_{eng} = 0.02 \div 0.03$, relative power required to drive tail rotor in hovering $\Delta\overline{N}_{hyd.r} = \Delta N_{hyd.r} / N_{eng} = 0.02 \div 0.03$ (in level flight $\Delta\overline{N}_{TR}$ = 0.02...0.04).

Relative to helicopter category it could be accepted the following values of ξ in hovering: ξ = 0.84...0.86 for light helicopter; ξ = 0.81...0.83 for medium helicopter; ξ = 0.78...0.80 for heavy helicopter.

In cruising flight for medium helicopter it could be accepted that ξ = 0.86...0.88.

Equation (5.2) gives clear relation between helicopter parameters and static ceiling being characterized with air relative density Δ at this altitude. Having used

equation (5.2) it could be selected the helicopter power-to-weight ratio $(N/mg)_{st}$, required in hovering at specified static ceiling and given value of p or to determine helicopter take-off weight for given engine.

One of the most important characteristic defining helicopter flight properties and maneuverability is its rate of climb. It is useful to estimate rate of climb by excess of power ΔN, determined as difference between power available and power required for hovering N_{hov} at design flight altitude.

According to formula proposed by V.I. Shaydakov a helicopter rate of climb

$$v_y = \frac{2\left(\Delta\bar{N} + 2\eta_0\right)\Delta\bar{N}\sqrt{p/\chi\Delta}}{\left(\eta_0 + \Delta\bar{N}\right)\eta_0}, \qquad (5.3)$$

where $\Delta\bar{N} = \Delta N/N$ – end losses factor.

Analysis of expression (5.3) shows that for given relative surplus power $\Delta\bar{N}$ the helicopter rate of climb is proportional to $\sqrt{p/\Delta}$. The same conclusion can be done with respect to the helicopter climb along an inclined path.

To consider the impact of parameters on a helicopter flight range you need to establish relation between these parameters and fuel consumption per kilometer q, since

$$L = \frac{m_{fuel}}{k_{fuel} q}, \qquad (5.4)$$

where m_{fuel} – fuel reserve, kg; L – flight range corresponding to this fuel reserve, km; k_{fuel} – coefficient accounting en-route fuel reserve (EFR), fuel consumption in transition modes, during take-off and landing as a rule k_{fuel} = 1.19.

It is known that fuel consumption per kilometer [3]

$$q = \frac{m_0 g c_{e\,cr}}{3.6 k_v \xi k_e}, \qquad (5.5)$$

where $c_{e\,cr}$ – fuel specific consumption in cruising flight, kg/(W·h); k_e –helicopter equivalent lift-to-drag ratio; ξ – factor accounting power usage and rotor efficiency; k_v – coefficient equal to ratio of scheduled speed v_{sch} to cruising speed v_{cr}.

As was noted the power usage coefficient in level flight of single-rotor helicopter has values, which differ from those in hovering. Having substituted (5.5) in (5.4), we got

$$L = \frac{3.6 k_v \xi k_e m_{fuel}}{k_v m_0 g c_{e\,cr}}. \qquad (5.6)$$

Maximum flight range with the same fuel amount and constant takeoff weight will be reached by a helicopter possessing the lowest fuel consumption and higher lift-to-drag ratio.

Helicopter equivalent lift-to-drag ratio

$$k_e = \frac{m_0 g v}{3.6 N_{l.fl}},$$

where v – speed in level flight, km/h; $N_{l.fl.}$ – power required in horizontal flight, W.

Using formula to determine power required for level flight proposed by L. S. Vildgrube [13], it may be written

$$N_{l.fl} = m_0 g \left(\frac{16.4 p I_{ind}}{\upsilon \Delta} + \frac{\omega R m_p}{C_T} + 13.2 \cdot 10^{-3} v^3 \overline{S}_{eq} \Delta \right), \qquad (5.7)$$

where I_{ind} – main rotor induction coefficient; $m_p/C_т$ – parameter characterizing profile losses of main rotor power; $\overline{S}_{eq}$ – relative specific drag of non-carrying members, $\overline{S}_{eq} = \sum C_x S / m_0 g = k_i \sum C_x S / m_0 g$, k – coefficient accounting interference between fuselage elements and main rotors.

Components in brackets in formula (5.7) represent power consumed to overcome induced and profile drags of main rotors and drag of non-carrying helicopter parts.

As it was shown in calculations [1], a share of this consumption differs with flight speed. At low speed the highest power consumption (up to 70...75 %) is taken to overcome induced drag N_{ind}, at medium speed up to 45...50 % power is taken to overcome the profile drag N_{pr} and up to 40 % are induced losses, and at high speed main losses are due to overcome profile and parasitic drag N_{adv}.

Accordingly helicopter parameters are selected on condition of maintaining the highest lift-to-drag ratio during the longest and often repeated regimes of flight. At low speed it is necessary to reduce inductive losses by selecting proper shape of planform and twist of blades, MR diameter and therefore specific load p. At medium and high speed pay more attention to selection of cross-section profiles and better surface finishing, to reduce profile drag and area of adverse plate $\sum C_x S$ that is necessary to shorten losses for overcoming drag.

Fig. 5.2 [1] shows values of fuel relative consumption per kilometer $\overline{q} = q/(m_0 g)$ in relation to flight speed v for helicopters having takeoff weight within the range of 1...50 tons, having averaged statistics values of $\sum C_x S/(m_0 g)$ and p, MR rectangular blades with twist $\Delta\varphi$ = 7°, set of cross-sections NACA 23012, NACA 23009, NACA 23006 and blade tip speed ωR = 220 m/s.

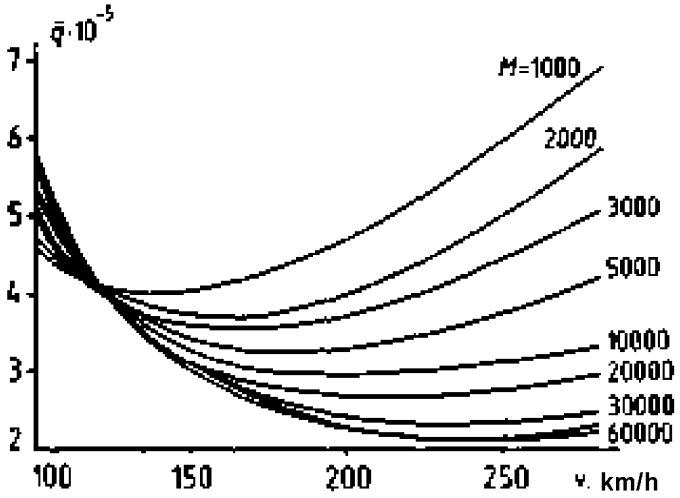


Fig. 5.2. Change of Relative Fuel Consumption per Kilometer q in Relation to Flight Speed v for Helicopters of Different Weight Category

Fig. 5.3 shows values of $\bar{q}_{min}$ and corresponding flight speed $v_{q_{min}}$ for different helicopter takeoff weight m_0. With rising takeoff weight and corresponding change of $\sum C_x S/(m_0 g)$ and p minimum value of relative fuel consumption per kilometer rapidly reduces, and corresponding flight speed grows.

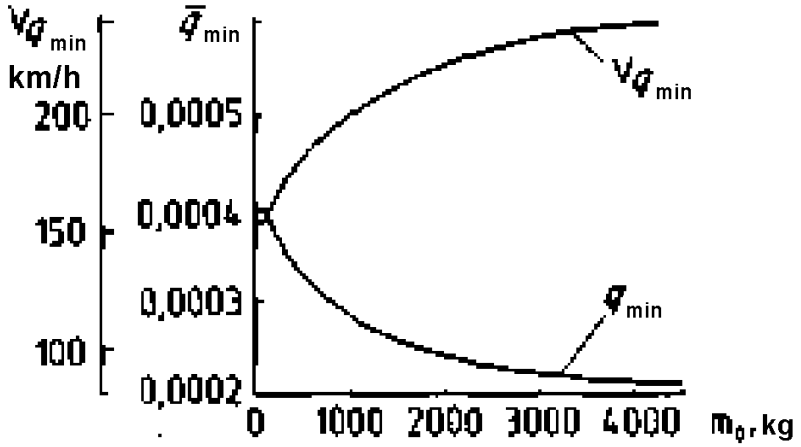


Fig. 5.3. Minimum Relative Fuel Consumption per Kilometer $\bar{q}_{min}$ and Corresponding Flight Speeds

Fig. 5.4 shows values of coefficient K_{Emax} – maximum equivalent lift-to-drag ratio of helicopters with medium cargo-carrying capacity inversely to relative fuel consumption per kilometer.

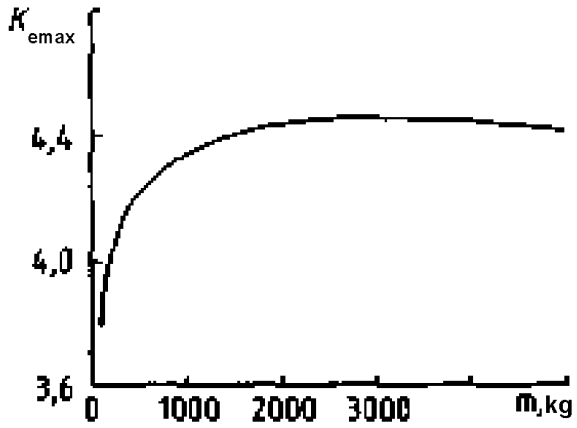


Fig. 5.4. Coefficient K_{Emax} – Maximum Equivalent Lift-to-Drag Ratio of Helicopters with Medium Cargo-Carrying Capacity

Helicopter basic parameters are interrelated and directly influence the performance characteristics. That is why the process of selecting basic parameters is of iterative nature and is executed by means of method of successive approximations considering acting restrictions and given precision of calculations. For that the following order of calculations may be accepted.

1. Determination of helicopter takeoff weight in the first approximation

Having specified values of payload m_{pl} and crew members m_{crew} on the basis of basic data and values of relative weight of empty helicopter $\bar{m}_{emp}$ and fuel $\bar{m}_{fuel}$ by statistics, calculate helicopter takeoff weight

$$m_0^1 = \frac{m_{pl} + m_{crew}}{1 - \bar{m}_{emp} - \bar{m}_{fuel}}.$$

Using formula (5.6) it is possible to determine relative fuel weight $\bar{m}_{fuel}$ with respect to flight range

$$\bar{m}_{fuel} = \frac{k_{fuel} g c_{ecr} L}{3.6 k_v K_e \xi} = k_{fuel} \bar{q}_{fuel} L, \qquad (5.8)$$

where $\bar{q}_{fuel}$ – fuel relative consumption per kilometer, kg/(kg·km).

In this case on the basis of statistics it should be given c_{ecr} and K_e, and values of k_v and ξ are taken according to given recommendations.

Modern helicopters of medium cargo-carrying capacity with turboprop engines have relative fuel consumption per kilometer $\bar{q}_{fuel}$ =(2.2...2.4)·10^{-4} kg/(kg·km).

If the Terms of reference does not contain given flight range but flight duration then relative fuel weight for flight in economic regime may be calculated as

$$\bar{m}_{fuel} = 0.007t\sqrt{p}, \quad (5.9)$$

where t – flight duration, hours; p – disc area specific loading, dPa.

If hovering is a prevailing regime then

$$\bar{m}_{fuel} = 0.012t\sqrt{p}. \quad (5.10)$$

2. Determination of MR Disc Area Specific Load

When selecting specific load p first of all consider the helicopter purpose, its carrying capacity, number of engines and operating conditions. As specific load p defines MR diameter and therefore its weight then it is observed a clear tendency of growing p with increasing carrying capacity. That is why the upper limit of MR specific load is to be defined.

For single-engine helicopter with the purpose to provide higher safety in flight there must be provided its landing in autorotation mode with vertical speed v_y not exceeding maximum allowable speed of descent for safety landing.

As $v_{y\,allow}$ = 6...8 m/s, and vertical speed while gliding in autorotation mode $v_a \approx 1.5\sqrt{p}$, that corresponds to gliding speed along the trajectory of 120...150 km/h and gliding angle not more 10...14°, having equated v_{yallow} with v_a, maximum allowable values of p = 20...25 dPa were obtained.

For two-engine, three-engine and multiple-engine helicopters the upper limit is defined depending on the helicopter purpose and restrictions given above.

Lower limit of main rotor specific loading is defined by the ratio of MR growing weight from one hand and reducing required thrust-to-weight ratio in hovering from another hand. That is why when selecting specific load it is recommended to accept values close to similar vehicles and define more exactly by the following parametric analysis. For that use I.A. Serov empirical formula [40] for p, dPa:

$$p = 2.05m_0^{0.314}. \quad (5.11)$$

Using selected loading p and previously determined helicopter takeoff weight m_0^I it is easy to obtain MR diameter, m:

$$D = \sqrt{\frac{4gm_0^1}{10\pi p}} = 1.12\sqrt{\frac{m_0^1}{p}}.$$

3. Calculation of MR rotary velocity. When selecting MR rotary speed consider different effect of this parameter upon required helicopter thrust-to-weight ratio and performance characteristics under different flight conditions. It is known that in hovering

value of ωR essentially influences MR relative efficiency η_0 and therefore required thrust-to-weight ratio. The lower rotary speed the lower power is required for helicopter hovering at the given ceiling. As reduction of ωR is limited then for given engine power and static ceiling and selected values of D and solidity ratio σ an optimal value of ωR exists, providing maximum MR thrust.

In horizontal flight value of ωR influences flight characteristics caused by compressibility and stall.

The measure of compressibility effect upon main rotor characteristics is the Mach number at the end of advancing blade (that is the blade moving towards the oncoming flow caused by horizontal speed component):

$$M = \frac{(v + \omega R)}{a} = M_0(1 + \bar{v}), \tag{5.12}$$

where a – acoustic speed; $\bar{v} = v/\omega R$; $M_0 = \omega R/a$.

Hence

$$v = aM\left(\frac{\bar{v}}{1 + \bar{v}}\right). \tag{5.13}$$

Allowable value of M should not exceed definite value of M_{cr}, at which compression shocks originate at MR blades and profile drag rises abruptly. On modern helicopters value of M_{cr}<0.92...0.95 (except for Westland G-Lynx, with M_{cr} =0.97).

Having specified values of M and $\bar{v}$, from (5.12) it may be determined

$$\omega R = \frac{aM}{L + \bar{v}} \tag{5.14}$$

And built graph of limit values $\omega R = f(v)$ (Fig. 5.5).

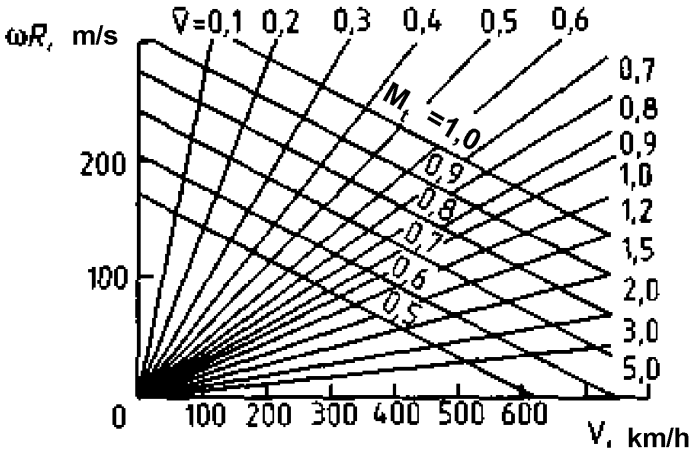


Fig. 5.5. Rotary Speed ωR vs Flight Speed v
for Different Values of M_{cr} and $\bar{v}$

At high flight speeds and together with rising altitude when in addition to compressibility the stall influences at the retreating blade then to reduce C_T and postpone stall in such way it is recommended to increase ωR. That is why when selecting MR rotary speed as a rule the capability of gas turbine engines to change frequency of rotation of free turbine within the range of 10…12 %, that is n_{max}/n_{min} = 1.1…1.12 is used. With that value of ωR in hovering is accepted to be equal to optimal value (for given p and σ), which by statistics equals to the following numbers: for heavy helicopters ωR = 200...230 m/s; for medium helicopters ωR = 200...210 m/s; for light helicopters ωR = 180...200 m/s. In cruising flight at altitude of 500 m with the purpose to reduce power required parameter ωR is reduced by 3...5 %, and during flight with maximum speed or when flying at dynamic ceiling to postpone stall the value of ωR is increased by 5...6 %.

MR number of revolutions is determined by the selected value of ωR

$$n = \frac{30\omega R}{\pi R}. \qquad (5.15)$$

Maximum allowable MR number of revolutions in flight under any regime is limited by the following conditions:

- Providing strength of main reducer, hub, swash plate, blades with regard to centrifugal forces that results in growing weight;
- Preventing compressibility stall at the MR advancing blade tips when flying at high speeds and altitudes;
- Providing sufficient margin in MR blade flutter;
- Excluding possibility of engine free turbine protection from spin-up that is from the engine spontaneous stop in flight;
- Abrupt rise of vibration level and variable stress in MR blades due to non-stationary nature of the compressibility stall development;
- Rising shaking of helicopter;
- Beginning of power abrupt rise required for MR rotation.

4. Determination of main rotor solidity ratio. As it was mentioned above there is the following condition of no stall at main rotor

$$\frac{C_T}{\sigma} \le \left(\frac{C_T}{\sigma}\right)_{cr}. \qquad (5.16)$$

Since at the selected values of p and σ thrust coefficient is determined explicitly

$$C_T = \frac{2p}{p(\omega R)^2}, \qquad (5.17)$$

Then condition (5.16) may be provided by proper selection of σ.

As a rule

$$\sigma = \frac{C_T}{(C_T/\sigma)_{cr}} = \frac{2p}{p(\omega R)^2 (C_T/\sigma)_{cr}}. \qquad (5.18)$$

Value of $(C_T/\sigma)_{cr} = \bar{C}_{T.cr}$ may be determined by expression $\mu = \bar{v}\cos\alpha_{\acute{y}}$ (Fig. 5.6) [1]. With that $\bar{v}_{max}$ is found by Fig. 5.5. Such matching allows to select parameters so to account stall and compressibility simultaneously. Data shown in Fig. 5.6 well match with wind tunnel test results and in flight near ground (at H = 500 m). To account stall at the dynamic ceiling look at Fig. 5.1. Having assumed $C_т \approx C_y$ of main rotor and giving altitude of dynamic ceiling H_{dyn} on the graph left side it is possible to determine allowable values of $(C_т/\sigma)_{max}$, which are accepted as $(C_т/\sigma)_{cr}$. Note that increase in σ, leading to postpone stall at maximum flight speed, causes in its turn, increase in main rotor weight and reduction in helicopter weight efficiency. Therefore account this fact when selecting main rotor solidity ratio. Higher solidity ratio σ may be obtained in the following ways: increase of blade chord at their constant number, or increase of number of blades at the same blade chord. Excessive increase of blade chord results in rise of hinge moments and loads in control linkage. That is why more often for main rotors with hinged blades the number of blades is increased, having established solidity ratio for one blade within the range σ/k = 0.016...0.022, where k – number of blades. With that optimal number of blades is determined considering other factors influencing operating conditions and flight safety (vibration level, noise etc).

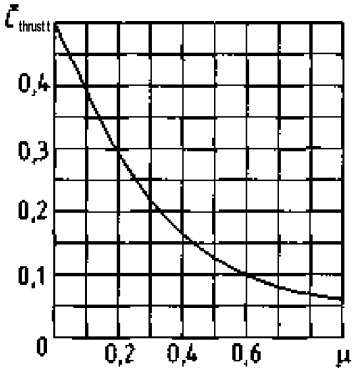


Fig. 5.6. Change of Critical Values $C_{T.cr}$ for Stall Condition on Helicopter Relative Speed μ

Real filling of area with blades $F = \pi R^2$ depends on number of blades z and their planform. Value of this filling is characterized with solidity ratio σ:

$$\sigma = \frac{zF_{bl}}{\pi R^2} = \frac{zb_{0.7}}{\pi R} = \frac{z}{\pi\lambda_{av}},$$

where z - number of blades; F_{bl} - blade area in planform, $F_{bl} = b_{0.7}R$; $b_{0,7}$ - blade chord in section at $0.7R$; λ_{av} - averaged blade aspect ratio, $\lambda_{av} = R / b_{av}$.

5. Identifying required power-to-weight ratio – the ratio of engine power requirement to the helicopter weight $N/(m_0 g)$ is determined by calculation and comparison of required power available under variety of extreme flight conditions. As the most intense flight regimes usually the following modes are considered:

Hovering at static ceiling;

Flight with maximum speed at altitude H = 500 m;

Flight with economical speed at dynamic ceiling;

Flight with one engine failed (horizontal flight or continued flight).

The amount of required energy available to hover at predetermined static ceiling, as already mentioned, at the chosen value of p can be determined by formula (5.2):

$$\left(\frac{N}{m_0 g}\right)_{H_{st}} = \frac{\sqrt{p}\overline{T}^{3/2}}{0.5\eta_0 \xi\sqrt{\Delta}}.$$

For calculation of required power for horizontal flight modes it is necessary to carry out the helicopter aerodynamic estimation in order to determine the power consumed for overcoming the profile and inductive drag of the carrying system and harmful (frontal) drag of non-carrying helicopter parts. This calculation is quite complex and requires a large number of the basic information, which in the early stages of the design is still missing. Therefore, for the first pre-calculation of the required power, you can use the approximate method of aerodynamic estimation proposed by L.S. Vildgrube [1]. In accordance with this method, the relative power need for helicopter level flight not fitted with a wing, can be determined by formula similar to (5.7),

$$\frac{N_{h.fl}}{m_0 g} = 16.4\cdot 10^{-3}\omega R\left(1 + 7.08\cdot 10^{-8} v^3\right) + \frac{16.4 p I_{ind}}{v\Delta} + 13.2\cdot 10^{-3}\overline{S}_{eq} v^3 \Delta, \quad (5.19)$$

where $N_{h.fl}$ – power, required for level flight, W.

Rotor induction coefficients I_{ind} having rectangular blade with twist angle of 7° are shown in Table 5.1. For double-rotor helicopter take into account mutual inductive influence of rotors:

$$I^*_{ind} = I_{ind} + \Delta I_c,$$

where I^*_{ind} – double-rotor helicopter induction coefficient; ΔI_c – additional induction coefficient taking into account the rotor mutual influence.

Table 5.1

Main Rotor Induction Coefficients

V, km/h	150	200	250	300	350	400	Configuration
I_{ind}	1.09	1.10	1.12	1.18	1.28	1.38	Single-rotor
ΔI_c	0.66	0.65	0.64	0.63	0.60	0.56	Coaxial $y/R = 0.2$
	0.66	0.65	0.64	0.60	0.55	0.49	Tandem $y/R = 0.2$
	-0.34	-0.35	-0.36	-0.38	-0.40	-0.43	Side-by-side $z/R = 0.2$

Relative drag of the projected helicopter $\overline{S}_{eq}$ can be determined in the way of element-by-element calculation of parasitic drag of all helicopter non-carrying parts. You can use the results of statistical analysis and forecasting of helicopter technological evolution at the initial stages of the preliminary design. Table 5.2 shows the values $\overline{S}_{eq}$ obtained for the materials of paper [1].

Table 5.2

Helicopter Relative Drag

m_0 (t)	2.5	5.0	10	20	30	40	50	Helicopter
$\overline{S}_{eq} \cdot 10^4$	0.61	0.46	0.37	0.29	0.25	0.22	0.2	1960
	0.38	0.3	0.25	0.18	0.15	0.14	0.12	Enhanced shape
	0.24	0.2	0.17	0.13	0.11	0.09	0.086	Perspective

Value of $\sum C_x S$ in m^2 depending on the helicopter take-off weight can also be calculated using the empirical formula proposed by I.A. Serov [2]:

$$\sum C_x S = 0.018 m_0^{0.5646}. \quad (5.20)$$

Economical flight speed, km/h, at which helicopter flies at the dynamic ceiling and in case one engine failure can be found from the condition $(N_{h.f})_{min}$ according to the following relationship:

$$v_{ec} = 260 \sqrt[4]{\frac{pI_{ec}}{\left(\omega R + 11.4 \cdot 10^6 \overline{S}\Delta\right)\Delta}}. \quad (5.21)$$

Calculation of the necessary power-to-weight ratio available with maximum flight speed is conducted under basic values of v_{max} specified in the terms of reference, and when flying at dynamic ceiling – for set values of H_{dyn}. The power-to-weight ratio required for take-off (index ab.t-off) and en-route flight in case of one engine failure for helicopter of category B is determined by the conditions of maintaining of horizontal flight at the most advantageous flight speed v_e corresponding to the highest rate of climb:

$$\left(\frac{N}{m_0 g}\right)_{ab.t-off} = \left(\frac{N}{m_0 g}\right)_{v_{ec}} \cdot \frac{n}{n-1}, \qquad (5.22)$$

where n – number of engines.

For helicopters of category A the power-to-weight ratio required is determined based on the conditions of maintaining rate of climb of at least 0.5 m/s at safe flying speed, which can be taken equal to economic speed.

5.3. HOW TO SELECT ENGINE

Currently, in most cases double-shaft (with free turbine) turboprop engines are used as a power plant having possessing smaller dimensions, lower specific gravity and being more simple and reliable in operation compared with piston engines. The use of piston engines is limited to only light sporting or training-communicating helicopters, since for this weight category the engine with acceptable performance is not yet created. Turboprop engine with free turbine is a turboshaft engine having good stability in operation with small (within the range of 10 ... 12%) change in the free turbine speed, which is used when choosing main rotor circumferential speed for different flight modes.

When selecting an engine you need to determine its maximum drive power sufficient to provide all the helicopter flight characteristics, given in the terms of reference. This relates primarily to the used engine mode under current flight conditions, as well as its altitude, speed and temperature performances. The engine operating mode is largely determined by the flight duration, as take-off and rated engine speed is limited in time (respectively 5 minutes and 1 hour). Furthermore, it is necessary to consider the change of the engine performance (power and fuel rate) depending on the altitude, airspeed and the ambient temperature.

To account for all these factors when choosing the required engine power the design data for extreme flight regimes should be reduced to the international standard atmosphere conditions at zero height ($H = 0$). The values of the reduced power requirement for the four considered modes of flight are defined as follows:

$$\left(\frac{N}{m_0 g}\right)^{red}_{H_{st}} = \frac{\left(N/m_0 g\right)_{H_{st}}}{\bar{N}_H \bar{N}_t};$$

$$\left(\frac{N}{m_0 g}\right)^{red}_{v_{max}} = \frac{\left(N/m_0 g\right)_{v_{max}}}{\bar{N}_H \bar{N}_t \bar{N}_v \xi_{v_{max}}};$$

$$\left(\frac{N}{m_0 g}\right)^{red}_{H_{dyn}} = \frac{\left(N/m_0 g\right)_{H_{dyn}}}{\bar{N}_H \bar{N}_t \bar{N}_v \xi_{v_э} \bar{N}_{nom}};$$

$$\left(\frac{N}{m_0 g}\right)^{red}_{ab.t-off} = \frac{\left(N/m_0 g\right)_{ab.t-off}}{\bar{N}_H \bar{N}_t \bar{N}_v \xi_{v_э}}.$$

Functions $\bar{N}_H$, $\bar{N}_t$, $\bar{N}_v$, characterizing the power change with regard to altitude, ambient temperature and speed, are well enough described by the following empirical dependence:

$$\bar{N}_H = 1 - 0.0695\left(\frac{H}{1000}\right);$$

$$\bar{N}_t = 1.1 - 0.0066t;$$

$$\bar{N}_v = 1 + 5.5 \cdot 10^{-7} v^2,$$

where H – flight altitude, m; t – air temperature, °C; v – flight speed, km/h; N – engine part-loading; $\bar{N} = N/N_{t-off}$.

Engine is selected by the largest value of the required reduced power and helicopter take-off weight:

$$N_{eng} = 1/n\left(N/m_0 g\right)^{red}_{max}\left(m_0 g\right), \qquad (5.23)$$

where n – number of engines.

By the parameter N_{eng} a real engine is selected with power not lower than this value. If take-off power of real engine essentially differs from the highest reduced power demand, then correct takeoff weight, payload and flight range on the basis of real engine power, and also change helicopter selected parameters and performance characteristics.

In most cases terms of reference for helicopter under development contain type, number and power of engines to be installed in the helicopter. In this case maximum reduced power allows to determine (Ref. formula 5.23) helicopter maximum permissible takeoff weight m_0 and to obtain payload for specified flight range or duration having performed weight calculations. With that characteristics $\bar{N}_H, \bar{N}_t, \bar{N}_v$, γ, c_e and others are taken from the engine certificate.

Helicopter power demand as known essentially depends on flight regimes. Engine power available within the range of operating speeds depends on flight speed not much. Different relation of available and required power with flight speed allows to obtain a margin of power available for other flight regimes having agreed the engine parameters and helicopter in hover. With that if engine in hovering operates with take-off power than during any other flight regime power required will be provided by cruising power.

As a rule the first (maximum) regime with not limited engine operation time corresponds to 0.75...0.80 of takeoff power. Flight with cruising power may be executed within wide range of speeds. Two specific regimes are of interest: maximum flight duration with speed close to economic value and maximum range close to optimal regime. The first regime corresponds to minimum fuel consumption per hour, and the second – to minimum consumption per kilometer.

If while selecting engine there are several engines having the same takeoff power and the same technical level then that engine to be preferred, which has better characteristics c_e and γ. These factors essentially influence the flight range and payload.

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Chapter 6. CALCULATION OF HELICOPTER WEIGHT

6.1. CLASSIFICATION BY WEIGHT

Takeoff weight m_0 is one of the helicopter main parameters, which defines its dimensions, configuration and performance requirements. Its correct determination is the most important task of the entire design process. The complexity of calculating a helicopter take-off weight at the early stages of the design process is linked primarily to lack of the basic information about the helicopter and its power unit as well as impossibility of taking into account requirements for the helicopter at this design stage.

Therefore takeoff weight must be determined by the method of successive approximations, using first the approximate empirical formulas and calculation methods, and then, with the expansion of the available information, more accurate formulas and methods of calculation taking into account the conditions of loading the units, the requirements of strength, stiffness, etc.

An important and necessary condition for the correctness of the take-off weight at all stages of the design is the unity of its classification. To date, there is no single standard helicopter classification by weight. Classifications of helicopters by weight used in a variety of companies and organizations involved in the helicopters design and operation differ from each other as in degree of completeness and composition of weight groups, as in the terminology used.

Currently, in the classifications of aircraft and helicopters all over the world the concept of "standard empty weight" appears as the weight of an empty helicopter in the minimal complete set of delivery, which is used when the helicopter is certified for flight. At the request of the customer in a complete set of delivery of the empty helicopter an additional removable equipment (optional) may be included that enhances its operational capabilities (by reducing payload at the same values of fuel amount and take-off weight).

Fig. 6.1 shows a variant of the helicopter weight classification corresponding to the established in the Russian helicopter engineering breakdown by the take-off weight.

At the same time the following definitions are adopted:

Gross weight **–** helicopter total weight at any time of its operation, including weight without payload, with minimal fuel amount for the helicopter flight;

Take-off weight **–** helicopter total weight at the beginning of takeoff (beginning of takeoff with running) or movement to lift-off moment in the take-off area (without run off);

Estimated take-off weight **–** helicopter maximum take-off weight set for all load cases, under which operation of the helicopter is made without any special limitations;

Normal take-off weight **–** helicopter weight, which provides fulfillment of the requirements put forth for the helicopter flight data with normal take-off weight, including

range and mission load during operation under the given conditions;

Maximum take-off weight **–** helicopter weight, which provides fulfillment of the requirements put forth the helicopter flight data with the maximum takeoff weight.

At the same time the following breakdown of the helicopter take-off weight by groups is adopted:

Empty weight m_{emp} consisting of airframe weight $m_{a.fr}$, powerplant weight m_{pp} and equipment weight m_{eq};

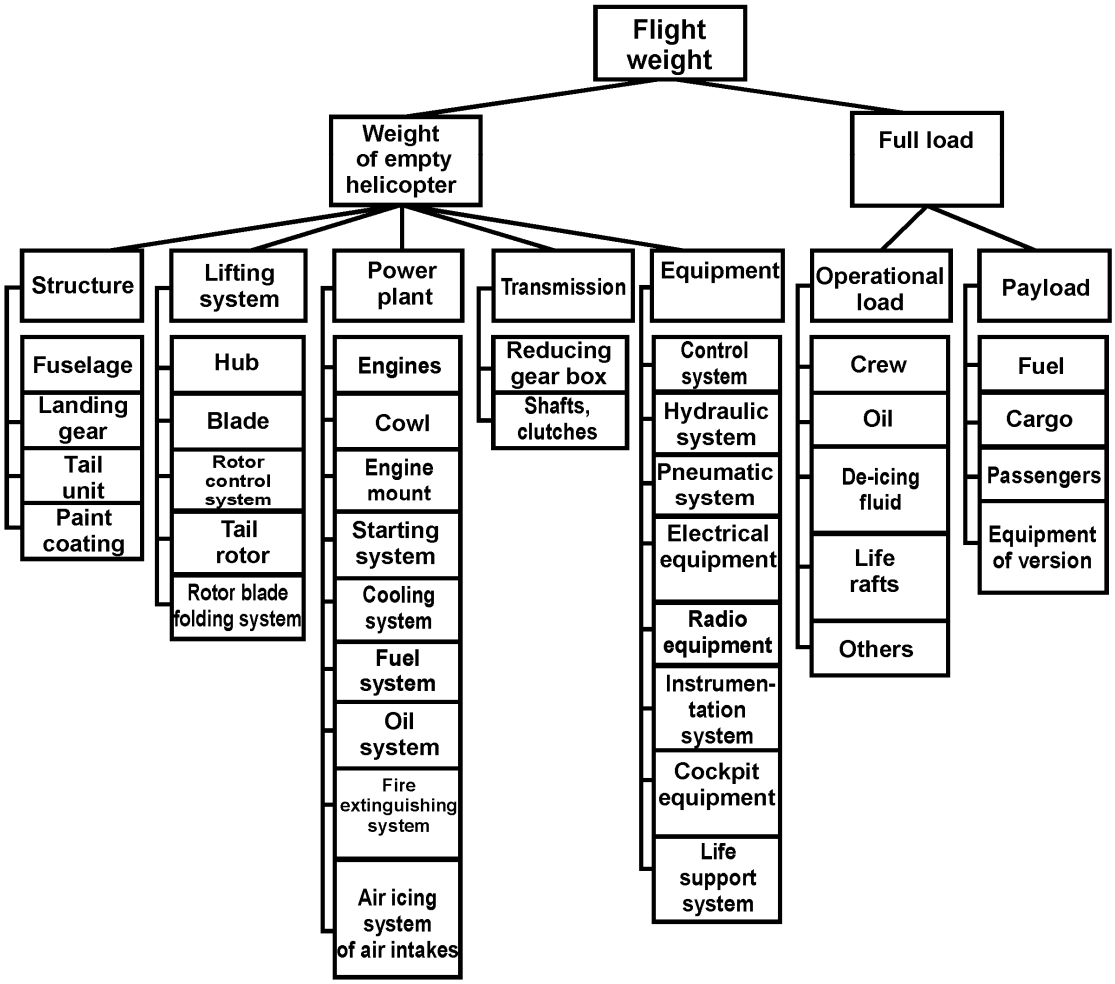


Fig. 6.1. Optional Weight Classification

Full load weight, defined by the helicopter purpose and consisting of weights of crew, equipment, consumable fuel (including reserve stock) and payload.

Weights of airframe, power plant and non-removable general-purpose equipment form the basis original empty weight. The term "basis weight of empty helicopter" normally missing in aircraft classifications, allows to allocate the bulk of the weights of the empty helicopter, common to all modifications of this type helicopter, which facilitates comparative analysis of helicopters, with different purposes and provides

additional comfort during operation and further modifications of the helicopter.

Airframe weight $m_{a.fr}$ includes:

Fuselage weight m_{fus}, including the nose and central parts, tail boom and tail pylon, ladders, cargo flaps, fuselage joints, fuel tanks and other units included in the loading pattern of the fuselage;

Wing weight m_w including panels and center section, struts, strut braces and wing fuel tanks and other units entering into the wing loading pattern;

Tail unit weight m_{tu} including fins, stabilizers, plates, struts, strut braces, manhole covers, etc.;

Cowling weight m_{cowl}, including firewalls, air intakes, etc.;

Landing gears weight m_{LG} including struts and shock absorbers, wheels (floats and skids), tail wheel fairings, retraction and extension mechanisms;

Control system weight m_{cs}, including control stations, control linkage, hydraulic actuators, servos, differential mechanisms.

Power plant weight m_{pp}:

Propulsion system weight $m_{pr.s}$, including engines in delivery set, starting systems, intake and exhaust systems, engine mounts, engine frames, etc.;

Transmission weight m_{tr}, including reducers and their mounts, shafts, bearings and their components, transmission brake, lubrication systems, gear frame;

Weight of rotors m_{rot}, including blades and hubs of main and tail rotors, dampers, mechanisms of folding and retraction of blades;

Fuel system weight m_{fs} with undrainable fuel;

Oil system weight m_{os}, including external lubrication system of engines and transmissions;

Cooling system weight m_{cs}, including fans and air flow ducts;

Fire extinguishing system weight m_{fe}.

Weight of general purpose equipment m_{eq} mandatory for installation includes:

Electrical equipment, including the sources of electricity, power generating sets, converters, transformers and rectifiers, junction boxes, cables, wiring and mounting of electrical connectors;

Mounts of instrumentation, electrical and electronic equipment;

Pneumatic and hydraulic equipment, with the exception for pneumatic and hydraulic power consumers;

General purpose equipment of crew cabins, including ventilation and heating

systems, lighting, fixed rigging equipment, seats, etc.;

Auxiliary equipment ensuring the efficiency and safety of the helicopter in special conditions, such as units of the anti-icing systems non-embedded in the structure, means of escape and evacuation, emergency equipment, windscreen wipers, fire extinguishers, etc.

Non-removable equipment weight m_{nr} to be added to the original weight of empty helicopter to obtain standard weight of the empty helicopter includes the following weights:

Control onboard equipment, including devices and signaling system for the electronic display board, automatic registration and control board recorders;

Navigation equipment, including instruments and signaling devices in standard set, electronic display navigation equipment, heading system and other flight control and navigation systems, external lighting and wiring equipment, flares, windshield wipers, anti-icing system;

Communication equipment, including the command and connected radio identification systems, intercoms;

Equipment of cabins, including air conditioning, cabin pressurization system, heating and ventilation enclosures, vibration dampers, finishing and thermal and acoustic insulation of cabins, passenger, rigging, mooring, search and rescue and other equipment;

Weapons;

Armoring;

Other mission hardware;

Switching equipment in standard set;

Wiring and harnesses in standard set;

Fixtures and racks in standard set.

Weight of additional equipment includes target weight of related equipment to be installed in addition to the standard equipment of general purpose:

Mission instrumentation and electronics not included in the helicopter basic empty weight (e.g., automatic control systems, radars, etc.);

Mission equipment for cabins (for example, for cargo containers);

Additional mission equipment (weapons, armoring, additional sources of hydraulic and pneumatic power).

Weight of crew and operational items includes the following:

Crew (pilot, navigator, radio operator with clothing, parachutes and other personal equipment);

Operational items including weights of removable and discharged pieces of equipment, structure and power plant, spent fluids and substances, removable ballast etc.;

Oil in the outer oil systems of the engines and transmission, including oil in the tanks, radiators, pipes;

Unusable and undrainable fuel residues.

Weight of fuel m_{fuel} includes fuel that can be consumed from main, additional and discharged fuel tanks (including en-route fuel reserve).

Weight of payload refers to weights of all kinds of goods, which could be transported by the helicopter.

Helicopter empty weight with the weight of crew with equipment forms a curb weight of the helicopter.

6.2. PREDEFINED HELICOPTER TAKEOFF WEIGHT

Helicopter takeoff weight m_0 consists of weights of empty helicopter m_{emp}, weights of crew and operational items m_{crew}, payload weight m_{pl} and fuel weight m_{fuel}.

$$m_0 = m_{emp} + m_{crew} + m_{pl} + m_{fuel}. \quad (6.1)$$

Parameters m_{fuel}, m_{emp}, to a large extent depend on the helicopter take-off weight, its parameters and performance characteristics. To reduce this impact divide both sides of equation (6.1) by m_0:

$$1 = \bar{m}_{emp} + \bar{m}_{fuel} + \frac{(m_{crew} + m_{pl})}{m_0}. \quad (6.2)$$

The equation of the relative weights (6.2) allows to determine the helicopter take-off weight as the first approximation, taking as input data for the technical assignment the weights of payload and crew. Solving the equation (6.2) with respect to m_0, we get

$$(m_0)^1 = \frac{(m_{crew} + m_{pl})}{1 - \bar{m}_{emp} - \bar{m}_{fuel}}. \quad (6.3)$$

Parameters m_{emp} and m_{fuel} can be determined according to the statistics based on the analysis of prototypes.

The *relative helicopter empty weight* characterizes the degree of helicopter perfection and relates with its ***useful load fraction*** $\bar{m}$

$$\bar{m}_{emp} = \frac{m_0 - m_{pl}}{m_0}; \quad (6.4)$$

$$\bar{m}_{emp} = 1 - \bar{m} \ .$$

Since the useful load fraction of modern helicopters, depending on their purpose and duty $\bar{m}$ = 0.42 ... 0.52 for transport helicopters and $\bar{m}$ = 0.32 ... 0.38 for anti-submarine helicopters (Fig. 6.2), respectively it can be taken that $\bar{m}_{emp}$ = 0.48 ... 0.6

for transport helicopters and $\bar{m}_{emp}$ = 0.62...0.68 for anti-submarine and special purposes helicopters.

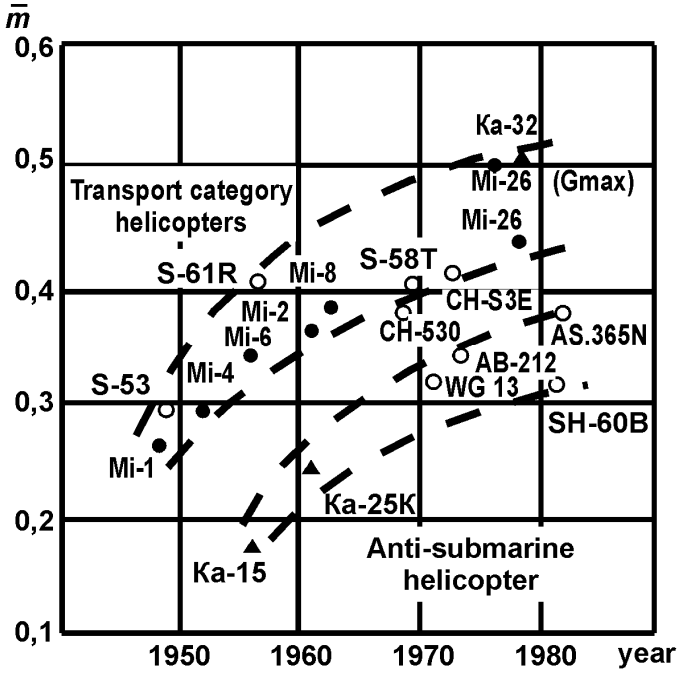


Fig. 6.2. Useful Load Fraction $\bar{m}$ of Antisubmarine and Transport Helicopters According to First Flight Year

The estimated change in the relative weight of empty transport helicopter $\bar{m}_{emp}$ depending on its take-off weight m_0 is shown in Fig. 6.3.

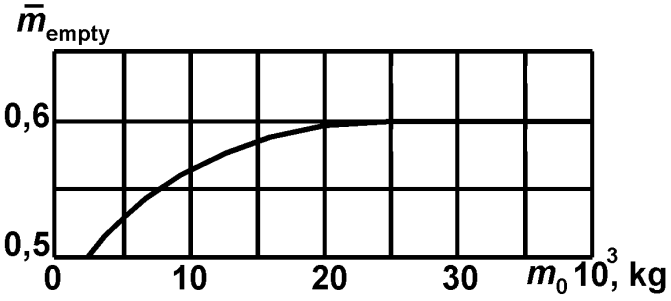


Fig. 6.3. Change in Relative Weight of Empty Helicopter $\bar{m}_{emp}$ Depending on Its Takeoff Weight m_0

The fuel relative weight $\bar{m}_{fuel}$ in the first approximation can be determined depending on the given flight range by the formula

$$\bar{m}_{fuel} = \bar{q}_{fuel} L + 0.33 \bar{Q}_h, \quad (6.5)$$

Where $\bar{q}_{fuel}$ - relative fuel consumption per kilometer (divided by the helicopter take-off weight), 1/km; $\bar{Q}_h$ - relative fuel consumption per hour, 1/h; L – flight range,

km. For helicopters with take-off weight m_0 less than 10 tons it may be taken

$\bar{q}_{fuel} = (0.25...0.3) \cdot 10^{-3}$ и $\bar{Q}_h = (5.9...6.3) \cdot 10^{-2}$;

For medium-lift helicopters (10 t < m_0 < 25 t)

$\bar{q}_{fuel} = (0.22...0.24) \cdot 10^{-3}$ and $\bar{Q}_h = (5.7...5.8) \cdot 10^{-2}$;

For heavy helicopters (m_0 > 25 t)

$\bar{q}_{fuel} = (0.19...0.21) \cdot 10^{-3}$ и $\bar{Q}_h = (5.5...5.6) \cdot 10^{-2}$.

Smaller values of $\bar{q}_{fuel}$ and $\bar{Q}_{fuel}$ correspond to helicopters of heavier carrying capacity.

6.3. CALCULATION OF WEIGHTS OF HELICOPTER MAIN UNITS

To clarify the helicopter predetermined take-off weights at all subsequent stages of its design and manufacture the helicopter empty weight is to be determined, as it is 50 ... 60% of the helicopter take-off weight, and it determines the degree of the weight perfection. Depending on the design stage to determine the weights of helicopter units and systems use various methods of calculation, differing with accuracy, volume of basic data necessary for calculations, time of calculations.

Basic methods for calculating weights of helicopter units and systems at various design stages can be divided into three groups:

1) based only on the use of statistical data (empirical);
2) semi-empirical or combined, based on the account of fundamental characteristic dimensions of units and main loads defining section details. These methods widely use statistics, since the weight factors included in the formulas are determined on their basis;
3) based on individual calculation of every part of the units and systems. This calculation is only possible after issue of working drawings, which have passed the last comprehensive review in terms of manufacturability and durability of parts.

At the preliminary design stage, when the amount of information on future helicopter is still insufficient and many of the basic parameters are only under selection, it is advisable to use simple empirical formulas.

On the subsequent more detailed study of the project, specifying the basic parameters and the flight characteristics of the helicopter it is advisable to use semi-empirical formulas for calculating the weights, reflecting theoretically proved relationship between parameters and weights of units, the basic conditions and restrictions in the actual design.

The final exact weights of assemblies and systems, and the weight of empty helicopter along with them are determined at the detailed design stage by calculating

the weights of individual parts.

At all design stages in accordance with the adopted classification the helicopter empty weight is determined as

$$m_{emp} = m_{a.fr} + m_{pp} + m_{eq}. \quad (6.6)$$

In its turn airframe weight

$$m_{a.fr} = m_f + m_w + m_{tu} + m_{LG} + m_{ctl}. \quad (6.7)$$

Power plant weight

$$m_{pp} = m_{eng} + m_{tr} + m_r + m_{fs} + m_{os} + m_{co} + m_{sp}. \quad (6.8)$$

Equipment weight

$$m_{eq} = m_{n.r} + m_{t.eq}. \quad (6.9)$$

Main and tail rotors. Weight of main rotors include weight of the rotor blades and hubs with fasteners and dampers. *Rotors weight* in kg at the early design stages in the first approximation can be determined from the empirical relationships proposed by V.B. Barshevsky:

$$m_{MR} = 6.2D^{2.6}\sigma \text{ when D = 5.8 ... 22 m;}$$

$$m_{MR} = 2D^{3}\sigma \text{ when D = 18 ... 35 m,} \quad (6.10)$$

where D - rotor diameter, m; σ – rotor solidity ratio.

For a more precise determination of the rotor weights at the subsequent design stages you must calculate the weights of the blades and rotor hub separately. In this case, it is advisable to use a semi-empirical formula given by A.V. Nekrasov [2].

The *total weight of the blades* is then determined by the formula

$$\sum m_{bl} = k_{bl}\sigma R^3, \quad (6.11)$$

where $k_{bl} = c_y^{\alpha}\rho_0\pi / 2k_1\gamma_0$, c_y^{α}; – derivative of the lift coefficient on the angle-of-attack of the blade airfoil; ρ – density of air at height H = 0 by ISA; γ_0 – blade weight characteristic; $\gamma_0 = c_y^{\alpha}\rho_0 b_{0.7}R^4 / 2I_{fl.h}$; $b_{0.7}$ – blade chord at relative radius $\bar{r}_0 = 0.7$; $I_{fl.h}$ – blade moment of inertia relative to the axis of the flapping hinge; R – rotor radius;

Included in the formula (6.11) weight factors k_{bl} depend on the rotor radius R and for a number of helicopters are as shown in Fig. 6.4. The shaded area corresponds to the best blade designs being in operation as per weight.

Constructive-technical and strength limitations necessitate introduction of the *minimum possible value of the weight factor* $(k_{bl})_{min}$. Then formula (6.11) should be added with

$$k_{bl} \geq (k_{bl})_{min}. \quad (6.12)$$

For modern rotor blades at the maximum permissible values of the weight characteristic $(\gamma_0)_{max} \cong 7.0$ parameter $(k_{bl})_{min} = 5.5$. Comparative analysis of different types of blade design must take into account several additional factors (blade aspect ratio, structure type, material etc.). The formulas that determine the blade structure weights in this case are more complex. Weight of blades

$$\sum m_{bl} = k^*_{bl} \sigma R^{2.7} / (\bar{\lambda})^{0.7}, \quad (6.13)$$

where $\bar{\lambda} = \lambda / \lambda_{av}$; λ – blade aspect ratio; $\lambda = R / b$; λ =18.

The weight factors of the blades being the best with regard to weight regardless of the rotor diameter k^*_{bl} = 12.65...13.8.

Rotor hub weight m_{hub} is determined depending on the centrifugal force of the blades, number of blades and reached design level. These factors are reflected in the formula proposed by M.A. Leykand:

$$m_{hub} = k_{hub} k_z z_{bl} (N_{bl})^{1.5}, \quad (6.14)$$

where N_{bl} – blade centrifugal force; k_{hub} – hub weight factor taken from statistics; z_{bl} –number of MR blades; k_z – coefficient accounting additional hub weight increase due to increased number of blades;

k_z = 1 at $z_{bl} \leq 4$;

$k_z = 1 + \xi_z(z - 4)$ at z_{bl} > 4.

It could be accepted that coefficient ξ_z = 0.05, although in some cases it may be less. The weight factors k_{hub} for different hubs of domestic and foreign helicopters are shown in Fig. 6.5. The shaded area corresponds to the best articulated hubs being in operation with regard to weight.

Weights of single-rotor helicopter tail rotor blades and hubs are subject to roughly the same laws as the weights of these elements of the main rotors. But since the tail rotors operate in a much more severe conditions than the main rotors, the maximum weight characteristic of the tail rotor blade (index TR) usually $(\gamma_0)_{max} \leq 3$. This value $(\gamma_0)_{max}$ corresponds to $(k_{bl.TR})_{min}$ = 13.

Analysis of blades and hubs of helicopter tail rotors being in operation shows that the best examples of tail rotor blades and hubs structures have the same weight factor as those of the main rotors. This means that formulas (6.13) and (6.14) are valid in a wide range of parameters, including both steering and main rotors.

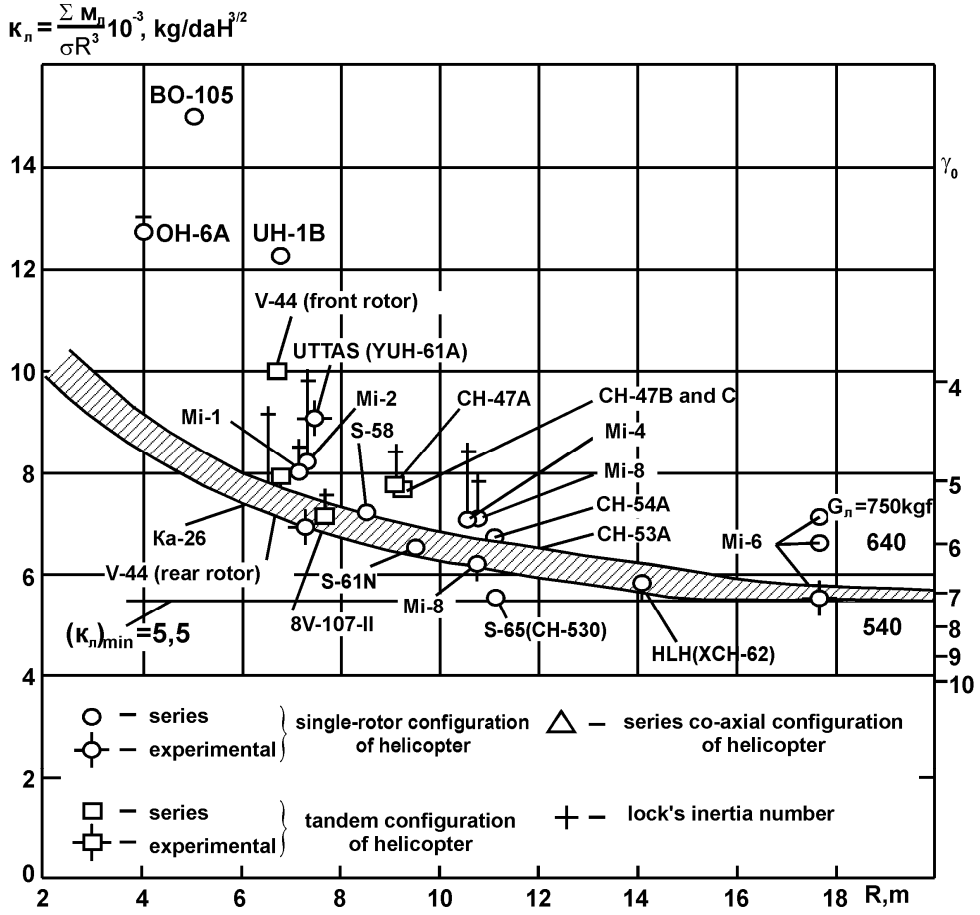


Fig. 6.4. Variation of Blade Weight Factors k_{bl} and Weight Characteristics γ_0, Depending on Rotor Radius R

Helicopter control. The control system includes swash plate assembly, control lever, pedals and collective pitch control stick in the flight cabin, the control linkage from the control lever, pedals and control stick to boosters, booster control system for main and tail rotors.

In the first approximation, the control system weight can be determined depending on the take-off weight and diameter of the rotor according to the formula proposed by G.K. Zhustrin [1]:

$$m_{ctl} = c_{1ctl} m_0 + c_{2ctl} D_{MR} + c_{3ctl}, \qquad (6.16)$$

where, c_{1ctl} = 0.0107 c_{2ctl} = 2; c_{3ctl} = 0.025 in accordance with the average statistics data for single-rotor helicopters; for a double-rotor helicopter of tandem configuration c_{3ctl} = 3.3; for single control c_{3ctl} = 8; for double control c_{3ctl} = 20.

To clarify the helicopter control system weight it should be considered a weight of its two component parts: booster control system m_{boost} and control linkage system

m_{link} from levers or pedals to the boosters [2]:

$$m_{ctl} = m_{boost} + m_{link}. \tag{6.17}$$

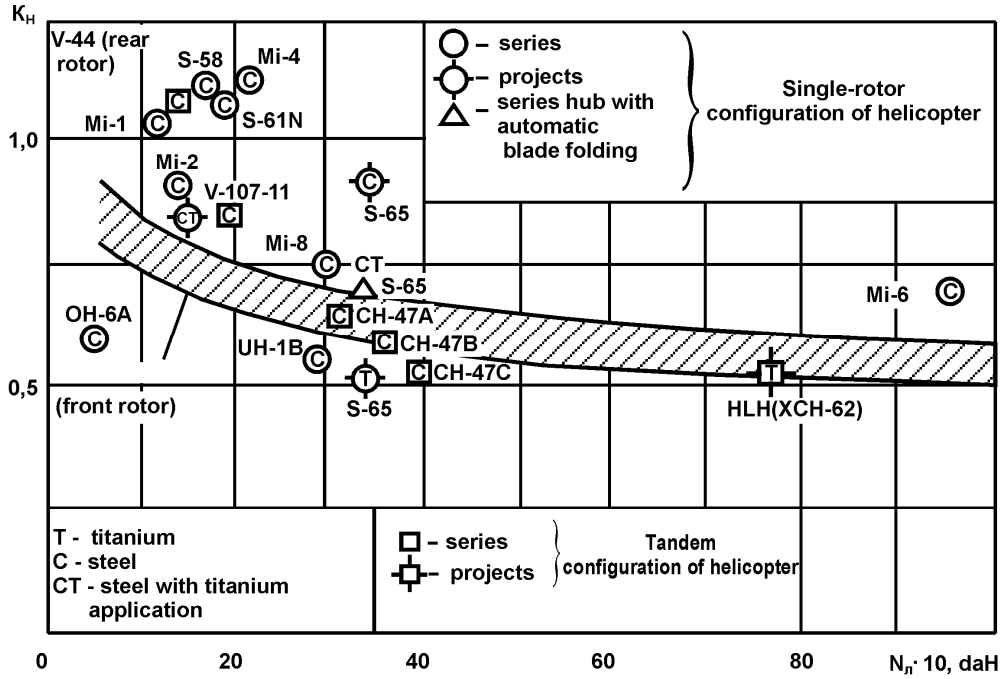


Fig. 6.5. Nature of Change of Rotor Hub Weight Factors k_{hub} Within Wide Range of Blade Centrifugal Force N_{bl}

The *booster control system*, which includes the swash plate, main boosters with their attachment systems, the control system from these boosters to the swash plate and main hydraulic system, is calculated for loads created by the rotor blades, and therefore it could be assumed that its weight is proportional to the sum of blade hinge moments:

$$m_{boost} = k_{boost} z_{bl} b^2 R, \tag{6.18}$$

Where z_{bl} - number of blades; b - the blade chord; k_{boost} - the booster control weight factor.

The weight analysis of a number of modern helicopters reveals that the achieved weight level of booster control can be assessed by the values k_{boost} = 16...19 kg/m^3. In view of further improving the booster characteristics the weight factor may be reduced to k_{boost} = 13...14 kg/m^3.

Manual control linkage weighs approximately 20 ... 30% of the control system total weight and depends almost exclusively on its total length and can be defined for single-rotor helicopter configuration in proportion to the blade radius R:

$$m_{link} = k_{link} R . \quad (6.19)$$

For helicopters that do not have auxiliary control systems for opening freight shutters, ladders, cowlings and extending landing gear, the weight factor can be taken k_{link} = 7...10.5 kg/m^3. For helicopters having auxiliary control system k_{link} = 18...25 kg/m^3.

For double-rotor helicopter the linkage length is greatly longer. Assuming that it is proportional to the distance L between the rotors, we get

$$m_{link} = k_{link} L . \quad (6.20)$$

With that value of k_{link} for tandem helicopter will be slightly less than of the side-by-side aircraft as total length of control linkage of the tandem configuration is shorter compared with the side-by-side one. For tandem configuration it could be accepted that k_{link} = 30...31 kg/m^3, and for side-by-side configuration k_{link} = 35...36 kg/m^3.

Helicopter Transmission. The transmission weight m_{tr} includes the weights of all reducers, shafts, couplings, shaft bearings, main rotor brake, as well as the oil filled in the lubricating gear system.

In the first approximation, the transmission weight can be determined depending on the shaft torque M_{trq} by the formula

$$m_{tr} = 0.48 M_{trq}^{0.83}, \quad (6.21)$$

$$M_{trq} = \frac{5 l \xi \sum N_{tr} D_{MR}}{z_r \omega R}, \quad (6.22)$$

where ΣN_{tr} – maximum engine power allowed by transmission, kW; ξ – power use rate; z_r – number of rotors.

For a more precise definition of transmission unit weights it could be calculated the weights of the individual transmission units using formulas given by A.V. Nekrasov [2].

Weight of the main reducer (with fasteners and oil)

$$m_{m.r.} = k_{m.r.} z_r (a_{trq} M_{trq})^{0.8}, \quad (6.23)$$

where $k_{m.r.}$ – weight factor; a_{trq} - coefficient of uneven torque distribution between the rotors; a_{trq} = 1.15 for tandem configuration; a_{trq} = 1.0 for other configurations.

The weight factors for a large number of built main reducers vary within wide

limits; $k_{m.r.}$ = 0.34 ... 0.525. In the calculations, reflecting the average achieved level, you can take $k_{m.r.}$ = 0.465. The rotor shaft torque can be determined from formula (6.22).

Weight of intermediate reducers

$$m_{in.r.} = k_{in.r.} z_{in.r.} (a_{trq} M_{eq})^{0.8}, \quad (6.24)$$

where $z_{in.r.}$ – number of intermediate reducers; M_{eq} – equivalent torque; $k_{in.r.}$ – weight factor; for modern helicopters it is accepted $k_{in.r.}$ = 0.7...0.9.

Equivalent torque on the intermediate reducer of a single-rotor helicopter

$$M_{eq} = 974 N_{TR} / n_r, \quad (6.25)$$

For double-rotor helicopter

$$M_{eq} = \frac{974 \sum N_{eng}}{z_{dr.sh} z_{sh} n_{sh} \alpha}, \quad (6.26)$$

where N_{TR} – maximum power transmitted to tail rotor, kW; n_{sh} – transmission tail shaft rotation frequency; ΣN_{eng} – total maximum engine power transmitted; z_{sh} – number of synchronizing shafts; α – coefficient depending on transmission configuration; α = 1, where reducer transmits power of all engines, α = 2, where it transmits power of a half of engine number.

Weight of tail reducer (with oil)

$$m_{t.r.} = k_{t.r.} M_{trq}^{0.8}, \quad (6.27)$$

where M_{trq} is determined by the formula (6.22), and weight coefficient $k_{t.r.}$ = 0.65...0.8.

Weight of transmission shafts

$$m_{sh} = k_{sh} L_{sh} (M_{trq})_{brk}^{0.8}, \quad (6.28)$$

where L_{sh} – shaft length, m; $(M_{trq})_{brk}$ – breaking torque; k_{sh} – weight factor.

Breaking moment for transmission shafts of single-rotor helicopter may be accepted equal to $(M_{trq})_{brk}$ = (2.2...2.6)M_{trq}, in hovering at static ceiling, and for double-rotor helicopters

$$(M_{trq})_{brk} = 974 \frac{n_{op} f \sum N_{eng}}{2 n_r}, \quad (6.29)$$

where f - safety factor; f = 1.5; n_{op} - operational overload; n_{op} = 1.8 ... 2.2, if all the

engines are located in one place, and n_{op} = 0.8 ... 1.2, if the engines are located in two groups and the rotors are separated.

The weight factors of transmission shafts may be taken k_{sh} = 0.06...0.085. Higher values of k_{sh} belong to helicopters of lower weight classes, as for technological reasons, the ratio of the transmission shafts sizes differs from the optimal one (towards their weighting).

Power plant. The propulsion system weight consists of engine weight m_{eng}, propulsion system weight m_{ps} and fuel system weight m_{fs}:

$$m_{pp} = m_{eng} + m_{ps} + m_{fs}. \tag{6.30}$$

Propulsion system weight at the preliminary design stage can be determined from the expression

$$m_{ps} = 1.835\gamma\sum N_{t-off}, \tag{6.31}$$

where γ – engine specific weight, kg/kW; ΣN_{t-off} – engine take-off power, kW.

Engine specific weight γ of different turboshaft engines depending on their take-off rating ΣN_{t-off} is given in Fig. 6.6.

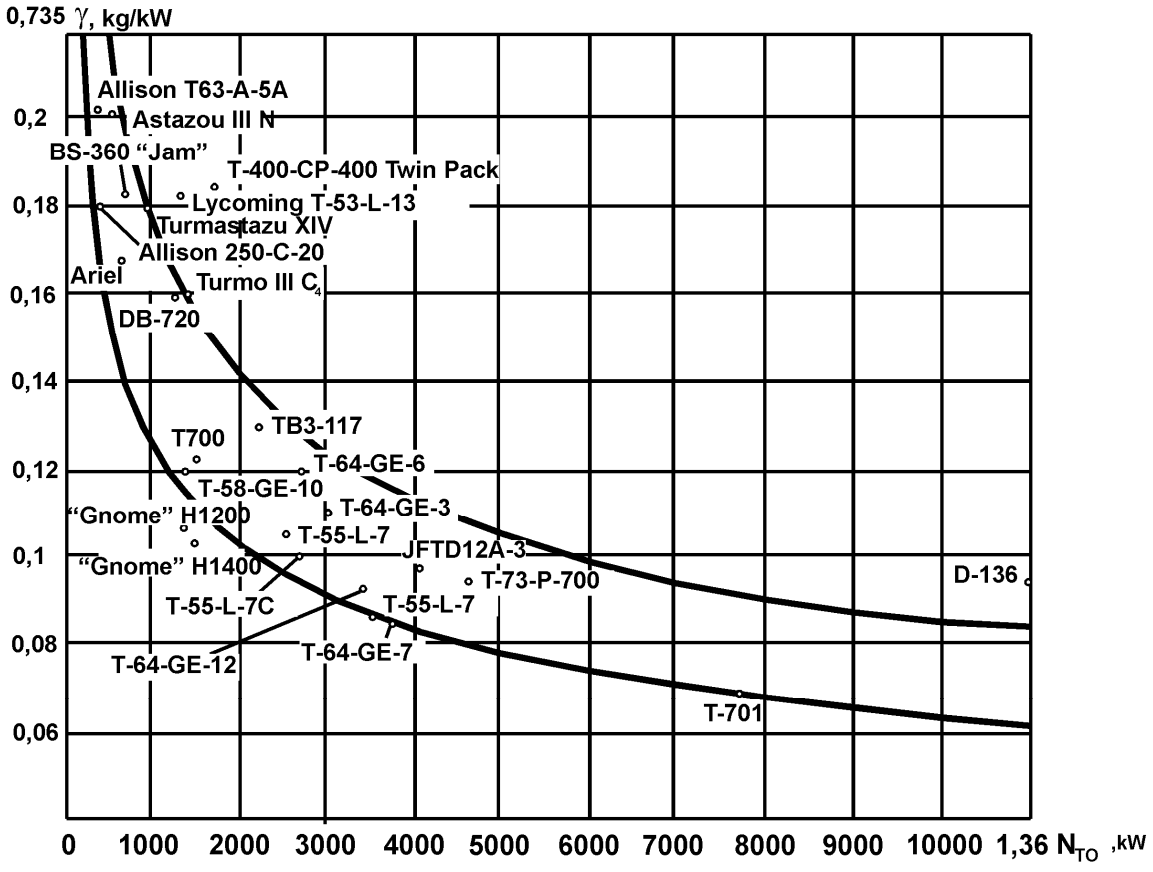


Fig. 6.6. Engine Specific Weight vs Take-off Power

You can also use the formulas obtained on the basis of processing the statistical data:

$$\gamma = 2.02 / N_{t-off}^{0.356} \text{ at } N_{t\text{-}off} < 3000 \text{ kW};$$

$$\gamma = N_{t-off}^{0.17} / 30.34 \text{ at } N_{t\text{-}off} > 3000 \text{ kW}.$$

A more accurate value of the propulsion system weight can be obtained by individual calculating of weights. If the design process is executed for the given engine, then the engine weight is already known. If the engine is selected based on the needs of required power, then the engine weight is determined depending on the maximum power:

$$m_{eng} = k_{eng} (N_{max})_{red}^{0.7}, \quad (6.32)$$

where $(N_{max})_{red}$ – engine reduced maximum power – the engine power take-off height H = 0 m at ISA.

The weight factor for the perfect modern engines can be taken k_{eng} = 1...1.2, which roughly corresponds to the specific weight of medium and high power engines with γ = 0.09...0.1.

The propulsion systems weight can also be expressed in terms of the reduced total power of the engines installed on the helicopter in the form of

$$m_{ps} = k_{ps} \sum N_{red}. \quad (6.33)$$

Weight factor of the propulsion systems can be accepted k_{ps} = 0.04...0.05.

The *fuel system weight* is determined depending on the total fuel amount taken onboard:

$$m_{fs} = k_{fs} (m_{fuel})_{tot}. \quad (6.34)$$

For the fuel system with self-sealing fuel tanks weight factor k_{fs} = 0.07...0.09. For systems without self-sealing tanks it is reduced to k_{fs} = 0.06...0.07, while the use of pressurized compartments, which weight generally refers to the airframe weight can result in reduced fuel system weight factors to k_{fs} = 0.035...0.04.

Thus propulsion system weight together with systems

$$m_{ps} = \gamma_{ps} (\sum N_{max})_{red} + k_{fs} (m_{fuel})_{tot}, \quad (6.35)$$

where specific gravity of propulsion system

$$\gamma_{ps} = k_{pss} + \frac{k_{eng}}{(N_{max})_{red}^{0.3}}.$$

Helicopter airframe. The helicopter airframe weight as already noted includes weights of fuselage, wing, tail unit, hoods and landing gear. All of these elements in one

way or another depend on the helicopter take-off weight. Therefore, in the first approximation, the weight of the helicopter airframe can be found using the formula proposed by V.B. Barshevsky to determine the weights of the body:

$$m_{body} = k_{body} m_0, \tag{6.36}$$

where k_{body} = 0.28 for transport helicopters of single-rotor design; k_{body} = 0.38 for amphibious helicopters; k_{body} = 0.23 for helicopter-cranes.

It should take into account that body weight includes weight of fuselage, landing gears, tail unit, tail rotor, as well as the control system and equipment.

For more accurate determination of the airframe weight one should calculate the weights of its individual elements:

$$m_{a.fr} = m_{fus} + m_{LG} + m_w + m_{t.u}. \tag{6.37}$$

The *fuselage weight* can be determined depending on the helicopter takeoff weight m_0, the fuselage outer surface area S_{fus} and linear fuselage dimension L_{fus}:

$$m_{fus} = k_{fus} m_0^{0.25} S_{fus}^{0.88} L_{fus}^{\alpha}. \tag{6.38}$$

As a linear fuselage dimension L_{fus} defining the bending moments acting upon the fuselage, for single-rotor and tandem configurations the distance between the rotors is adopted, and for side-by-side configurations it is the distance from the line connecting the axis of the rotor with the point of forces application on the control surfaces. It is recommended to take an exponent α = 0.16 for a single-rotor and side-by-side and α = 0.19 for tandem helicopter configurations.

The fuselage outer surface area of the single-rotor design transport helicopter with a sufficient degree of accuracy can be determined according to the formula proposed by V.P. Petruchik:

$$\begin{aligned} S_{fus} &= 4.34 H_{fus}(H_{fus} + B_{fus}) + 1.1R(0.5R - B_{fus}) + \\ &+ 1.25 R_{hd}(R - 0.5L_w - 1.4H_f) + 1.32 R_{TR}^2 + 0.13 N_{t-off}^{0.55}, \end{aligned} \tag{6.39}$$

where H_{fus} – fuselage height; B_{fus} – fuselage width; R – radius of the main rotor; R_{TR} - radius of the tail rotor; L_c - cargo compartment length.

For crane helicopter

$$S_{fus} = 0.088 R^2 + 1.32 R_{TR}^2 + 7.88R + 0.13 N_{t-off}^{0.55} + (8 + \frac{0.68 m_0}{1000}). \tag{6.40}$$

For modern transport helicopters weight coefficient is accepted equal k_{fus} = 1.3...1.6.

Fuselage weight calculated by formula (6.38), for helicopters with engines not

placed in the separate nacelles includes weight of power plant cowlings m_{cowl}. As in helicopters of side-by-side configuration the power plant is fully separated from fuselage then weight of cowlings should be calculated separately using the formula

$$m_{cowl} = 2k_{cowl}(N_{eng}/2)^{\frac{2}{3}}. \quad (6.41)$$

It is recommended to accept k_{cowl} =0.9.

Weight of helicopter wing for single-rotor aircraft depends on aerodynamic forces acting upon the wing and its dimensions: chord, span, airfoil thickness ratio. These factors account along with the wing weight the component not taking external loads:

$$m_w = \frac{k_w(\lambda_w S_w)^{3/2} v_{des}^2}{\bar{c}_w} + q_w S_w. \quad (6.42)$$

Wing weight of side-by-side helicopter unlike single-rotor helicopter is defined by the requirements of the wing relative stiffness with the purpose to prevent self-oscillations of the "ground resonance" type of the main rotors at elastic base (wing).

In relation to helicopter parameters the wing weight is defined by the requirements of stiffness either in bending or in torsion.

At $L_w/i>(L_w/i)_0$ wing weight is defined by stiffness in bending

$$m'_w = \frac{k'_w L_w^4 m_n}{\bar{c}_w S_w^2 (1+\frac{2m_n}{m_{mid}})}; \quad (6.43)$$

At $L_w/i<(L_w/i)_0$ – stiffness in torsion

$$m''_w = \frac{k''_w L_w^2 i^2 m_n}{\bar{c}_w S_w^2}, \quad (6.44)$$

where L_w – wing span equal to distance between main rotors, if they are placed at the wing tips; i – nacelle inertia radius; $\bar{c}_w$ – wing airfoil height ratio; S_w – wing area; m_n – nacelle weight including weights of main rotor blades and hub, main and intermediate reducer, propulsion system, control system, cowlings, reducer attachment units, electrical equipment; m_{mid} – weight of all units, components and cargo transported in the helicopter mid portion (of fuselage);

$$(\frac{L_w}{i})_0 = \frac{\sqrt{k''_w[1+(\frac{2m_n}{m_{mid}})]}}{k''_w \bar{c}_w}. \quad (6.45)$$

In calculation the following weight coefficients may be taken: k'_w = 0.0091 and k''_w = 0.191.

Weight of tail unit in general may be determined in relation to specific weight of 1m^2 of tail unit q_{tu}:

$$m_{tu} = q_{tu} S_{tu}, \tag{6.46}$$

where S_{tu} – helicopter tail unit area, m^2.

As per statistics data the tail unit specific weight lies within rather wide range (q_{tu} = 5.6...12.4 kg/m^2), and tail unit area at the initial stage of design is not determined then in practice use empirical function relating tail unit weight with helicopter take-off weight:

$$m_{tu} = k_{tu} m_0, \tag{6.47}$$

where k_{tu} = 0.00136 for single-rotor helicopter stabilizer; k_{tu} = 0.0076 for tandem and coaxial helicopters; k_{tu} = 0.0125 for tail unit of airplane type used on side-by-side configurations.

Helicopter landing gear weight is also often determined in relation to helicopter weight:

$$m_{LG} = k_{LG} m_0, \tag{6.48}$$

where k_{LG} – averaged statistics weight factor; k_{LG} = 0.025...0.028 for single-rotor configuration; k_{LG} = 0.031...0.035 for tandem configuration ; k_{LG} = 0.01...0.015 for helicopters with skid-type landing gear.

When determining weight of landing gear consider helicopter and landing gear configurations, as helicopter weight reduced to LG strut and force acting upon it depends on above mentioned configurations. More over consider LG structural features: ability to retract, availability of brake, tire type etc. For example weight of retractable landing gear is in average exceeds weight of non-retractable landing gear by 15...20 %.

Weight of tail bump of single-rotor helicopter as a rule does not exceed 0.1 % of helicopter take-off weight.

Helicopter equipment. It is reasonable to determine weight of all equipment as a sum of weights of electrical equipment and miscellaneous equipment:

$$m_{eq} = m_{el.eq} + m_{misc}. \tag{6.49}$$

Electrical equipment weight

$$m_{el.eq} = k_{wire} L_{wire} + k_{el.eq.} F_{bl}, \tag{6.50}$$

where L_{wire} – electric wiring length, m; F_{bl} Fl - main rotor blade area, m^2.

For single-rotor helicopter electric wiring length can be set equal to the rotor

radius (L_{wire} = R), and for double-rotor helicopter it is set equal to the distance between the rotors. The weight factor k_{wire} = 22...24 kg/m for medium single-rotor helicopters; k_{wire} = 35...40 kg/m for double-rotor helicopter; k_{wire} = 10...12 kg/m for light helicopters; $k_{el.eq.}$ = 5... 6 kg/m^2 for medium helicopters of any configuration; $k_{el.eq.}$ = 12...16 kg/m^2 for light helicopters.

The miscellaneous equipment weight

$$m_{misc} = k_{eq} m_0^{0.6}. \tag{6.51}$$

The weight factor k_{eq} ranges from $(k_{eq})_{min}$ = 1.6 to $(k_{eq})_{max}$ = 2.65. The value of $(k_{eq})_{min}$ is determined by composition of utility equipment, and $(k_{eq})_{max}$ also includes removable mission equipment.

6.4. CALCULATION OF HELICOPTER TAKEOFF WEIGHT

Calculation of the helicopter take-off weight in the second and subsequent approximations, as the analysis of weight formulas, can be carried out only after selection of the helicopter parameters, estimation of the demand power capacity, selection of the engine and determination of its specific characteristics. An important place in this calculation a definite needs fuel supply takes. When calculating the weights of fuel to fly a predetermined distance for a transport helicopter it is necessary to know the flight altitude, cruising speed and approximate typical flight profile (model production operations), which allows to determine time of each flight phase and the engine operating condition.

Needs fuel supply

$$m_{fuel} = 1.12 \sum_{i=1}^{n} N_i c_{ei} T_i, \tag{6.52}$$

$$m_{fuel} = 1.12 N_{t-off} c_{et-off} T_{fl} \sum_{i=1}^{n} \overline{N}_i \overline{c}_{ei} \overline{T}_i,$$

where N_{t-off} – total engine take-off power, kW; c_{et-off} – specific fuel consumption at takeoff, kg/(kWh); T_{fl} – flight time, h; n – number of elements of production operation (stages of flight); $\overline{N}_i$ – degree of throttling engines during the i-th stage of the flight; $\overline{N}_i = N_i / N_{t-off}$ $\overline{c}_{ei}$ – throttle fuel consumption characteristics; $\overline{T}_i$ – relative time of the flight i-th stage.

The coefficient of 1.12 takes into account the start-up and testing of engines,

steering, 5% fuel reserve for navigation and others.

A typical profile of a transport helicopter flight includes:

- takeoff and controlled hovering;
- climb and level flight;
- en-route flight;
- descent, hovering and landing.

Total flight time is defined as the sum of time periods needed for the individual phases of flight:

$$T_{fl} = T_1 + T_2 + T_3 + T_4, \tag{6.53}$$

where T_1 - time for take-off and landing; T_1 = 0.015...0.025 hours; $T_2 = H_{l.fl} / v_{y.cl.3600}$ – time for climb and reaching level flight, h; $T_2 = L / v_{cr}$ - time of level flight, h; $T_4 = H_{l.fl} / v_{y.des.3600}$ – time for descending to hovering altitude, h; $H_{l.fl}$ – altitude of level flight, m; L – level flight distance (accounting path for climb and descend), km; v_{cr} – cruising speed, km/h; $v_{y.cl.}$ – helicopter climb rate, m/s; $v_{y.des.}$ – descent vertical speed of the helicopter, m/s.

In the absence of the engine specific flow characteristics you can use the following empirical dependence:

$$c_{e.t-off} = 0.545 / N_{eng}^{0.081} \text{ at } N_{eng} > 3000 \text{ kW}; \tag{6.54}$$

$$c_{e.t-off} = 1.067 / N_{eng}^{0.167} \text{ at } N_{eng} \le 3000 \text{ kW};$$

And proposed by A.V. Nekrasov

$$c_{e.t-off} = k_{ce} / N_{t-off}^{0.1}, \tag{6.55}$$

where k_{ce} = 0.620...0.686.

Larger values of k_{ce} are for less powerful engines and smaller - for heavy-duty engines.

Change in fuel consumption, depending on the degree of the engine throttling can be approximately is determined by using the relation

$$c_{eN} = \left[c_{e.t-off} - 0.16(1 - \bar{N}) \right] / \bar{N}, \tag{6.56}$$

where $\bar{N} = N_{l.fl} / N_{t-off}$, $N_{l.fl}$ – power required for level flight at cruising speed.

It should be borne in mind that calculation of the required power for level flight must be maintained for helicopter average take-off weight, which can be approximately determined, given thus change helicopter flight weights due to fuel combustion.

$$m_{0av} = m_0 - 0.5 m_{fuel}. \tag{6.57}$$

The take-off weights of the second approximation being a sum of the weights of empty helicopter m_{emp}, fuel m_{fuel}, payload m_{pl}, crew m_{crew} and equipment m_{eq}:

$$(m_0)_{ii} = 1.1m_{emp} + m_{fuel} + m_{crew} + m_{pl}. \quad (6.58)$$

The coefficient of 1.1 at m_{emp} takes into account margin for the construction overweight. The process of determining take-off weight is an iterative and at each iteration step you have to find the relative change in take-off weight from the previous step, and compare it with a given accuracy of calculation, i.e., check condition

$$\frac{\left[(m_0)_i - (m_0)_{i-1}\right]}{(m_0)_{i-1}} \le \varepsilon. \quad (6.59)$$

where ε - specified accuracy of calculating take-off weight.

Parameter ε is different for different stages of the helicopter design. It defines above all the degree of elaboration of the helicopter project, accuracy of the basic data, level of aerodynamics, strength and weight calculations, carried out at each design stage: at the stage of development of the technical proposal or technical project ε = 8 ... 10%, in the development of conceptual design ε = 3 ... 4%, in the detailed design ε = 1 ... 2%.

When calculating the helicopter take-off weight you should always born in mind that, as a rule, the real weight of the empty helicopter exceeds its estimated value. There are many objective reasons for this phenomenon (insufficient study of the structure and its subsequent refinement in the process of static, dynamic and flight tests, additional requirements of the customer), and designer challenge is to properly take this into account and avoid exceeding margins put in the calculation to prevent of overweight design by an average of 10% at most.

For this purpose, at all design stages try to develop weight limits for each helicopter unit, which is permanently adjusted and refined in the process of the helicopter project structural study. Only thorough analysis of the design, introduction of new designs, materials and technologies would provide meeting the challenges of continuous helicopter weight perfection.

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Chapter 7. DEFINITION OF HELICOPTER LAY-OUT AND CENTER OF GRAVITY POSITION

Definition of the helicopter lay-out is the final stage of the project development and means spatial linking of all helicopter parts.

The laying out process consists of simultaneously happening and being interrelated processes of defining the aerodynamic, volume-weight and load-carrying configurations.

Definition of aerodynamic configuration is commonly understood as selection of helicopter configuration, relative positioning of main and tail rotor, choice of parameters and arrangement of control surfaces, definition of airframe external shapes (fuselage, wings, tail), placing the air intake and exhaust systems of the power plant.

The task of the ***volume-weight layout*** includes relative positioning of helicopter individual units and assemblies, arrangement of cargo, passengers, crew and equipment and definition of the helicopter center of gravity position (CG) for various flight conditions.

Load-carrying layout provides spatial linking of helicopter load-carrying structures, defining of scheme of transferring loads from the helicopter basic units to the fuselage load-bearing components, technological breaks, access to individual units and equipment for helicopter storage etc.

The helicopter layout depends on many factors (helicopter purpose, its configuration, type and number of engines), which are mutually related, and therefore, as a rule, the laying out process is interactive. For each layout option the helicopter CG position is calculated, since one among the layout tasks is to obtain a predetermined center of gravity position range. The basic requirements that must be met in the process of laying out are defined by the range of tasks to be solved during process of placing the individual units.

7.1. AERODYNAMIC CONFIGURATION

Tasks of making aerodynamic configuration

Making the helicopter aerodynamic configuration is primarily directed to provision of the basic aerodynamic requirements:

- Achievement of the highest lift-to-drag ratio;
- Obtaining maximum rotor thrust at the lowest consumed power;
- Ensuring necessary stability and controllability;
- Creating favorable conditions for power plant operation;
- Exclusion of possibility of critical flight conditions being life-threatening for pilots and passengers (stall, «vortex ring», flutter, «ground resonance» etc).

These requirements are satisfied by:

- Mutual arrangement of main and tail rotors;
- Choice of wing shape and it sizing and positioning on the fuselage;
- Sizing and locating of stabilizer;
- Selection of LG configuration and layout;
- Selection of fuselage external shapes and linking of helicopter individual parts;
- Placing fairings, fillets and other means of improving the aerodynamic flow around;
- Choice of shape of air intakes, air ducts and power plant exhaust systems.

All these problems are reflected in the development of the layout drawing consisting of the helicopter longitudinal side section and a number of cross-sections, individual views, and in some cases (for complex helicopter configurations) section in plan projection. If the engines are placed in separate nacelles outside the fuselage, then usually the nacelle arrangement drawing is done.

Relative position of main and tail rotors. The helicopter layout to a large extent is defined by the mutual position of the rotors, depending on the helicopter configuration.

For single-rotor helicopter an important factor in this connection is the choice of the tail rotor position. Maximum tail rotor thrust in hover with a view to ensuring helicopter yaw control under unfavorable wind direction should be at least

$$T_{TR} = 1.25 M_{reac} / L_{ax},$$

where M_{reac} – main rotor reactive torque;

L_{AX} – distance between axes of main and tail rotors, m; as a rule

$$L_{ax} = \frac{d + D}{2} + 0.5.$$

To exclude the fuselage roll during vertical takeoff and hovering, tail rotor axis of rotation is recommended to be placed in a plane passing through the center of the main rotor hub, perpendicular to its axis (Fig. 7.1).

Fig. 7.2 shows helicopter fuselage roll when main rotor hub is not placed in the main rotor plane of rotation. It is conventionally accepted that the helicopter center of mass (cm) lies on the main rotor axis and main rotor hub has no offset of flapping hinges.

Staggering of the tail rotor upward improves the helicopter lateral stability in level flight, which is an important fact for the operation and reduces the risk of touching the ground by tail rotor blades during helicopter landing or maneuvering at extremely low altitudes. Furthermore, this tail rotor arrangement reduces harmful influence of the

tapered vortex sheet upon aerodynamic loads on tail rotor blades in level flight conditions.

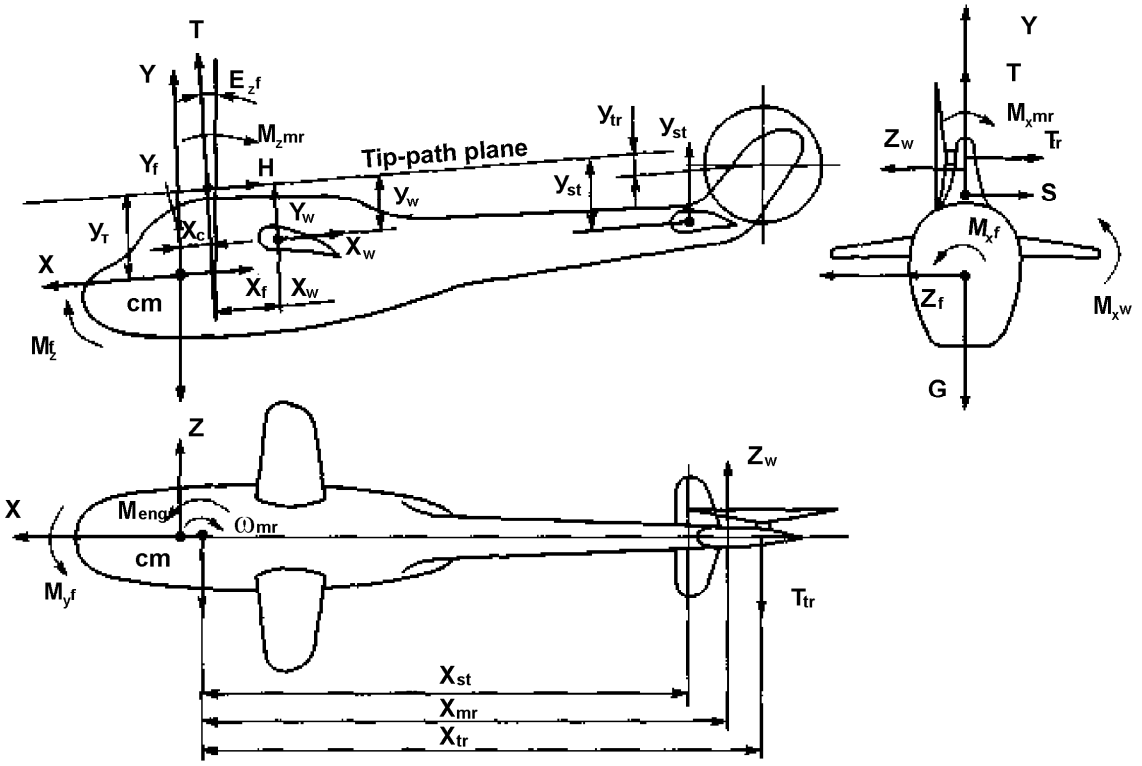


Fig. 7.1. Coordinate Systems and Loads Acting upon Helicopter

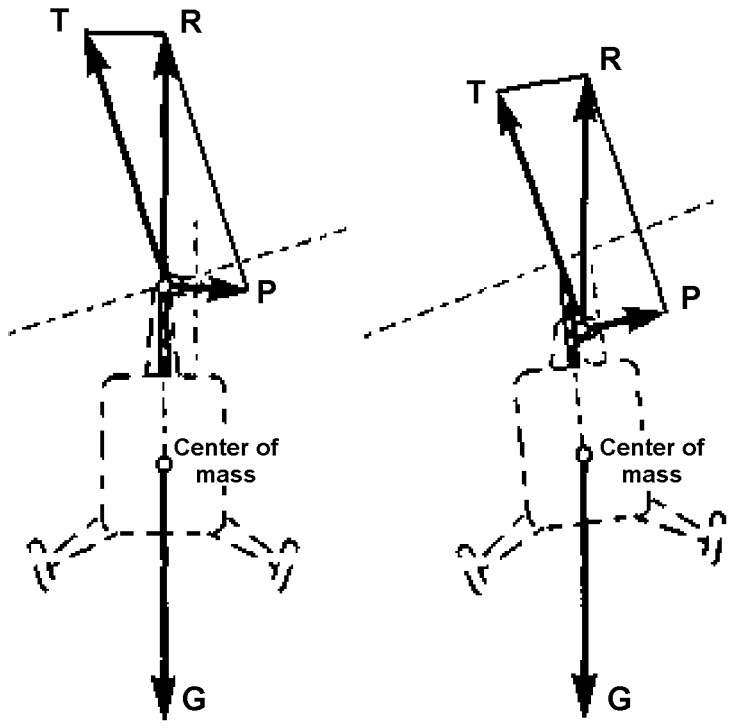


Fig. 7.2. Forces and Moments Acting on Helicopter

However placement of the tail rotor axis in the main rotor plane of rotation necessitates considerable elongation of tail boom or fin that causes weight growth for structures, growing drag, and other alignment deterioration. Therefore, in practice tail rotor axis is placed lower than main rotor hub. This causes the helicopter roll during takeoff and hover for 2...3° in the direction opposite to the main rotor thrust action.

During takeoff of the helicopter, having, for example, left direction of main rotor rotation (clockwise in plan view), at the lift-off moment first the left wheel separates from the ground, and at the beginning of vertical landing the right wheel first touches the ground.

On helicopters Mi-10 and Mi-24 cockpits are turned about the helicopter longitudinal axis with respect to the fuselage central part to the left (when viewed from behind) for 1.5...2.5°. Due to this fact the balancing roll angles measured at the cabin, are close to zero, and helicopter vertical take-off and landing are performed on both wheels simultaneously.

Sometimes when installing tail rotor its axis is deflected from the horizontal position (at helicopters CH-53E «Super Stadium» and the UH-60A «Black Hawk»), with the purpose that the tail rotor vertical thrust component creates pitching up or pitching down moment to improve the helicopter longitudinal balancing. Main rotor axis is not strictly perpendicular to the horizontal construction of the helicopter fuselage and tilted forward from the normal at an angle $E_{z\varphi}$ = 3...5° (ref. Fig. 7.1). This is necessary to ensure that in cruising flight the fuselage axis has to be directed along the flight path and therefore the fuselage would have the least drag.

Such tilt of main rotor axis also leads to reduction of bending stresses on the main rotor shaft in maximum continuous operation.

Distance from the blade tip and tail boom or the fuselage nose part must be such as to prevent any possibility of the blade impact against the structural members under any flight conditions, including maneuvering (enter into dive, and exit out of nose-up maneuver).

Position of main rotor hub above the fuselage is defined by the following factors (Fig. 7.3):

– angle of hub sleeve deflection down to the blade droop stop $(\sim 1.5...2°)$;

– static blade deflection ($\gamma_R \approx 0.12 \cdot R$);

– distance over vertical between the end of non-rotating blade and helicopter tail boom $(0.05 – 0.07) \cdot R$.

Safety spacing between the main rotor blades and airframe components thus should be at least 3°.

For ***double-rotor helicopter*** the relative position of the rotors is of considerable importance. On coaxial helicopters distance between the planes of the upper and lower

rotors must be of at least 0.1D, to avoid the collision («crossing») of rotor blades with a sharp gust of wind in hovering or when maneuvering in the event of helicopter sharp roll or pitch.

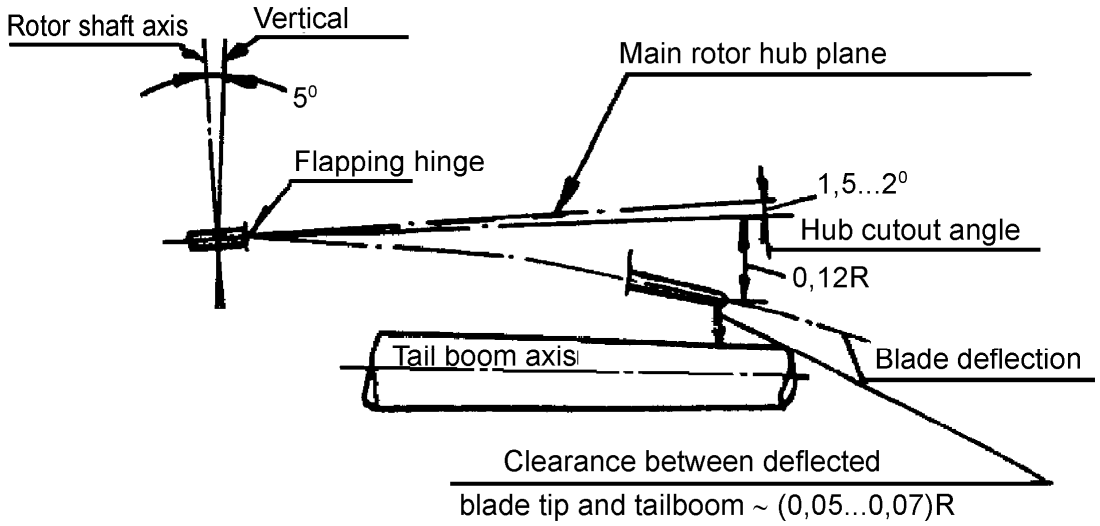


Fig. 7.3. Location of Main Rotor Hub Above Fuselage

Increasing the distance between the rotors for more than 0.1D is undesirable since it leads to weighting of carrier system and increases helicopter drag. To reduce the risk of «crossing» of the rotors blades it is recommended to use special dampers in coaxial helicopter control system to compensate sharp movement of the control lever. In practice, the role of such damper is played by hydraulic actuator used in almost all the helicopter control systems at the moment.

Main rotors in double-rotor helicopters of tandem configuration are also installed with stagger of the rear rotor above the front rotor by the (0.08 – 0.1)D (Fig. 7.4). This must be done to reduce the impact of the front rotor on the required power of the rear rotor in level flight conditions. In this case, the vortex sheet that comes from the forward rotor disc passes below the plane of the rear main rotor.

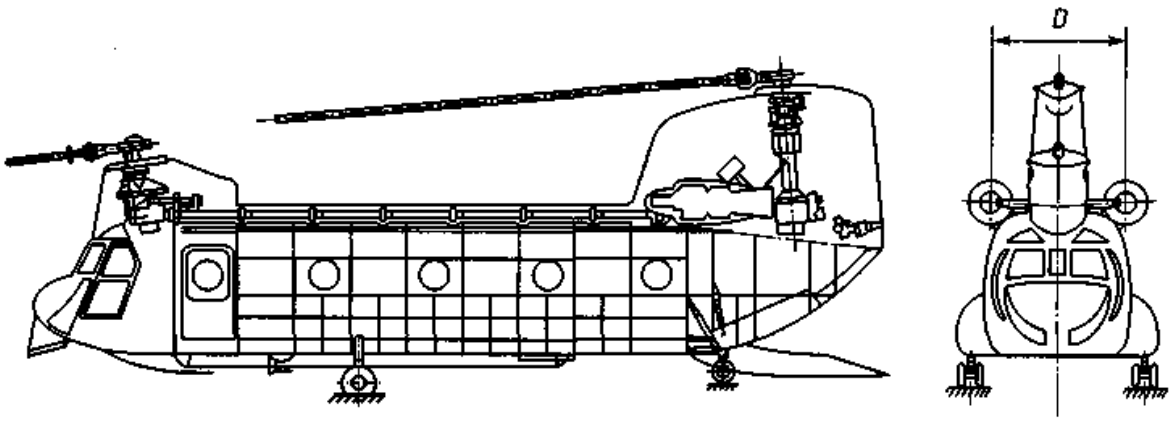


Fig. 7.4. Helicopter «Chinook»

Optimum tilt angle of rotor axes is to be selected to equalize the required power of the front and rear rotors under the most extensive flight conditions. At optimal speed of level flight close to the cruising speed, the optimal tilt angle is about 6°. Since to reduce the fuselage drag at cruising speed it is also recommended to tilt the axis of the main rotors forward by 3...4°, then total tilt of the front rotor axis is 9...10°, and tilt of the rear rotor axis is about 3...4°. The direction of rotation of rotors can be any, but mutually opposite. Main rotors of the double-rotor helicopter of the side-by-side configuration operate under the same conditions and, therefore, are on the same level. Both rotors are tilted forward by slight angle (3...4°) in order to reduce fuselage drag in cruising flight.

When designing a double-rotor helicopter of the tandem and side-by-side configuration it is important to shorten distance between the axes of the rotors, since it allows to make smaller fuselage or wing, and to increase the weight ratio. This can be achieved due to rotors overlap.

The amount of overlap $\bar{a} = a / R$ is limited by possibility of rotor blades to cross in the process of their counter movement in the rotor plane of rotation even if rotation synchronization is provided. This should take into account elastic vibrations of the blades relative to the drag hinges, torsion deformation of the synchronization shafts, gaps in the transmission gears, possible differences in rotor torques, nature of their changes with time, torsion deformation of wing panels of the side-by-side helicopters.

For helicopters of tandem configuration having stagger of the rear rotor plane of rotation relatively to the front rotor plane of rotation, crossing is possible when pulling the control lever, with that the front rotor taper angle increases, and the rear rotor taper angle decreases.

Maximum possible overlapping of rotors depends on the number of blades. It was found that when choosing double-rotor helicopter parameters with the general order of the hub hinges and synchronizing shaft frequency n_{shaft}<3000 min^{-1} the limit overlap of rotors should be $\bar{a} \leq 0.63$, $\bar{a}$ = 0.4 at z_{bl} = 4 and $\bar{a}$ = 0.17 at z_{bl} = 5. At $z_{bl} \geq 6$ overlap of the rotors is not permitted. To ensure safety flight with such overlap values you need to implement a set of measures to ensure necessary margins for crossing: appropriate offset of drag hinges, increased stiffness of transmission systems, availability of synchronizing rotors, limitation of range and speed of the control lever travel when controlling differential collective pitch of rotors.

Selection of parameters and placement of wings

Helicopter wing is designed to reduce loads acting upon main rotor at high flight speeds, and thus to increase maximum horizontal speed by distancing in time stall at main rotor. In addition, the wing is used for suspension of various equipment, accommodation of fuel tanks and retractable landing gear struts. Installation of wing, usually slightly improves helicopter balance and stability under level flight conditions.

However wing increases (up to 7%) weight of the helicopter structure, thus reducing the load ratio, and leads to additional losses in thrust for blowing about, notably critical under vertical flight conditions. Therefore, the feasibility of the wing installation must be decided on the basis of the helicopter purpose and nature of its missions.

The wing, which has typically thick symmetrical airfoil and trapezoidal shape (in plan), begins to carry, i.e. creates significant lift at flight speeds exceeding 100 km/h. When the flight speed is about 200 km/h wing already reduces loads acting upon main rotor by 10 ... 15%, and at speed of 300 km/h – by 20 ... 25%.

Wing setting angle (angle of wing installation relative to waterline) is selected on the basis of conditions to ensure the wing effective operation at high flight speeds, when the angle of the helicopter pitch reaches negative values. Its value can be determined as the sum

$$\varphi_{wing} = \alpha_{wing} - \vartheta + \Delta\alpha_{MR,}$$

where α_{wing} – wing angle-of-attack at maximum speed; ϑ – pitch angle dictated by helicopter longitudinal balancing; $\Delta\alpha_{MR}$ – downwash field caused by main rotor inductive speeds.

Since the pitch angle is largely dependent on the helicopter CG position, with the aft CG position the wing setting angle is less than with the forward CG position and shifting back center of mass leads to increase wing lift due to increasing α_{wing}.

Downwash flow from the main rotor inductive velocity field differs on the right and left wing outer panels due to the unevenness of the field as per main rotor azimuth. Maximum downwash occurs at azimuth of 90°. Therefore, angle-of-attack on the left wing panel for main rotor left rotation, and hence the lifting force on it will be smaller than on the right one. However, this fact enhances the helicopter lateral balance, as there is a roll moment, which counteracts the rolling moment of main rotor lateral force and reduces the need for balancing deflection of control stick to the left with increasing flight speed.

In order to increase the wing positive impact on the helicopter lateral balance the right wing panel angle of setting is increased by 1.5 ... 2° compared to the left panel.

To improve the helicopter longitudinal balancing the wing is usually placed in the fuselage central part with that 1/4 of wing mean aerodynamic chord (MAC) is aligned with the helicopter center of mass or the wing is positioned slightly behind it. In the first case, the lifting force arising on the wing, has no essential effect on the helicopter longitudinal balance, while in the second case it creates a dive moment, improving longitudinal static stability of the helicopter airframe in angle-of-attack, but at the same time, deteriorates it in the helicopter flight speed.

The wing has a significant effect on the airframe transverse static stability when helicopter flies with slip. Installing the wing slightly enhances the fuselage transverse stability, but begins to complicate piloting the helicopter. In some cases the wing is set with a negative angle of the cross-roll (see Fig. 7.1) to reduce the airframe lateral stability as is done, for example, on the Mi-24 helicopter.

To reduce the adverse aerodynamic interference between the wing and fuselage the joints are closed with fairings to provide smooth mating surfaces.

Selection of the parameters and placement of the tail unit. The helicopter tail unit is designed for longitudinal balancing in level flight as well as to provide helicopter longitudinal and directional stability. It consists of a fin and a stabilizer.

A ***stabilizer*** having sufficiently thick symmetrical airfoil and trapezoidal shape (in plan), improves characteristics of helicopter longitudinal balancing and stability. Two types of stabilizers are used: uncontrollable and controllable.

Controllable stabilizer is kinematically connected to main rotor collective pitch lever so that increase in main rotor collective pitch causes increase in stabilizer setting angle φ_{st}. With that longitudinal moment generated by the stabilizer increases in the direction of dive, opposing the main rotor pitching moment increment with an increase in collective pitch in forward flight.

Sizes, and therefore area of horizontal and vertical stabilizers are largely dependent on their location. For single-rotor helicopters stabilizer is usually mounted at the end of the tail boom in one of three positions *I-III*, shown in Fig. 7.5. The position *I* corresponds to hover and forward flight at low speed and stabilizer is in the fully air-blowing area of flow generated by main rotor. At the same time on the stabilizer, as well as on the wing, there is significant inductive downwash from main rotor, causing loss of main rotor thrust due to blowing about stabilizer in hover and reduces the stabilizer effectiveness in low-speed level flight.

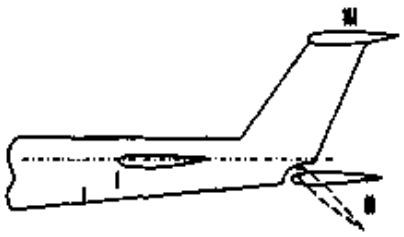


Fig. 7.5. Illustration of Possible Stabilizer Location
on Helicopter of Single-Rotor Design

In level flight in addition to inductive downwash from main rotor on the stabilizer a downwash is created by fuselage and wing. Due to significant total downwash the stabilizer angle-of-attack α_{stab} usually is negative, even if the stabilizer setting angle is positive $\varphi_{st} > 0°$. Due to this the stabilizer creates negative lift and, consequently,

pitching down moment relative to the helicopter center of mass. With the increase in level flight speed downwash from main rotor decreases, thereby reducing the pitching down moment, which is one of the causes of «spoon» - the characteristic of the profile balancing of single-rotor helicopters.

Installing the stabilizer in position ***I*** particularly adversely affects when helicopter hovers under action of back wind: the wind deflects the main rotor inductive flow forward and the stabilizer having a negative setting angle, creates the pitching down moment requiring an additional pulling the control stick for compensation. Therefore in position ***I*** stabilizer area is limited by the condition to ensure the required control margin when helicopter hovers with extreme forward CG position and back wind, and also because of the additional power loss of main rotor thrust on flow about stabilizer when hovering.

When set to position ***II*** in hover the stabilizer is outside the air flow from the main rotor and the wind behind it does not have a major impact. However, as soon as the helicopter starts to move forward, the air flow from main rotor being deflected backward, immediately gets on the stabilizer. Downwash inductive flow caused by main rotor at that leads to origination of strong pitching down moment, which compensation requires significant pulling of the control stick.

To reduce effect of main rotor inductive downwash on the stabilizer characteristics on some helicopters (UH-60A «Black Hawk», AH-64A «Apache») stabilizer is installed in the position ***II*** and made controllable in flight speed. In hovering and at low flight speed the setting angle φ_{st} = 30 ... 40 °. Inductive downwash from main rotor does not cause large pitching moment at the same time. As the flight speed grows the stabilizer setting angle is automatically reduced to a value ensuring the helicopter longitudinal balancing. In autorotation the stabilizer is set at negative angle.

The smallest impact of main rotor inductive downwash is observed on the stabilizer installed in position ***III***. But this option of the stabilizer installation results in, firstly, extra weight of tail unit, and secondly, that under level flight conditions, stabilizer is directly in the wake of the main rotor disturbed flow creating high variable loads on stabilizer.

Stabilizer area S_{st} is chosen mainly on the basis of requirements set as per helicopter longitudinal stability, which are defined by acceptable values of the helicopter longitudinal moment derivatives with respect to angle-of-attack and pitch angular velocity. Therefore, parameter S_{st} largely depends on the stabilizer's location, which defines its distance from the helicopter center of mass. In the single-rotor helicopter with a long tail boom the stabilizer area is much smaller than of co-axial configuration, and is typically (0.5 ... 1)% of the main rotor disc area. At some foreign helicopters (S-76, UH-60A, AH-64A, and others), It reaches values of 1.5 ... 2%. The coaxial helicopter layout does not permit an adequate stabilizer offset, so to ensure good longitudinal stability its area is enlarged.

Fin for single-rotor helicopter is optional. Moreover, during takeoff and landing, it even worsens helicopter flight data. But the need to install an end beam at the single-engine helicopters leads to feasibility of its implementation in the form of a fin, which partly reduces loads acting upon tail rotor at high speed of level flight and improves the helicopter directional stability.

For coaxial helicopter vertical tail made in a form of fins is the only means of ensuring directional stability. Because due to the compactness of coaxial-rotor helicopters distance from the fin to the helicopter center of mass is rather short, it is necessary to put a powerful double-fin vertical tail with rudders.

Fin, mounted on a single-rotor helicopter used instead of previously installed end beam has an asymmetrical thick profile, a trapezoidal shape, set at an angle of 30 ... 60° to the tail boom longitudinal axis and turned by an angle of 5 ... 7° relative to the vertical helicopter airframe plane of symmetry to the side of main rotor rotation (to the left in the left rotation direction). At the same time under the level flight conditions fin creates lateral aerodynamic force directed in the tail rotor thrust direction.

On coaxial helicopter fins are the only means of ensuring its directional stability, and rudders installed on them contribute to the improvement of the helicopter directional control in level flight. The coaxial helicopter as a rule is fitted with a powerful vertical tail, which includes two vertical plate fins with rudders, and some types of helicopters are fitted with additional upper and lower fins, mounted on the tail boom. Such vertical tail provides sufficient directional stability at helicopter small overall dimensions and greatly improves the helicopter directional stability in level flight, and under conditions of main rotor autorotation.

To improve the vertical tail efficiency when flying at high angles of slip when stall is possible on both fins at the same time, the plate fins of coaxial helicopter are turned inside by 15° relative to the vertical plane of symmetry.

The vertical stabilizer area on modern single-rotor helicopters is of (0.6 ... 1.5)% of main rotor disc area. For helicopters using multiblade rotor placed in the profiled duct – «fenestron» (SA-341 Gx, «Gazelle», SA-365N «Dofen»), instead of a conventional tail rotor the vertical tail area is increased to 2.5 ... 3.5 % of main rotor disc area. On coaxial helicopters the vertical tail area is of 2.5 ... 3.0% of the main rotor disc area.

Selection of fuselage external shapes. The fuselage shape is defined by helicopter configuration, layout, purpose and conditions of operation. In the layout of most modern Russian helicopters the engines are placed over the fuselage central part in front of MR reducer.

Combination of non-retractable landing gear and long tail boom of single-rotor helicopter fuselage gives bad streamline conditions that result mainly in higher drag and unfavorable moment characteristics. The coaxial helicopter fuselage features with smaller length, greater symmetry and compactness. Fuselage aerodynamic

characteristics defined by coefficients C_{xf} and C_{yf}, depends on the fuselage angle-of-attack (Fig. 7.6).

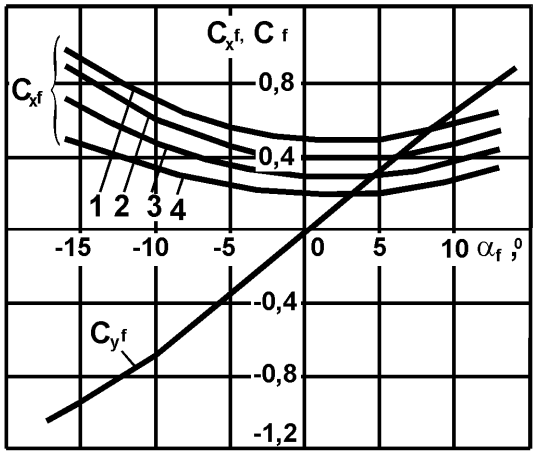


Fig. 7.6. Lift and Drag Coefficients vs Fuselage Angle-of-Attack α_f:

1 – hub; 2 – landing gear; 3 – other carrying members; 4 – fuselage

Drag coefficient is minimal within the fuselage angle-of-attack range $0° < \alpha_f < 10°$ and increases with its further increase.

This increase is different as per fuselage different shapes. Fig. 7.7 shows change in ratio $K_\alpha = C_{x\alpha}/C_{xo}$ of fuselage drag coefficients at an angle-of-attack α_f and $\alpha_f = 0$ for the most common fuselage shapes. For the most traditional shapes of Russian helicopter fuselage (type 1) at negative angle-of-attack K_α increases more intense than at positive values, when the tail boom is aerodynamically shaded by fuselage.

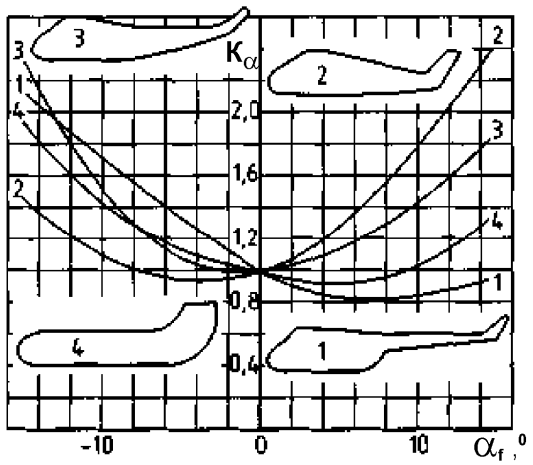


Fig. 7.7. Typical Curves Characterizing Change of Coefficient K_α
with angle-of-attack α_f for Four Traditional Fuselage Shapes

Lift coefficient of fuselage almost linearly increases with angle-of-attack within the range of $-30° < \alpha_f < 30°$, while keeping smooth flow about fuselage, corresponding to

its longitudinal contours. Although the operating range of angles-of-attack the fuselage lift is negative, its value compared to the main rotor thrust is low, and therefore drag fuselage plays a major role in helicopter aerodynamics.

Fuselage drag consists of drags of the fuselage itself, and structural elements installed on it: landing gear, tail bump, tail reducer, antennas, and external fuel tanks.

To reduce the fuselage drag and to improve helicopter flight characteristics the retractable landing gear and tail bump are used, rotor hubs, LG components and tail reducer are covered with cowlings, antennae are built-in the outlines, irregularities on the fuselage surface (skin joints, rivet heads, locks, handles and etc.) are removed, engine intake and exhaust units are provided with good aerodynamic shape.

Sealing and placing the fuel tanks inside the fuselage significantly contributes to reduction of fuselage drag. Fuselage cross-sectional shape has significant impact on parameter C_{xf} for different angles-of-attack. Fig. 7.8 shows the K_α depending on the fuselage angle-of-attack α_f for different cross-sections. In the area of operating angles-of-attack at cruising speed the fuselage with square cross-section has drag, which is four times larger than of the fuselage with circular cross-section.

Length of the central part cylindrical portion essentially affects the fuselage drag. A drop-shaped fuselage will have minimum drag at α_f = 0°, when the largest cross-section is located at a distance of $0.435L_f$ from the fuselage nose. When installing a cylindrical insert, which has length equal to three diameters of the cross-sectional area, the drag increases by 7.5% in comparison with fuselage drag of the same aspect ratio, but without the cylindrical insert.

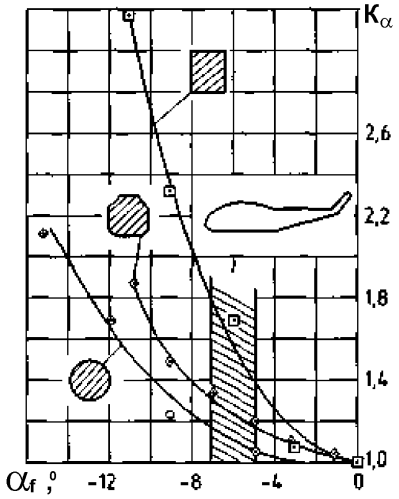


Fig. 7.8. Influence of Typical Fuselage Cross-section Shape upon Coefficient K_α

Fuselage tail section shape significantly affects its drag. To prevent flow separation in the aft fuselage, it is necessary to ensure a smooth narrowing, and the cross-section must have elliptical shape or large rounding radii.

As it is evident form Fig. 7.9, with tail section aspect ratio of λ_{tail} >2 base drag ΔC_{base} does not exist as flow becomes almost unseparated. Optimal aspect ratio of the tail section is that, at which its minimum drag is 2...2.5.

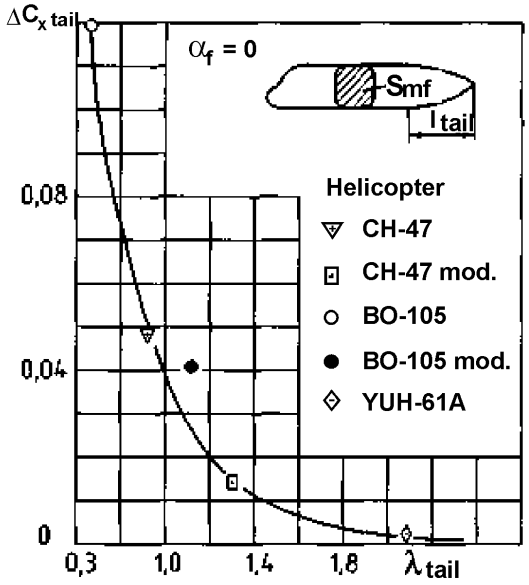


Fig. 7.9. Influence of Fuselage Tail Shape on Base Drag ΔC_{base}

The superstructure over fuselage central part, which includes main reducer, fan, power booster, expendable fuel tanks, generators, etc., contributes to great influence on the fuselage aerodynamic characteristics. The superstructure is usually closed with cowling, covering also swash plate at certain helicopters (such as on Mi-26).

In level flight within certain speed range there is a stall at the rear part of the superstructure with origination of vortices of the «Karman vortex street» type, which cause air flow pulsation in the area of tail rotor and the tail unit, and increased level of transverse vibrations in the helicopter. To eliminate this phenomenon the aerodynamic ridges are mounted in the fuselage upper part near place of transition to the tail boom (as on Mi-26 helicopter), preventing the overflow of the boundary layer from the superstructure side surfaces and promoting for continuous smooth outflow of air from the rear stagnation zone. Sometimes a special deflector is installed (Sikorsky S-61 helicopter) for this purpose in the superstructure rear part.

7.2. INBOARD ARRANGEMENT

The main objective of the inboard layout is to define relative positions of the helicopter units and equipment, accommodation of crew, passengers and cargo and to ensure the specified CG position for variants of the helicopter loading. When making the inboard layout it has to meet a wide variety of requirements, the most important among them are the requirements to ensure reliable and safe operation of units and equipment,

ease of maintenance, as well as the requirements as per accommodation of crew members, passengers, cargo and special equipment. The latter requirements, in particular include:

- Good view from the cockpit;
- Comfort for crew and passengers;
- Ease of evacuation out of helicopter in case of accidents;
- Possibility of rapid loading and unloading of cargo and reliability of its attachment;
- Easy access and use as per special equipment as well as ease of its mounting and dismounting.

Crew accommodation. Proper accommodation of crew members, providing ease of operation and a good view from the cabin, is one of the most important tasks of the inboard layout. Arrangement of crew seats and instrument panels, consoles, handles and levers should meet the requirements of Aviation Rules AP-29 and, as a rule, is governed by the relevant standards.

Levers controlling both individual units and systems, and the helicopter as a whole placed in the cabin at the pilot station should be reachable and visible to him from his station. The most frequently used controls, as well as designed for the most demanding phases of flight and in emergency situations, should be placed in the best areas with regard to reach capability and visibility thereof. Control levers used in a less intense phases of flight, can be installed in a location reachable for the pilot, but monitored at a certain deviation from the main operating position or with turning the head.

Control stick, collective pitch lever and pedals almost continuously used in flight, should be positioned relative to the pilot seat to meet physiological requirements, i.e. so that their use causes the least tiring.

Location, color and shape of the levers and buttons used in an emergency, should provide quick and error-free identification of them. This should eliminate possibility of inadvertent switching on and off in flight by setting the locking device (snap, caps, etc.).

Instruments mounted on the control panel and used by the pilot for piloting, navigation and control of the power plant and helicopter systems, should be clearly visible from the pilot workplace when he glances forward in the direction of flight at small deviations of the pilot of its main position in the seat.

Flight and navigational instruments that make up main instrument group providing information about the helicopter position in space and direction of its movement should occupy the central place on the control panel and to be permanently visible for the pilot.

Gauges of the power plant should be kept on the control panel in a compact group, and it is normal visibility and ease of control with pilot workplace should be provided.

An important task of the cockpit layout is to ensure good visibility. Required viewing area of the helicopter pilots are determined by three factors:

1) need to see all the rotating rotor blades;
2) sufficient view of the landing pad when landing with positive pitch angle;
3) need to view the transverse dimensions of the helicopter (one wheel of the landing gear and the wing tip.

Taking into account these factors the helicopter cockpit view area is provided.

For medium helicopters with takeoff weight of 10 ... 12 tons viewing angles α_{up}, down α_{down} and back β_{back} usually have the following values: α_{up} = 20 ... 22°; α_{down} = 55 ... 60°; β_{back} = 35 ... 37°.

For emergency evacuation the helicopter cockpit must be provided with emergency exits, which include doors and maintenance hatches, and their size and location should correspond to the AP-29 requirements outlined in the aviation regulations. Emergency exits in the cockpit must be provided as follows: one on each fuselage side or one overhead escape hatch and one emergency exit on any fuselage side. Size and location of exits should ensure rapid evacuation of the helicopter crew.

Accommodation of passengers and cargo. The layout of passenger and cargo cabin should provide accommodation of the given number of passengers and cargo volume (dimensions) with minimum weight of empty helicopter. In this case it shall be possible to re-equip passenger cabin by changing its layout, type of seats and their installation pitch. For transport helicopters variety of options for their use (transportation of goods with different dimensions, including on the external sling, transportation of people and others) should be considered.

Passenger cabin of the helicopter, as a rule, shall consist of the following basic sections:

Passenger compartment;

Coat stowage;

Entrance area or area near the entrance (main) door free of seats in multi-seated helicopters;

Toilet (in multi-seated helicopter). For helicopters with a flight time of less than 1 hour and the number of passengers at least 15 toilets can not be installed.

The passenger compartment, as a rule, should be separated from the part-time, baggage and cargo space and crew cabs by partitions. Passenger compartment, depending on the specific volume per one passenger, seat type, distance between seats and conditions of service, can be of three classes:

I. VIP, with increased seat width and distance between rows (seat pitch);

II. Tourist;

III. Economic. In classes II and III, respectively, narrower seats are installed and smaller pitch between their rows is provided.

Helicopters as a rule have passenger cabins of the third class, with specific volumes per one passenger, including coat stowage and service compartments. In passenger compartments necessary living conditions for passengers should be provided in accordance with the physiological and sanitary requirements: purity of air, amenities, facilities to assist in the event of an accident, emergency landing on ground and water.

The design of the main doors of passenger cabins and their location should provide easy entry and exit of passengers and convenient approach to the passenger gangway. Emergency exit for passengers must be rectangular in shape with rounded corners with radius not greater than 0.1 m and minimum size (see Table 7.1). Exits of type I and II are placed at the passenger cabin floor level, and exits of type I should be placed as far as possible from the places, which may be flammable during an accident. Exits of type III may be located not at the floor level. The extension over floor of the exit of type IV should be less than 0.75.

Table 7.1

Recommended Dimensions of Emergency Exits, m

Exit Type	Width	Height
I	0.61	1.22
II	0.51	1.12
III	0.51	0.91
IV	0.485	0.66

Emergency exits should be available for passengers. The number of the exits as per number of passenger seats is selected according to data given in Table 7.2.

For helicopters with 70 passenger seats or more, additional emergency exits must be provided. When choosing the location of emergency exits of type I, II and III take into account the following factors:

- Accommodation of passengers in fuselage and ease to reach the appropriate exit;
- Likelihood of damage to the fuselage various parts during emergency landing;
- Exclusion conditions when the passengers leave the helicopter through the dangerous area (zones of rotors, engine hot parts, fuel spray).

Each door can be classified as an emergency exit if its dimensions meet the requirements for emergency exits of type I or exceed them.

Each main and emergency exits must meet the following requirements:

– To have movable or removable door hatch, which provides free access of passengers;

– To be openable from both inside and outside, by at most two handles;

– To have means for locking both outside and inside, as well as a guard device, which precludes exit opening in flight as a result of inadvertent actions.

Table 7.2.

Number of Emergency Exits Depending on Number of Passengers

Number of Passenger Seats	Number of Emergency Exits at Every Fuselage Side			
	Type I	Type II	Type III	Type IV
1...19	-	-	1	-
20...39	-	1	-	1
40...69	1	-	-	1

The passenger compartment should have the following basic equipment:

– Seats;

– Luggage racks to accommodate carry-on baggage;

– Panels for personal use with button to turn on/off the lights and air blowing;

– Loud-speaking devices for passenger information;

– Light boards «No smoking» and «Fasten seatbelts» switched on by crew members.

Passenger seats should provide enough body support, be comfortable and soft enough that the passenger does not feel tired at the end of the flight. In helicopter emergency landing seats should prevent the passenger head, hands and feet hitting against the surrounding objects and structures (seat backs, armrests, walls, etc.). The number of passenger seats in a row depends on the shape and sizes of fuselage cross-section and should be no more than six in the presence of one main aisle. For more than six seats in a row the helicopter cabin has to have two aisles. The main aisle width in the passenger cabin between the seats should not be less than indicated in Table 7.3.

Table 7.3

Aisle Width in Passenger Cabin

Number of Passenger Seats	Height from Floor, m	
	less than 0.63	0.63 and more
Less than 20	0.30	0.51
20 and more	0.38	0.51

The required helicopter fuselage width can be defined by the formula

$$B_{fus} = Ba + b(a + 2) + B_1 + 2c + 2d,$$

where a – number of seats in a row; B – seat width; b – arm-rest width; b ≈

40...50 mm; B_1 – aisle width; c – gap between a seat and cabin wall; c ≈ 60 mm; d – cabin wall thickness.

The passenger cabin length is determined depending on the number of rows of seats according to the formula

$$L_{cab} = l_1 \left(\frac{n_{pas}}{k_c} - 1 \right) l_1 + l_3 ,$$

where l_1 – distance from the front cabin wall to the seat back of the first row; l_1 = 1200 mm; l_2 – distance between seat rows (pitch); l_2 = 780 mm; l_3 – distance from seat back of the latter row to the rear cabin wall; l_3 = 50 mm; n_{pas} – number of passengers; k_c – number of seats in one row.

Baggage and cargo stowage should provide accommodation of passenger luggage on the basis of the maximum number of seats provided on the helicopter, as well as mail and cargo in accordance with the helicopter maximum capacity.

Location and dimensions of cargo hatches should be comfortable enough for the entrance of loading and unloading mechanization and provide possibility of easy and quick execution of this work.

Cargo compartment of transport helicopters must be equipped with machinery, devices and other means of loading/unloading of the goods transported in non-aerodrome conditions. This cabin floor must be protected from damage during shipping movements.

For easy loading and unloading of cargo cabin transport helicopters usually have cargo hatches in the form of two rear flaps, as in Mi-4, Mi-6, Mi-8, or combination of the cargo ramp and two side flaps in the fuselage rear part as in the Mi-26 helicopter. Fig. 7.10 shows the transverse and longitudinal sections of the helicopter cargo cabin of various types.

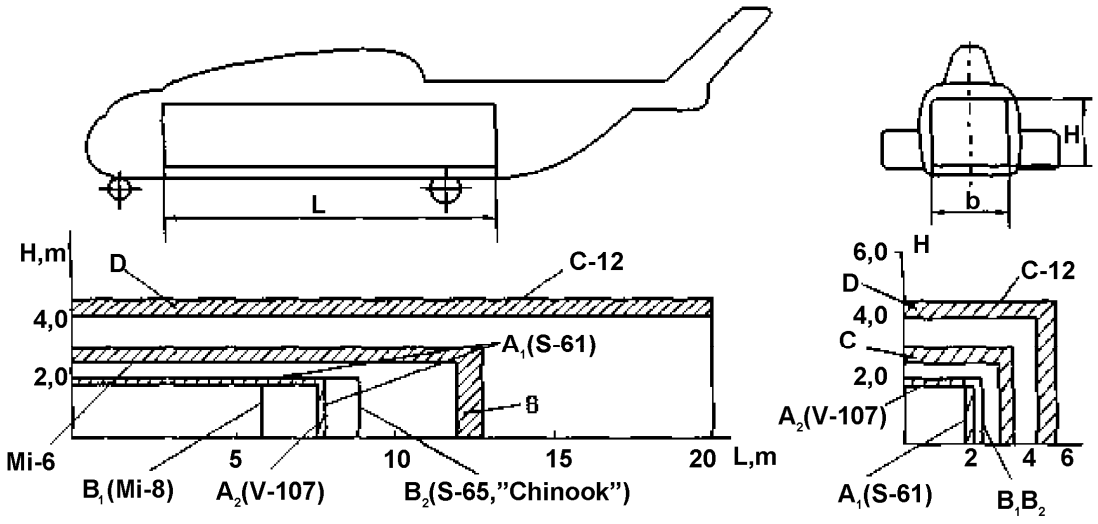


Fig. 7.10. Dimensions of Helicopter Cargo Cabin

Placing of engines. Location of engines on the helicopter depends on many factors. Main factors are: type and number, helicopter purpose and configuration, dimensions and weight category.

In single-rotor or coaxial helicopter the engine (or engines) can be located underneath, in front of, aside and back with respect to the main reducer.

The first production helicopters had an arrangement where the engine was placed under reducer. Such arrangement provided a very compact layout of the helicopter main units, but not allowed to use the most convenient stowage space under the reducer. Therefore, in the subsequent transport helicopters (Mi-4, S-55, S-58), the engine was placed ahead of the cargo compartment.

The use of turbo engine in helicopters led to development of the layout, which has become a classic and is used for many transport and multi-purpose helicopters: two engines are placed close to each other just ahead of the reducer. This arrangement allows the engine to ensure good CG range of the helicopter, minimum mass of the power plant and to use completely space inside the fuselage to accommodate cargo, passengers or special equipment.

Disadvantages of this arrangement are the following: low fire safety and the lack of helicopter combat survivability. Although the engines located side by side, are separated by the firewall, in the event of a fire in one of the compartments the fire could penetrate into the neighboring compartment. In addition, when using helicopters in combat conditions the attack of the projectile against power plant could disable both engines.

The next step is improving the layout that is placing engines on the fuselage sides: in front, from the rear or aside of the main reducer. Most US military helicopters, created at the end of the 60-70-ies (CH-53A, CH-53E, UH-60A, AH-64A and so on) have similar layout of the engines.

Such arrangement of power plant is characterized by presence of additional reducers and cross-shafts, which, depending on the engine's location, come to main reducer straight or at an angle. Although the additional elements increase mass of the power plant, but engines arranged so provide more comfort inside the cabin by reducing the noise level, increase fire safety and combat survivability of the helicopter.

In the light small helicopters of civilian use, when it is necessary to provide minimum structure mass and the survivability requirement is not actual the engines often placed close to each other behind the reducer (S-76, Bell-222, A-109, and others.), or behind the reducer with inclined shaft and engine (Hughes OH-6A, Hughes 500).

Definition of helicopter CG position. The most important task of helicopter inboard arrangement is its center of gravity position range, i.e., determination of the helicopter center of mass and holding it within a predetermined range of positions relative to the main rotor axis. This process is inextricably linked with definition of the

helicopter layout and permanently accompanies it. At the same time, along with the satisfaction of the above requirements to the arrangement the desired position of the helicopter center of mass is achieved, and its motion due to combustion of fuel or changing the payload is within acceptable limits. Extreme values of the front and rear (with respect to the rotor axis) CG position define the range of acceptable longitudinal CG positions, and vertical distance from the helicopter center of mass to MR hub center defines helicopter vertical CG position.

Range of allowable longitudinal center of gravity positions of single-rotor helicopter depends on many factors and is determined primarily by swash plate tilt limits in longitudinal direction and required longitudinal control margins on conditions of the helicopter balancing under extreme flight conditions. The range of swash plate tilt limits typically is ± 5 ... ± 8 °. It is limited by design considerations, because the more the swash plate tilt angles the greater overall dimensions and weight of its components, elements of control systems placed behind boosters and boosters themselves, as well as the need to exclude possible hit by rotating blades against fuselage and tail boom under extreme flight conditions and when maneuvering in the air.

Margins of swash plate longitudinal tilt back and forth depend on the design of the helicopter systems and corresponding balancing characteristics. With that the longitudinal control smallest forward margin is determined at the maximum speed of level flight at extreme aft CG position, MR minimum permissible speed of rotation and maximum helicopter flight weight as well as while climbing with maximum power of engines and high speed. The longitudinal control smallest backward margin is determined in hovering out of ground influence with the rear wind at extreme forward CG position.

Longitudinal control minimum margins depend on the flight conditions (flight altitude and ambient air temperature), as well as on the helicopter control effectiveness.

It is known that the control efficiency defined by the control moment increment with travel of control lever depends on MR type and design, helicopter vertical CG position, and MR speed and thrust. Thus, the helicopter with hinged MR having spacing of flapping hinges of 5 ... 6%, possesses control efficiency by 2.5-3 times higher than the efficiency of the helicopter with MR suspended on cardan. With increasing spacing of flapping hinges the helicopter efficiency increases, and the helicopter with hingeless MR possesses high efficiency, proportional to equivalent spacing of flapping hinges. The same positive impact on the control efficiency is contributed by the number and weight of the blades, as well as helicopter low center of mass, i.e., vertical CG position.

The higher the helicopter control efficiency the lower control margins should be provided; for the same range of swash plate limit tilt the helicopter having greater control efficiency has larger range of permissible center of gravity positions.

Although all of Russian single-rotor helicopters in order to exclude the possibility

of hitting by MR rotating blade tips against tail boom the swash plate extreme tilt backward is less than tilt forward, the allowable extreme forward center of gravity position is always greater than the backward CG position. This is due to the fact that moments arising in level flight at main rotor and stabilizer cause the helicopter to pitch up and easier to be balanced with the forward CG position.

In the helicopter operation the established longitudinal range of permissible center of gravity positions must be always observed. Because of the helicopter improper loading its CG position will go beyond admissible range, the longitudinal control margins during said flight conditions will be not enough, and the helicopter will lose control.

Helicopter center of gravity position is found as follows:

Compile a helicopter weight summary with a breakdown of all units by groups. Determine the coordinates of the center of mass of each unit or group of units, depending on the availability of information and the required accuracy. For the origin take the main rotor hub center. One axis *OY* coincides with the main rotor axis, another axis *OX* is perpendicular to it in the main rotor plane. Coordinates of cargos are taken from the side view of the helicopter given on the general arrangement drawing. This view represents the helicopter as a longitudinal section along the axis of symmetry. Determine the static moments caused by weightes of main units about the origin and compile the CG position sheet (Ref. Table 7.4).

The quotient of the sum of static moments by the sum of the masses gives the coordinates of the center of gravity position:

$$X_{CG} = \frac{\sum(mgx)i}{\sum(mg)i}; Y_{CG} = \frac{\sum(mgy)i}{\sum(mg)i},$$

where X_{CG} and Y_{CG} – longitudinal and vertical coordinate of CG position.

It is convenient to represent the center of gravity position with angle φ_{CG}, formed by main rotor axis and helicopter center of mass:

$$\varphi_{CG} \approx X_{CG} / Y_{CG}.$$

CG position is typically found for a number of loading options:

1) for maximum take-off weight – with full payload and corresponding fuel amount; full filling of main fuel tanks and corresponding payload;

2) for normal take-off weight – with full main fuel tanks and reduced payload; with full payload and thus reduced fuel amount;

3) for full payload without fuel (extreme landing case);

4) for full filling of main, suspended and additional tanks without payload (ferry option);

5) for empty helicopter without payload and fuel. For the last case the CG position is calculated in order to eliminate helicopter turn-over backward while parking due to proper placement of landing gear.

Table 7.4. Helicopter CG Position Sheet

Unit	Unit Mass m_i, kg	CG Pos. Along Axis x, м	Static Moment $m_i x_i$	CG Pos. Along Axis y, м	Static Moment $m_i y_i$
I. Main rotor:					
blades					
hub					
II. Control system:					
booster					
manual					
III. Transmission:					
main reducer					
intermediate reducer					
tail reducer					
transmission shaft					
IV. Tail rotor:					
blades					
hub					
V. Power plant					
VI. Fuel system					
VII. Fuselage:					
nose part (15%)					
middle part (50%)					
tail part (20%)					
reducer attachment (4%)					
cowls (11%)					
VIII. Landing gear:					
main (82%)					
nose (16%)					
tail (bump)					
IX. Electrical equipment:					
X. Equipment:					
instruments in flight compartment (25%)					
radio equipment (27%)					
hydraulic equipment (20%)					
pneumatic equipment (6%)					
auxiliaries (22%)					

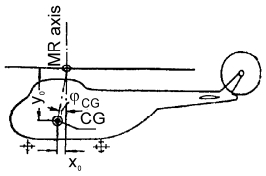
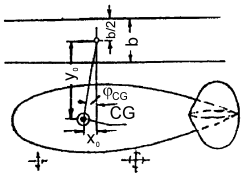
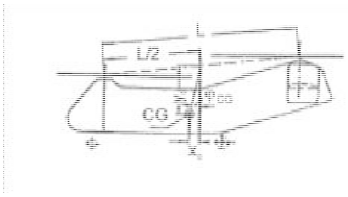
Often center of gravity position is calculated by the angle φ_{CG}, formed by the main rotor axis and line connecting the main rotor hub center with the helicopter center of mass (Ref. Table 7.5)

$$\operatorname{tg}\varphi_{CG} = \frac{x_0}{y_0}.$$

Calculation of CG position and compiling of the accompanying technical documentation (scaled helicopter sketch, CG position sheet and so on) are the part of the volume-weight layout.

Table 7.5 gives values of center of gravity position for normal, extreme forward and extreme aft helicopter CG position for three configurations.

Table 7.5

Helicopter Configuration	CG Position	Normal CG Pos.	Extreme Forward CG Position	Extreme Aft CG Position
Single-rotor with tail rotor 	φ_{CG}	-3°	With stabilizer $S/\pi R^2 = 0.004$ (at $\frac{y_0}{D} = 0.105...0.125$) -6°	2°
Coaxial double-rotor 	φ_{CG}	-3.0°	With stabilizer $\frac{S_{stab}}{F_{stab}} = 0.015$ (at $\frac{y_0}{D} = 0.12...0.14$) -5.0°	-2.0°
Tandem double-rotor 	φ_{CG}	-3.0°	With stabilizer $\frac{S_{stab}}{F_{ом}} = 0.015$ (at $\frac{y_o}{D} = 0.12...0.14$) -5.0°	-2.3°

It is very important in the process of making arrangement of a helicopter to place fuel and payload near the helicopter center of mass, in this case burn of fuel and change of payload had no significant effect on the balance of the helicopter.

Drop cargo and payload carried on the external sling, should be placed and fastened obligatorily close to the helicopter center of mass.

The resulting CG position rarely immediately coincides with required one. Correct CG positioning can be obtained by moving cargo, equipment, individual units or main rotor axis offset from the helicopter center of mass. However, displacement of main rotor axis usually entails displacement of engine, changing in transmission and usually leads to rearrangement of the helicopter. Therefore, positioning the helicopter CG is an iterative process, and in the course of further development of the helicopter project and its construction the CG position is permanently refined.

As a result, calculation of the different variants of the helicopter loading centering the helicopter CG position graph is created, in which it could be defined the helicopter CG at any of its loading option.

On the basis of these calculations the marking schemes for cargo location inside the cargo cabin and arrangement of seats in the passenger cabin are also created.

7.3. LOAD-CARRYING STRUCTURE

The main task of definition of a load-carrying structure is to create or to select the load-bearing schemes of the helicopter, which would provide:

- minimum structural weight of the helicopter parts and helicopter as a whole;
- inherent association (link) of structural load-bearing elements and placement of the payload, crew, equipment, power plant;
- accounting requirements of production and maintainability;
- necessary rigidity giving dynamic load and damping means for the purpose of static and dynamic strength of the structure in the air stream;
- getting the desired service life and safety in case of local failure from fatigue.

One of the first tasks of definition of a load-carrying structure is to establish the operational and technological (industrial) breakdown of the helicopter parts.

Simultaneously with the process of definition of the load-carrying structure a directive technology of manufacturing the helicopter parts and build a helicopter as a whole is developed since the processes that are selected may determine the features of the load-carrying structure.

In the process of definition of the load-carrying structure the maintenance arrangements (access means) must be provided in the zone where the units and wiring of the equipment and control systems are placed, and approaches to the construction parts to be inspected during operation of the helicopter.

This process is completed with drafting a drawing. The drawing should depict longitudinal section and a plan view of the helicopter, and cross-sectional locations of the cargo (passenger) cabin, engine mounts and the main reducer. The arrangement should also show distribution of cargo, passengers and crew, all the bearing elements of the frame, marked contours of the basic units and the scheme of their attachment to the bearing frame. The structural components drawn on the general arrangement drawing must be sized according to the designed helicopter.

General arrangement drawing, together with an overall view of the helicopter (in three projections) serves as the basis for drawing theoretical drawings of the fuselage and its interface with other parts of the helicopter.

Helicopter overall view allows to link the helicopter basic overall dimensions and specify their shapes. Helicopter overall view (Fig. 7.11) is necessary to make drawings of model intended for wind tunnel tests. Overall view is inextricably linked with the layout and balance, aerodynamic calculation, calculation of stability and control, etc. According to their results the helicopter general arrangement drawing gets necessary changes and additions.

The helicopter general arrangement drawing should be executed in accordance with the requirements of GOST and ESKD and linked to the layout and the text part of the project, in which the necessary calculations and justification of the decisions taken should be brought.

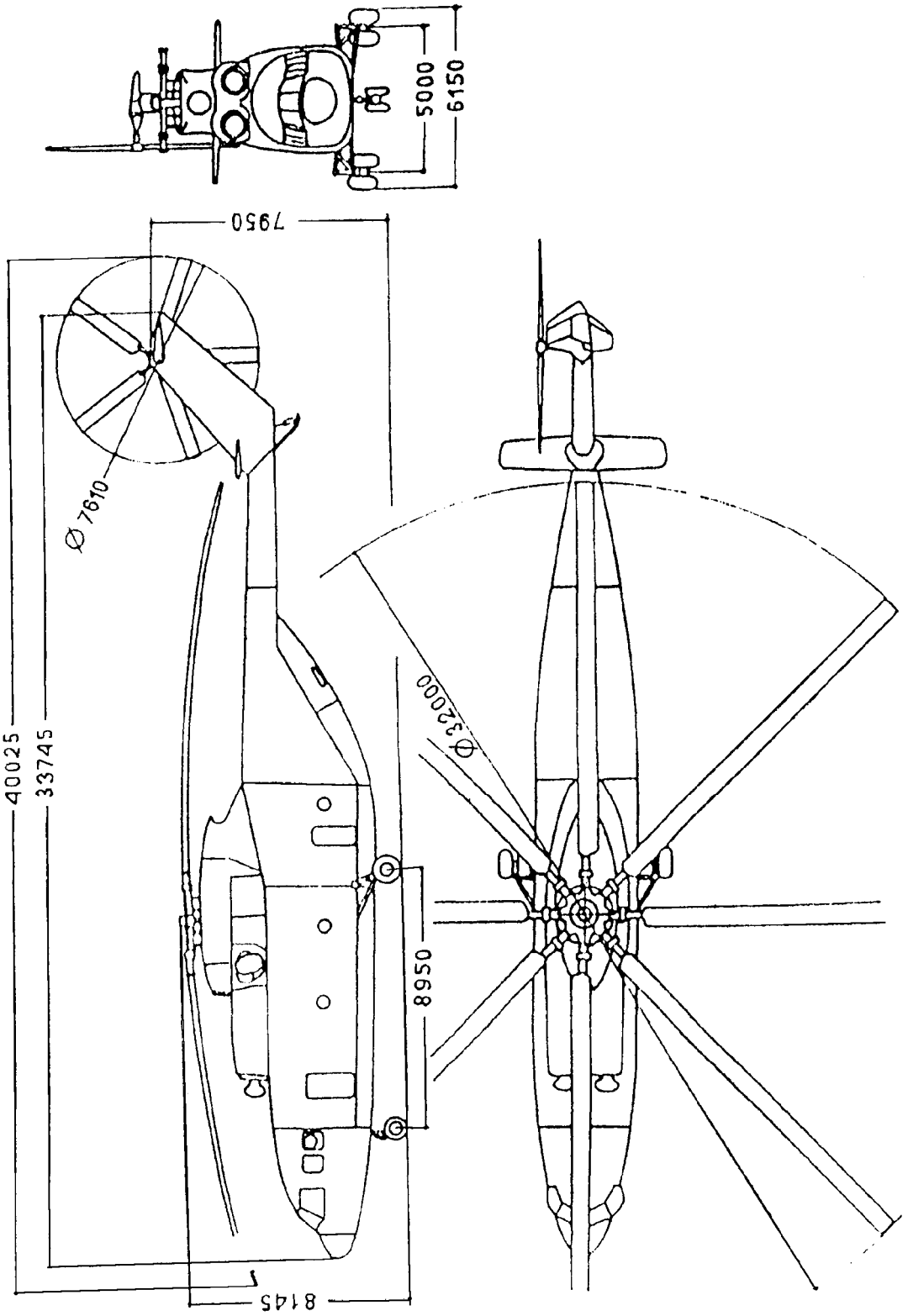


Fig. 7.11. Helicopter Overall View

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Chapter 8. DEVELOPMENT OF HELICOPTER UNITS

8.1. MAIN ROTOR

8.1.1. General

Main rotor is designed to develop lifting force Y_{MR} and propulsive force P_{MR} under any flight conditions, and also to provide helicopter longitudinal and lateral stability (Fig. 8.1). MR defines not only performance and maneuvering characteristics but degree of the aircraft reliability in a whole. This relates with its operation and loading features.

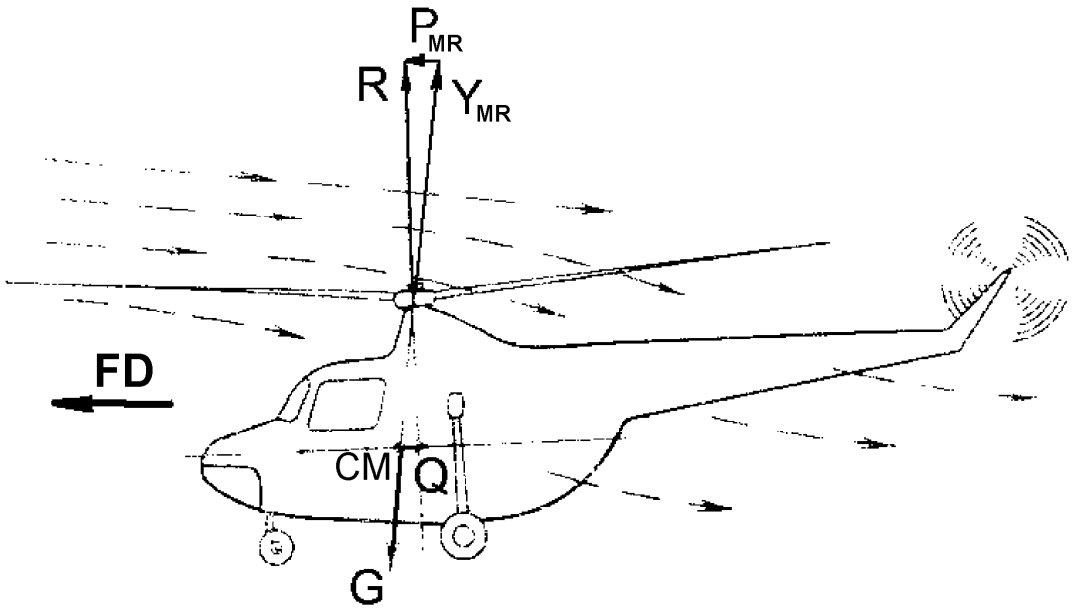


Fig. 8.1. Forces Applied to Single-Rotor Helicopter in Vertical Plane:

P_{MR} – MR propulsive force; Y_{MR} – MR lifting force; G – weight; Q – drag;
R – aerodynamic resultant; FD – flight direction; CM – center of mass

Main rotor of up-to-date helicopter consists of a hub and blades. Blade being a rotating wing analogue is attached to the hub. The blades create lifting force. The hub is a kinematic mechanism providing transmission of torque from MR shaft to the blades, angular movement of a blade in the plane of rotation and plane of flapping, and also changing angle-of-attack of the blade sections.

Specific requirements applied to the developed or modified helicopter (features of purpose, operation and so on) are the reasons of variability of the rotors and their structures. The main features can be taken into account by a number of criteria, to be put at the initial stage of helicopter design:

1. ***The criterion of purpose*** is a measure of general design, developed on the basis of helicopter technical specifications, which should provide the ability to perform specified characteristics (range, speed, payload, takeoff and landing conditions) and certain specific requirements (for example, maximum specific load).

2. ***The criterion of performance characteristics***. Performance characteristics are an obligatory item as per designed helicopter. One of the most important design stages is to determine the limits of flight modes. This should consider:

– Characteristics in hovering for different altitudes;

– Maximum and minimum speed of horizontal flight, and maximum rate of climb at any altitude;

– transition to gliding flight and landing in autorotation under any flight condition;

– improved maneuverability with high overloads in the entire range, which requires installation of additional wings and (or) auxiliary traction devices.

3. ***The criterion of controllability.*** Today's helicopters are defined not only by performance characteristics, but also by controllability and stability characteristics. According to the recommendations it is necessary to ensure the level of these factors not below the minimum permissible values under any flight conditions.

4. ***The criterion of noise and vibration levels.*** Noise produced by helicopter is determined both inside and outside the aircraft. The noise level inside is limited on the basis of empirical standards, taking into account comfort, possibility of verbal communication and physiological factors. External noise is regulated by normative documents, which can be explained using a helicopter as means of transport in densely populated areas (such as an air taxi in megacities).

The problem of vibration is more important than the noise because it leads not only to worse passenger comfort, but also to significant reduction in the level of helicopter operational reliability. The solution to this problem is complex and requires a comprehensive approach, so now we can only talk about the attempts to reduce the level of vibrations.

MR design process is carried out taking into account these criteria on the basis of information on the overall configuration of the aircraft, taking into account the number and position of main rotors, rotor type, and type of drive system (mechanical, compressor, etc.). In addition, the designer should know main mass and performance characteristics of the helicopter.

Creation of MR and its components comprises the following stages:

1) MR preliminary design – to select specific load p, diameter D, number of blades z, solidity ratio σ and circumferential speed ωR;

2) blade aerodynamic design – development of an external appearance and theoretical contour. Selecting of blade planform, a set of airfoils, distribution law of thickness ratio $\bar{c}$ and twist φ the blade spanwise;

3) definition of the limit flexure angles in the planes of rotation and flapping, of the blade twisting angles in the process of helicopter control under any flight conditions;

4) blade construction:

- choice of material for the blade main elements, of spar shape on the basis of existing constraints (design, technological, operational, economic, etc.);
- development of blade mass-elastic characteristics and cross-section of the blade spar chordwise and radiuswise on the basis of static strength calculation;
- correction of blade mass-stiffness characteristics chordwise and spanwise to prevent resonance within the rotor operating speed range and to ensure sufficient margins for aeroelastic stability (calculated kinds of flutter, divergence, ground resonance);

5) MR hub construction:

- development of the hub concept based on the terms of reference or preliminary calculation;
- hub kinematic scheme selection (type and location of the hinges, or equivalents thereof) in view of the design and operational constraints;
- choice of materials for the hub main elements;
- design and technological work on the details of the hub elements (definition of the geometry and production technology);
- evaluation of stiffness in the plane of rotation and selection of type and parameters of the drag hinge damper;

6) production of experimental samples of MR elements, strength and endurance full-scale tests with the subsequent completion of design and documentation of test results;

7) production of the rotor prototype, performance of ground, flight and fatigue tests;

8) preparation for mass production of MR blades and hubs, factory testing and certification.

8.1.2. Classification of Main Rotors

MR type is selected at the helicopter preliminary design stage. The rotor type largely determines the aircraft controllability, strength characteristics of the carrying system structure elements. Generally, it depends on the design of the blade root and its attachment to the hub. The design of the blade root has a significant impact on the blade motion in the flapping plane and in the plane of rotation, and hence the helicopter sensitivity characteristics, its vibration, loads and aeroelastic stability.

MR types are distinguished by presence of flapping (fl.h) and drag (dr.h) hinges, and hence the fact, whether the blade rotates as a rigid body or under bending deformation of its root. Currently, there are four main types of rotors:

- fully articulated (Fig. 8.2, a);
- semi-rigid (Fig. 8.2, b);
- hingeless (Fig. 8.2, c);
- "rigid".

Fully-articulated Rotor

Each blade of this rotor type is attached to the hub separately, typically by means of three hinges: flapping, drag and feathering ones. The main advantage of the fully-articulated rotor is that each blade performs flapping movement independently due to action of aerodynamic and inertia forces. This gives significant reduction of bending loads (as the bending moment at the hinge is zero). Articulated rotor is a classic design solution of the load compensation in the blade root and moment compensation on the hub. Its concept is simple enough, and the blade movement analysis as a rigid body is not a major difficulty.

Unpleasant feature of the hinged MR lies in low moment characteristics. Because due to small offset of the flapping hinge axis the pitch and roll moments are practically not transferred from the blade to the hub. This results in reduced levels of controllability, damping and dynamic stability (especially at low speeds and in hovering), narrowing the operational center of gravity range.

This design has a number of significant drawbacks. The need to shift and offset of the flapping and drag hinge axes increases the overall size and hence mass of the hub. Introduction of the drag hinge into the structure has caused the need to install mechanical oscillation damper in order to avoid the instability condition such as “ground resonance”. All this leads to complicated construction, great amount of parts, need to ensure reliable lubrication of moving parts.

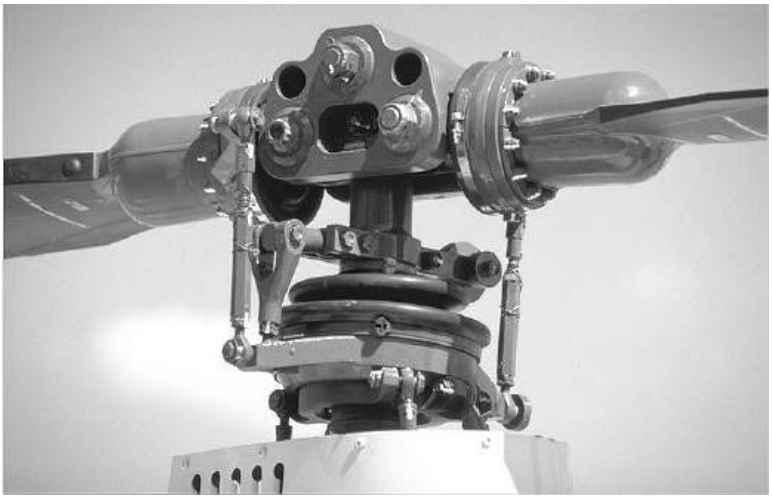
Bearings of the flapping and drag hinges work in very stressful conditions, as they transmit simultaneously blade thrust and centrifugal forces to the hub. As a result of the above, it can be noted that the hub requires a large amount of maintenance work. Its use also results in a significant increase of helicopter drag. Replacement of conventional "mechanical" bearings with the self-lubricating bearings, torsion bars or elastomer bearing can significantly simplify the design and operation.

This type of MR, despite rather significant shortcomings, is the most widespread.

a



b



c



Fig. 8.2. Main Rotor Types:
a – fully-articulated (Mi-24V); b – semi-rigid (Robinson R-22);
c – hinge-less (AS 355 ″Ecureuil 2″)

Semi-Rigid Main Rotor

The name of this rotor type is insufficiently precise, as it gives associations with a rigid rotor. However, semi-rigid rotor is a hinged rotor with a common central hinge. Two rotor classes implementing this principle can be distinguished:

- Rotor of the rocker type (having hub with common flapping hinge);
- Main rotor based on cardan (cardan rotor).

Cardan rotor usually has two or three blades connected to the hub by the feathering hinges only (there are on flapping and drag hinges). The hub is attached to the shaft of the main gearbox via universal joint – cardan. Essentially, a cardan rotor is an analog of the rotor with swinging hub and therefore it has similar advantages and disadvantages. This structure is widely used for the tail rotor hubs.

Hingeless Main Rotor

Hingeless rotor sometimes is referred to the "hard" rotor class. This is not entirely correct for the today's designs, as a blade or joining element has great flexibility. The blade is attached to the hub only via a cantilevered feathering hinge (or element performing its role), with no flapping and drag hinges. Functions of the hinges are implemented by elastic deformations of the hub elements or root blade sections. Development and evolution of this MR type became possible after introduction of composite materials that meet the requirements of high elasticity, low density and high fatigue strength.

This rotor type has simple structure (compared with hinged type), obtained thanks to absence of hinges, bearings, and so on. This reduces the parasitic drag and simplifies maintenance.

"Rigid" Main Rotor

"Rigid" rotor is a kind of a propeller analogue. Usually, the term refers to a hingeless rotor with increased blade stiffness in the plane of rotation and flapping plane. Such blades have minimal flexibility, intended for perception of extremely high bending loads. These rotors are characterized by the absence of flapping hinges, drag hinges, droop stops and dampers.

"Rigid" rotor is usually developed for aircraft having configuration, which partly differs from the classic helicopter. The first work in this direction was dedicated to rotor with ABC (Advanced Blade Concept) – rotor that implements the concept of advanced blades.

Main rotor is a source of vibrations, and there are a number of different methods developed to reduce fuselage vibration level as in a whole or of its individual parts.

Most widespread method lies in installation of dynamic dampers on the hub and

blades. Their operating principle is based on the phenomenon of antiresonance – termination of protected weight fluctuations.

For example, installation of bifilar (suspended at two points) pendulum having weight of 90 kg on the Mi-8 made it possible to reduce frequency of the fifth vibration harmonic in 3 ... 5 times, depending on flight mode.

8.1.3. Main Rotor Blades

Main rotor blades are the most responsible and loaded units, ensuring development of required lift. Blade must meet very diverse and contradictory requirements:

1) providing static blade droop at the parking lot (the blade end should not touch other elements of the airframe);

2) ensuring the requirements of static strength: stress acting in the blade spar from the centrifugal force in the air and gravity forces from the parking lot, should not exceed the allowable stress;

3) ensuring adequate margin in flutter;

4) absence of resonance oscillations;

5) obtaining the dynamic stresses, providing required lifetime;

6) providing sufficient rigidity for bending blades in planes of traction and rotation, as well as in torsion;

7) obtaining high lift-to-drag ratio;

8) ensuring high (as per lifetime) cost and manufacturability.

Blade Construction Tasks and Features

Main feature of MR blade is that it works under effect of loads variable in time. These loads not only depend on the aircraft operating modes but are largely determined by the mass and stiffness characteristics of the blade itself.

The process of blade construction lies in determination of stiffness, mass and geometrical characteristics of its basic structural elements. Thus it is necessary to solve complex structural and technological problems, considering the conflicting demands, and above all - the requirements of ensuring high fatigue strength and service life. Furthermore, when developing a blade design you must strive to reduce its weight, since it determines the magnitude of the centrifugal force acting on the blade, hinges, MR sleeve. However, we must bear in mind a number of restrictions: for mass characteristic, magnitude of flapping, minimal feasible size of parts in terms of production process.

Typically, the work aimed at the blade construction includes the following steps:

1) development of blade load-bearing structure (LBS);

2) selection of materials for individual structural elements;

3) determination of geometric characteristics of major load-bearing and structural members of the blade section on conditions of static structural strength;

4) calculation of mass, stiffness and centering blade characteristics;

5) calculation of natural frequencies of blade flexural vibrations and check the design dynamic strength;

6) calculation of strength of the blade attachment unit;

7) selection of production processes that ensure high fatigue strength of the blade elements, that is design technological study.

Selecting Blade LBS

When choosing a blade load-bearing structure (LBS) it is necessary to define all the design elements for their functional purpose. In general, it is believed that a blade includes a spar, ribs, stringers and skin, i.e. in principle it includes the same elements as the wing of an airplane. However, arrangement and execution of these elements are very specific, which is explained by the features of blade loading during operation. Typical LBS section of the blade is shown in Fig. 8.3.

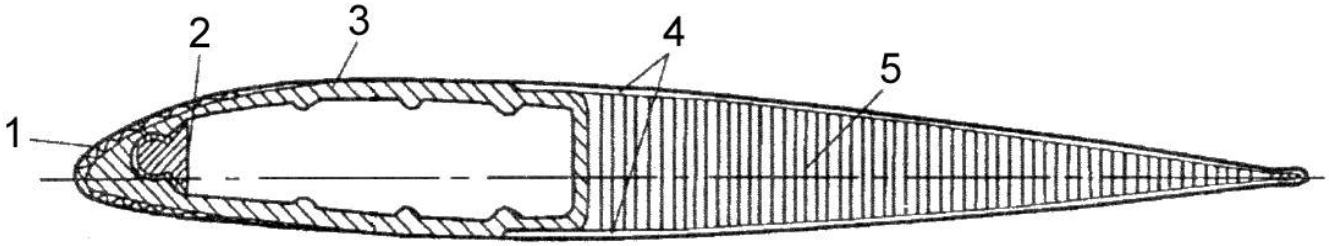


Fig. 8.3. Typical Blade Section LBS:
1 – set of anti-abrasion spacer, heater and dielectric;
2 – anti-flutter weight; 3 – spar; 4 – tail section skin;
5 – tail section filler

Forming the blade LBS substantially depends on MR type (blade attachment options to the sleeve). So, for articulated hubs (with flapping and drag hinges) you can use an arbitrary blade design, and for hingeless and with gimbal suspension – preferably choose a blade with working tail. Today we know a lot of blade LBS, designed by Britain, France, Russia, US companies and others. (Fig. 8.4).

When considering the cross-section of almost any blade you can distinguish two main design elements – spar and tail section.

Spar – main strong element receiving longitudinal and transverse forces, bending and twisting moments.

Tail section – more often it is not a strong element, but it forms an aerodynamic airfoil of the blade section.

Two main types of spar can be identified – tubular and skeleton.

Tubular spar has free shape of cross-section not inscribed in the profile of the blade. This type of construction is most simple as per technology. But it has low resistance to bending and gives the most unfortunate resonance chart with fluctuations in the thrust plane. In blades, equipped with tubular spar with skin not working in bending the resonances phenomenon the most sharply manifests at low flight speeds, that is a key factor for their fatigue strength and service life. Such blades have approximately the same natural frequency in flapping plane and plane of rotation. Significant advantage of using the blade design with tubular spar is its protection with a frame, which prevents mechanical damage in service.

Skeleton spar is inscribed in the front part of the airfoil. This allows to spread the material to the maximum distance from the stiffness axes, which increases blade rigidity, lowers acting stresses and consequently reduces its mass. To control the fatigue failure of the material the so-called damaged spar signaling system is used. Application of skeleton spars allows to reduce anti-flutter weight as spar itself represents a considerable mass concentrated at the front of the blade. This type of spars allows to realize maximum bending and torsional stiffness. With growth of the blade stiffness frequency of natural oscillations increases. This leads to the fact that certain frequencies can move away from the exciting harmonics (i.e. from resonance) and others - can approach it.

Skeleton spars may be produced of two types: of molded workpiece or as composed of separate longitudinal members. Extruded spars are the most widely used. However, it is very difficult to provide twist and change the blade airfoil spanwise in this type of spars. The required airfoil height is provided only by milling the outer surface. This design is best suited for a rectangular blade (narrowing $\eta = 1$) as well as it is advantageous in mass production of blades.

Increasing stiffness in the flapping plane can be achieved by increasing the airfoil thickness ratio. But in this case there is a significant increase in profile drag, which reduces the rotor lift-to-drag ratio. When using skeleton spars stiffness in the thrust plane can be achieved by using multiple contours (see. Fig. 8.4, k, l). It also allows you to increase the blade survivability.

Increased stiffness of the skeleton spar in the rotational plane can be achieved by increasing the spar relative width. However, if vspar relative width exceeds0.4 there is a significant shift of the blade section center of mass backward, which reduces the margin in flutter and requires installation of larger anti-flutter mass. By analogy with the flapping plane increasing rigidity in the plane of rotation is possible by introducing several horizontal contours (Fig. 8.5). This concept also allows to increase torsional rigidity and to realize simple and reliable fastening of the blade butt to the hub by joint of “ear-fork” type (each contour automatically forms a joint lug assembly when winding the spar of composite).

Selecting the LBS must provide the required stiffness and resonant characteristics of the blade. A good example is the provision of independent compartments in the tail part behind the spar.

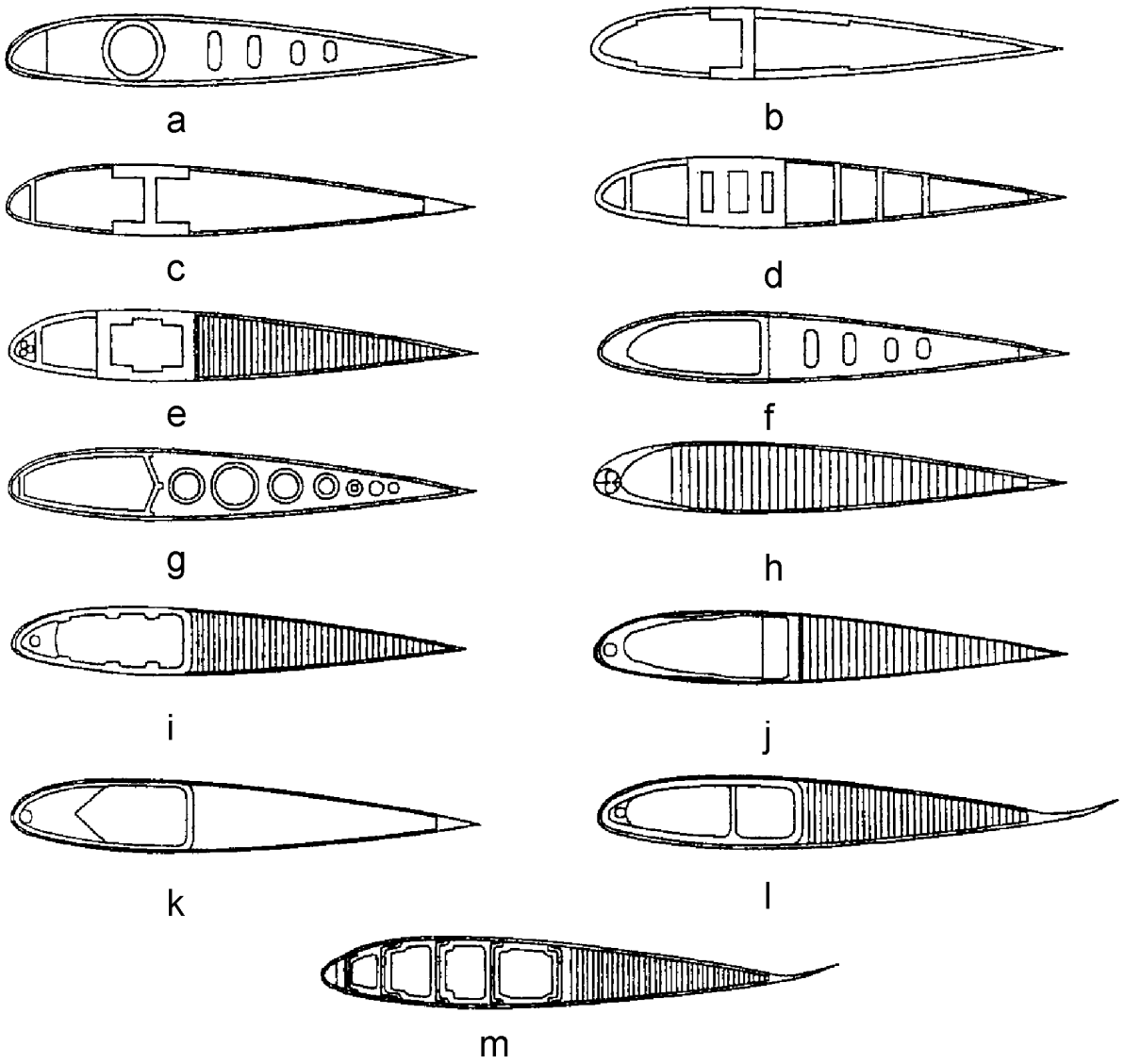


Fig. 8.4. Load-Bearing Structures of Helicopter Rotor Blades:
a – Mi-2, Mi-4, Mi-6; b, c, d, e – experimental structures; f – Belvedere; g – H-21B, CP-47A Sea King; h – H-16 Piasecki; i – Mi-8, Mi-24, Mi-28H, CH-47D Chinook; i – Sea king, Westland 30, Lynx, EH-101A; j – AS.335 Ecureuil, AS.365 Dofen; k – Ka-50, Ka-52, Mi-43; l – AH-64 Apache

Skin is the most loaded part of the tail section design. It transmits to the spar centrifugal force of the tail section, bending moments generated by aerodynamic and inertial forces, and deforms in the attachment area together with the spar (Fig. 8.6). The main objective of the tail sections is to prevent skin work in flexural vibrations of the blade, and, consequently, to prevent origination of sheathing alternating stresses with large amplitude in the material.

Accordingly, in the presence of a spar having high bending stiffness in the plane of rotation, there is no need for such provision. In this case, the trailing edge may be done jointless to increase rigidity of blades, if it is allowed by the resonant diagram.

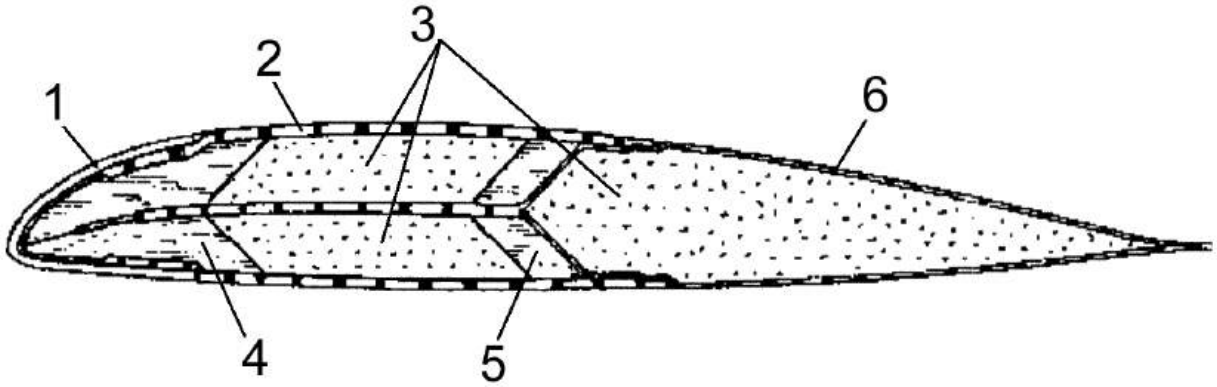


Fig. 8.5. Blade Section LBS with Spar of Two Horizontal Contours:
1 – anti-abrasion plate and heater set; 2 – spar with two horizontal contours;
3 – filler; 4 – front reinforcing part; 5 – rear reinforcing part; 6 – tail section skin

Blade LBS is selected simultaneously with the selection of materials and analysis of possible production methods for the structural elements.

Option of blade design with extruded spar (Mi-8) is shown in Fig. 8.7.

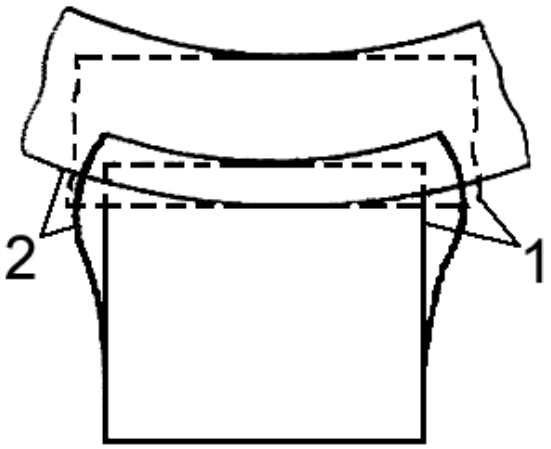


Fig. 8.6. Section Deformation,
Caused by Spar Bending:
1 – spar and skin prior to deformation;
2 – spar and skin after deformation

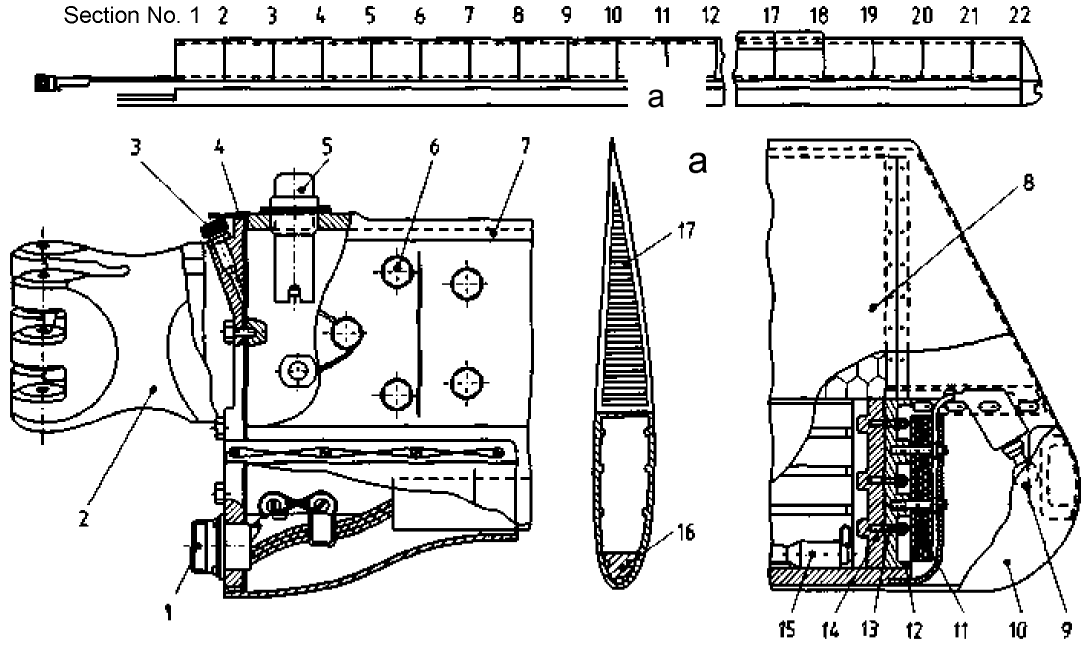


Fig. 8.7. Main Rotor Blade with Extruded Spar:
a – overall view; *b* – root; *c* – section; *d* – tip section;
1 – electrical connector; *2* – blade attachment fitting; *3* – charging valve of system indicating spar failure; *4* – plugs;*5* – spar failure indicator; *6* – bolts securing attachment fitting to spar; *7* – spar; *8* – part of blade section; *9* – signaling light; *10* – protective plate; *11* – balancing plates; *12* – spar cavity plug; *13* – rubber insert; *14* – clamp; *15* – screw stop; *16* – counterweight; *17*– honeycomb filler

Mixed blade structure is used on the Mi-26 helicopter (Fig 8.8.): steel spar, fiberglass frame. The spar and eyelinks of root joint are made as a single unit. The blade nose includes: steel tubular spar 1, foam filler 2, fiberglass skin 3, spar damage detection channel 5, package of electrical anti-icing system 6, anti-abrasion forged plate 7 and heating plate 8. The spar takes all the loads from the nose and tail sections. They are transmitted to the spar using special compensators 4, made of fiberglass, which connects blade skin and spar similar to longitudinal ribs in the blade nose part. Honeycomb is made of special paper such as "Nomex" having high fatigue resistance and corrosion resistance.

The blades made of composite materials (Fig 8.9) as compared with the blades having a metallic spar, are characterized by:

- ability to create and to maintain the exact form in the manufacture of the profile;
- long service life and high reliability;

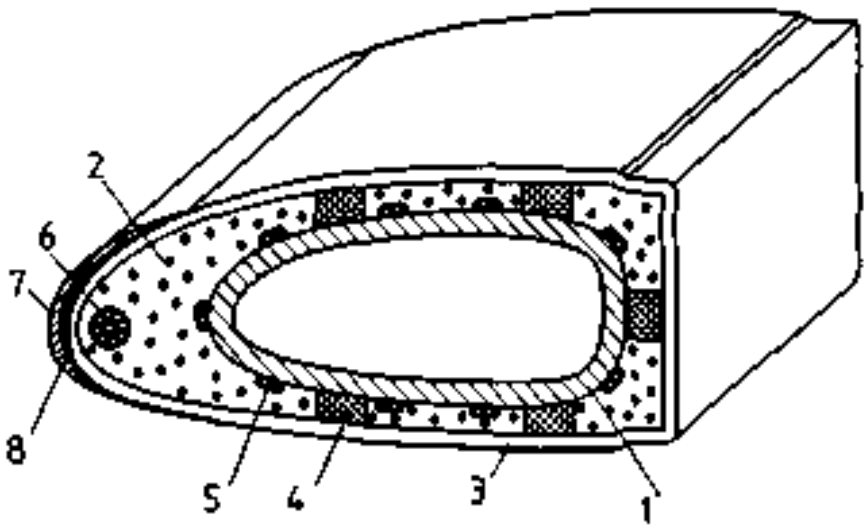


Fig. 8.8. Blade Nose of Mixed Structure

- high specific strength;
- insensitivity to stress concentrators;
- slow and limited spread of fatigue cracks;
- good corrosion resistance;
- internal damping, useful for energy absorbing vibrations;
- high adaptability and less labor-intensive manufacturing.

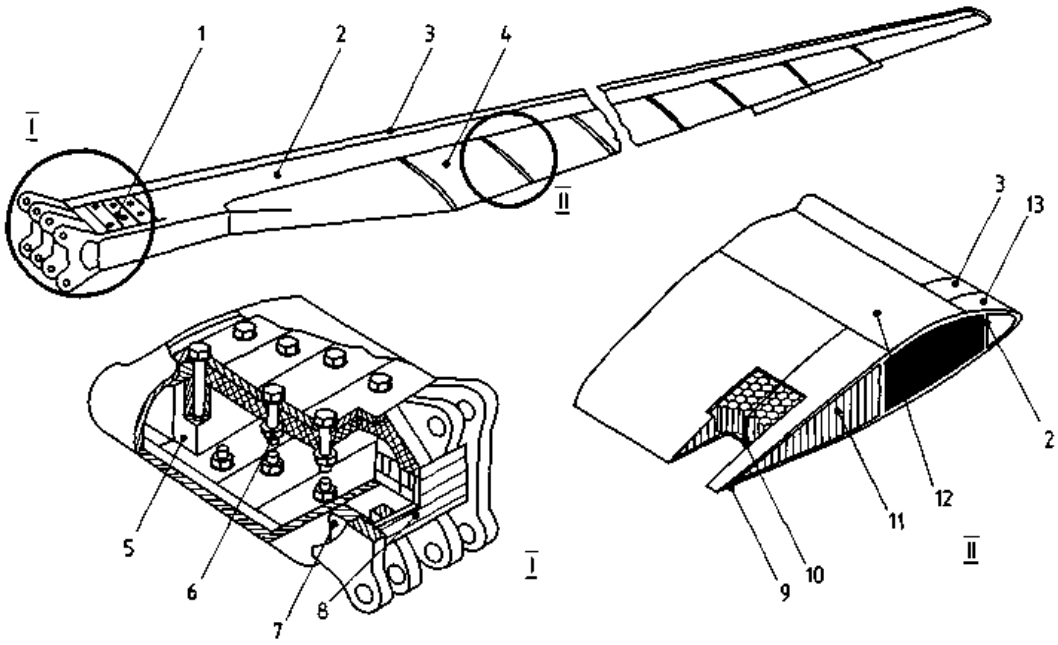


Fig. 8.9. Main Rotor Blade with Fiberglass Spar:
1 - blade mount; 2 - spar; 3 - rubber coating; 4 - tail section; 5 - boss;
6 - bolt of fastening of blade assembly; 7 - spar cover; 8 - inboard balancing weights;
9 - tail stringer; 10 - rubber liner; 11 - honeycomb; 12 - sheathing; 13 – forged piece

8.1.4. Main Rotor Hubs

The hub is designed for fastening the blades and to transfer torque from the main gearbox shaft as well as to take and transmit the rotor forces and moments to the fuselage.

Direct hub construction typically includes the following basic steps:

1. Definition of operating loads and loading conditions for hub elements.

2. Predesign of the hub basic elements.

3. Selection of a material having the required strength characteristics, and details of production technology.

4. Choice of permissible stresses based on the loading conditions, part configuration and selected material.

5. Determination of the design stress (σ_{des}, τ_{des}) in the most dangerous sections of pre-developed components. The following conditions must be satisfied

$$\sigma_{des} \le [\sigma]_{perm} \text{ and } \tau_{des} \le [\tau]_{perm}.$$

In default of these conditions it is necessary to clarify the dimensions of the cross sections so that the conditions are satisfied. If the section has excessive strength, it should be weakened.

6. The design finishing of parts: updating sizes based on design and technological considerations, coordination with existing standards and norms.

Main indicators of the design perfection of main rotor hubs include: hub weight, lifetime and possibility of its further increasing, simplicity and convenience of service, manufacturability of parts and components, load carrying capacity of bearings, level of stresses in the parts exposed to variable loads, efficiency of blade dampers.

Main Rotor Hub Configurations

The hub structural diagram is determined by the rotor type. There are four main types - articulated, semi-rigid, hingeless and "rigid" MR.

Fig. 8.10 shows four typical MR hubs Possible schemes of MR hubs are shown in Fig. 8.11. Most often, they are versions of the above-mentioned hubs.

Choice of the hub scheme is determined by the aircraft purpose, as well as by the traditions of the design office or company. Varying types of hubs, you can significantly change the nature of loading of the hub itself the blade root, helicopter controllability, to solve some problems of flight dynamics. In addition, the hub type may substantially affect the aerodynamics of the helicopter as a whole.

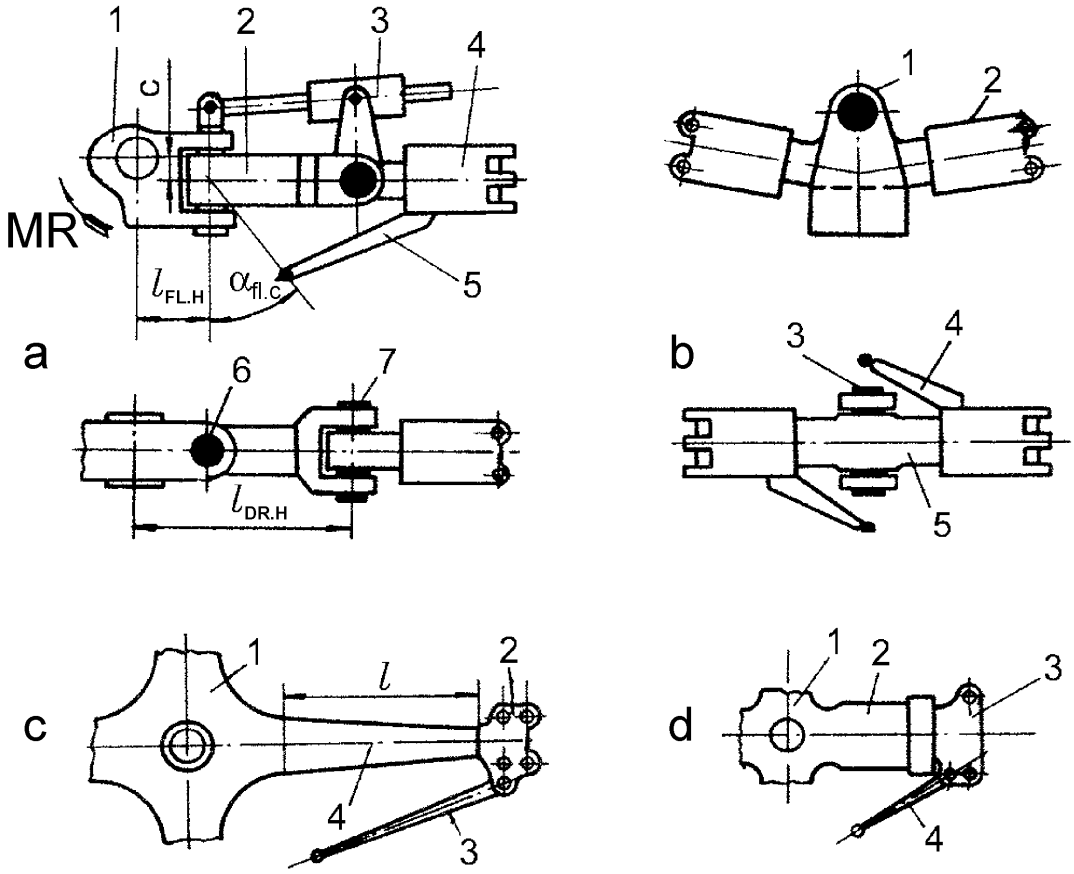


Fig. 8.10. Basic Structures of MR Hubs:
a – fully-articulated hub:
1 – body; 2 – intermediate link; 3 – damper; 4 – feathering hinge (FeH);
5 – driving link; 6 – flapping hinge (FlH); 7 – drag hinge (DrH);
b – hub with common flapping hinge: 1 – body; 2 – FeH; 3 – FlH;
4 – blade driving link; 5 – rocker lever;
c – hingeless hub: 1 – body; 2 – attachment fitting of blade and driving link;
3 – driving link; 4 – flexural sleeve – torsion;
d – blade rigid attachment: 1 – body; 2 – FeH;
3 – attachment fitting of blade and driving link; 4 – driving link

Geometric Parameters of Hubs

On the domestic production helicopters until recently the main rotor with hinged blades mounted to the hub was used. However, now a days rotors with hingeless attachment of blades by means of suitable elastic elements become more common.

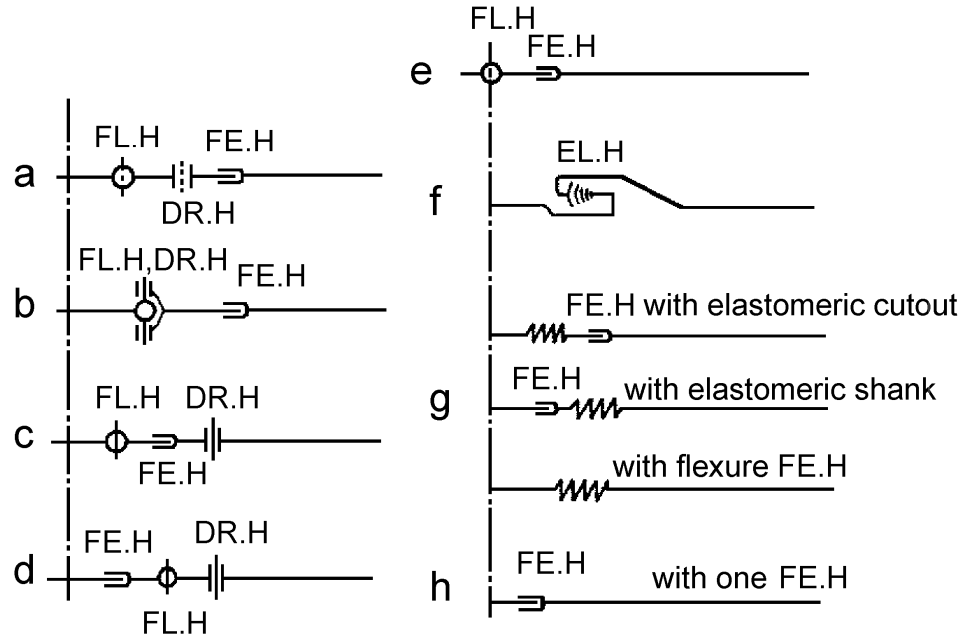


Fig. 8.11. MR Hub Particular Sketches:
a – classic fully-articulated; b – crossing flapping and drag hinges;
c – with extended drag hinge; d – with extended flapping and drag hinges;
e – on cardan; f – with elastomer common hinge; g – hingeless; h – "rigid" rotors

Basic geometric parameters of hubs defining transfer of all loads from blades to fuselage, as well as characteristics of stability and controllability of the helicopter:

1) distance from the hub axis of rotation to the axis of the flapping hinge (flapping hinge offset) – $\ell_{\text{fl.h}}$;

2) distance from the hub axis of rotation to the axis of drag hinge (drag hinge offset) – $\ell_{\text{dr.h}}$;

3) drag hinge stagger – b;

4) flapping compensator angle – α_{k}.

MR hub sketch with hinged blades is shown in Fig. 8.12.

For manufacture of the hub parts it is recommended to use the electroslag remelting steels, which contain a minimal amount of contaminants.

Due to wide use of hingeless main rotor the composite materials are applied as a material for hubs (fiberglass, organoplastics, carbon composites, and combinations thereof).

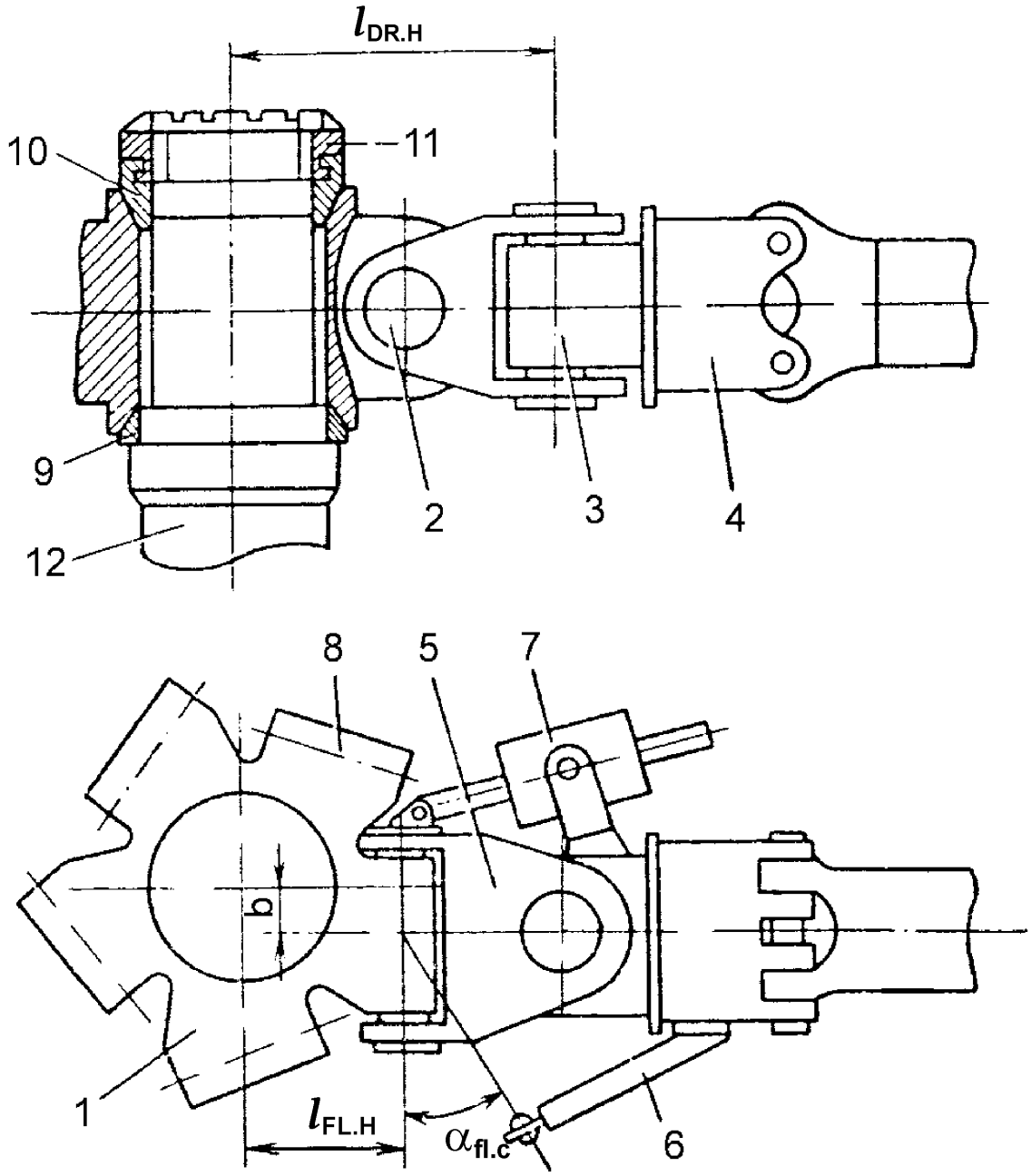


Fig. 8.12. Fully-Articulated Hub:
1 – body; 2 – flapping hinge; 3 – drag hinge; 4 – feathering hinge; 5 – clamp;
6 – blade driving link; 7 – drag hinge damper; 8 – lug; 9 – low ring;
10 – top ring; 11 – nut; 12 – reducer shaft

8.2. TAIL ROTORS

Tail rotor (TR) is designed to balance MR reactive torque, to provide directional control and stability of the single-rotor design helicopter. TR mounted on fin (or inside a fin) outside area of the MR rotation and intensive aerodynamic effect.

Similar to main rotor, TR consists of a hub and blades. Several types of TR schemes are distinguished as per design – traditional (pulling or pushing), X-shaped, fenestron.

Pushing TR is the most widely used. This is due to the fact that loss of traction force of pushing TR is several times smaller than of the pulling rotor. More over the additional dynamic loading of boom with pulsating air stream from TR is excluded.

X-shaped TR is formed by two double-blade rotors, mounted coaxially on the same shaft at a certain angle to each other and at a certain distance from each other. Thus, at the NA-64A helicopter "Apache" the rotors are set at angles of 55° and 125° to each other. This arrangement reduces the aerodynamic interaction of tip vortices of each pair of adjacent blades. This increases the lift-to-drag ratio (increased efficiency) and decreases noise level in comparison with the conventional rotor.

In some cases, fenestron is used on helicopters as TR that is multiblade rotor placed in an annular channel mounted directly in fin (Fig 8.13.). Each blade is attached to the hub by means of feathering hinge. Fin airfoil has certain camber and twist in order to create in horizontal flight forces directed as the fenestron thrust. This design has several advantages: specially designed duct profile allows you to create additional thrust; annular duct provides greater safety for maintenance personnel and maneuvering safety at the ground; fin surface reduces the required power in level flight, axial flow stream in the annular duct simplifies the hub design, placement of fenestron on the tail shaft axis makes it possible not to use intermediate reducer.

The fenestron disadvantages include: much more required power to hover, increase noise and aerodynamic drag in the horizontal flight (due to installation of the annular duct).

Selection of TR scheme is done at stage of the helicopter general design.

8.3. ROTOR DRIVE SYSTEM

The system of helicopter rotor drive - transmission - transmits power from the engines to consumers: main and tail rotors, energy systems (hydraulic, electric, etc.).

The transmission components are determined by the helicopter type and may include: primary and intermediate gearboxes, tail rotor gearbox, coupling and synchronizing shafts with couplings, shaft bearings, cooling fans, free-wheel clutch, rotor brake.

The rotor drive system should meet the following requirements:

in case of failure of any engine in multi-engine helicopter the remaining operating engines should provide drive of each rotor, operation of which is essential for flight and control; tail rotor necessary for control in autorotation of single-engine single-rotor helicopter, must be driven by the main rotor after its disconnection from the engine;

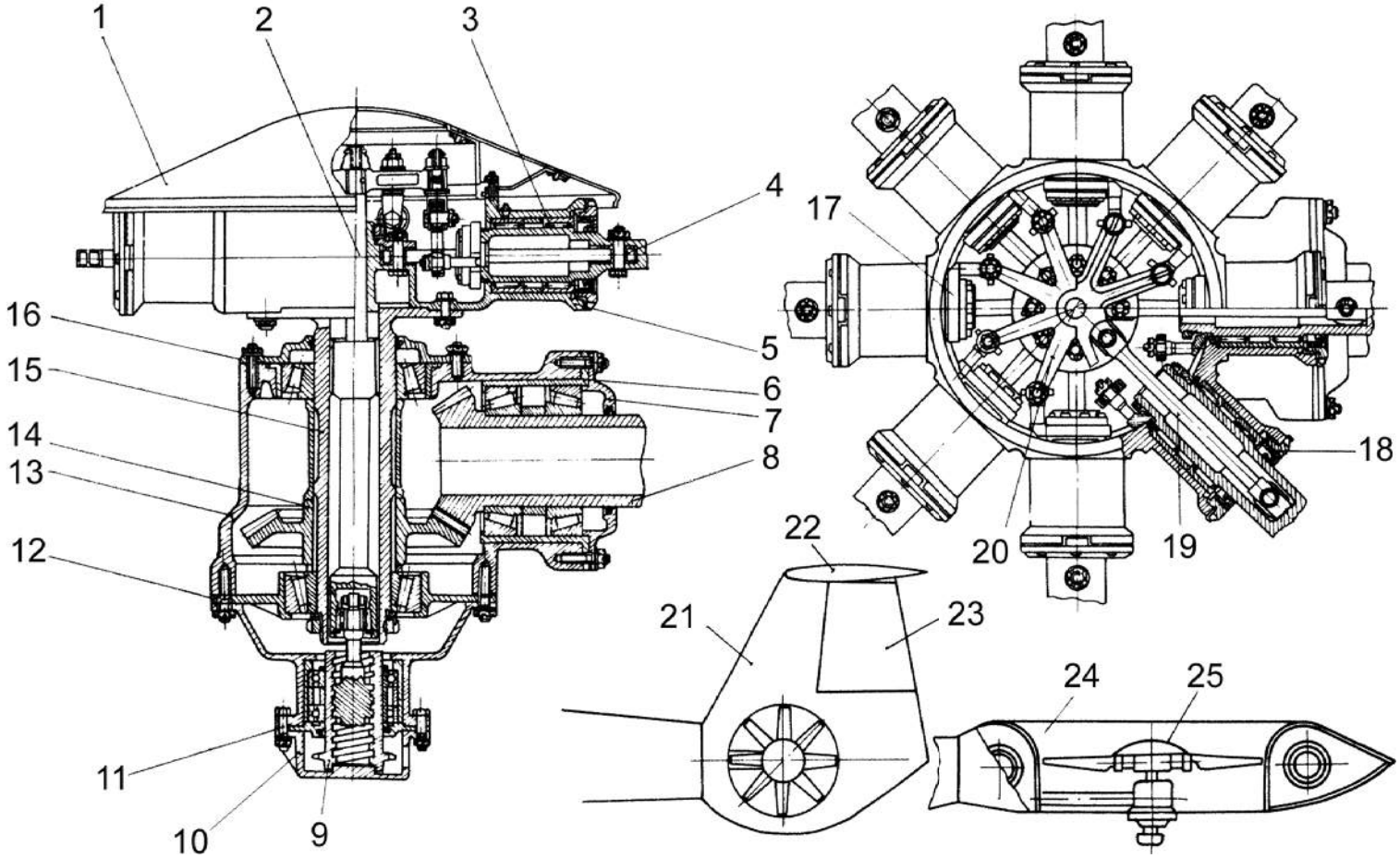


Fig. 8.13. Fenestron:

1 – hub fairing; 2 – control shaft; 3 – bearing; 4 and 19 – torsion; 5 – sleeve body; 6 – steel bushing; 7 – cover; 8 – tail shaft; 9 – control mechanism; 10 – control mechanism cover; 11 – bolt; 12 – membrane with thrust bearing; 13 – reducer body; 14 – driven gear; 15 – shaft; 16 – reducer front cover; 17 – control link; 18 – sealing; 20 – control ″spider″; 21 – tail fin; 22 – stabilizer; 23 – rudder; 24 – fenestron body; 25 – fenestron rotor

each rotor drive system must have one-way clutch for automatic disconnection from the main and auxiliary rotors in case of engine failure;

if the rotor drive system uses device to limit torque that it should be placed so as to ensure long-term helicopter control during the device operation;

if there is a need to set the rotors so as to avoid their crossing, the construction of each drive system should provide continuous reliable phase relationship under all operating conditions;

if there are means for braking the rotor, then management of the system should be independent of the engine;

auxiliary units must typically have a "weak link" to protect the drive system components from excessive torque.

Structural and Functional Diagrams of Helicopter Rotor Transmission

Single-rotor helicopter drive includes (Fig 8.14.): main gearbox (2), main rotor brake (3), tail rotor shaft (4), intermediate shaft (6), intermediate reducer (5), tail gearbox (7), and oil-air radiator fan drive shaft (1). Power from the engine is supplied to main gearbox (2) and then through it distributed to drive main rotor, tail rotor and units to ensure work of various helicopter systems.

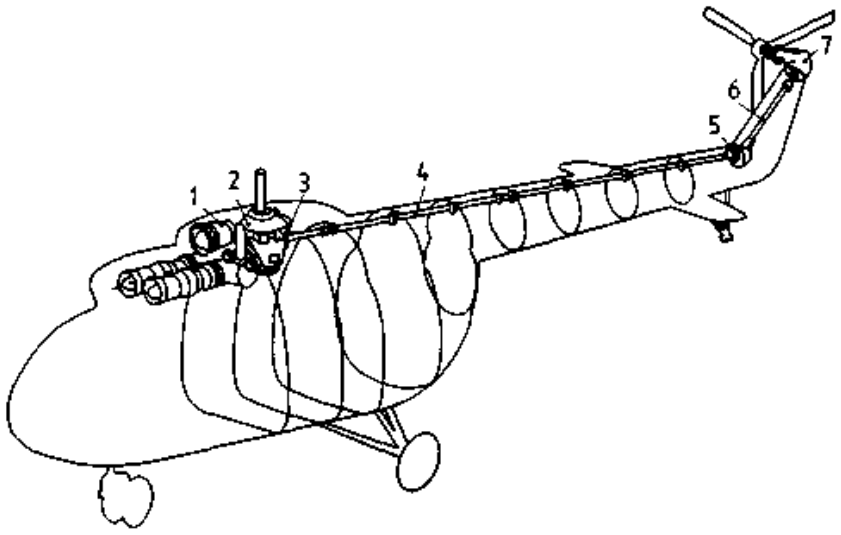


Fig. 8.14. Mi-8 Single-Rotor Helicopter Transmission

More information about transmission units is discussed in the relevant sections of the course.

8.4. CONTROL SYSTEM

Main helicopter control system is designed to change the trajectory of helicopter motion, its balancing and stabilizing at predetermined flight conditions.

The helicopter trajectory changes under action of forces and moments generated by main and tail rotors of single-rotor helicopter, or by main rotors - for double- and multiple-rotor configurations.

Control of forces and moments created by main and tail rotors is carried on four control channels - longitudinal, transverse, and vertical track.

Channels of longitudinal and lateral control are associated with MR cyclic pitch control stick, directional control channel is linked with pedals, vertical movement channel is connected to collective pitch control lever.

Today's helicopter control system is a set of mechanical, hydraulic, electrical and computing devices that provide the following tasks:

- helicopter flying (change in flight paths) by pilot in manual and semi-automatic modes;
- automatic helicopter control within the flight modes and stages provided by TTR;
- creating sufficient capacity to move controls;
- ensuring required characteristics of helicopter stability and controllability;
- stabilization of set modes of flight;
- improving flight safety by timely warning of crew about approaching dangerous (in speed, altitude, overload, angle of attack, glide, roll, and other parameters) flight mode and issuing, as far as possible, commands to move controls preventing these modes.

Helicopter control system development is started after determination of its major weight and aerodynamic parameters. Controls are selected and checked (by calculation or experimentally – in wind tunnel) for their effectiveness under any possible flight condition. Then, required travel of controls and their hinge moments are calculated for various flight conditions, including emergency landing in autorotation. Maximum necessary travel of the controls and their balancing travels for different modes of horizontal flight are determined.

Simulation of the motion processes of the helicopter under design by means of analog computers allows to determine necessary means of automation control system, set limits of the parameter change for the elements, which provide stability of the control loop.

On the basis of travel of controls in various maneuvers, as well as on conditions of the control loop stability the required characteristics of kinematic transmission from levers to the controls and features of loading the controls are found. |When calculating required loading characteristics also take into account the requirements for limiting the maximum deflection angle of controls under certain flight conditions imposed by the terms of the strength of the helicopter structure.

All of the above is the basis of the control system design.

In general, for the structural development of the control system elements it is necessary:

1) Determine components of the control system and its elements;

2) Receive basic data for system design:

- Balancing travel (balancing characteristics) of controls (main rotor and tail rotor) in various flight conditions, including failure conditions;
- Required travel and speed of the controls changeover in maneuvering and stabilization (to determine disposable overload characteristics of acceleration and braking, and turns);
- The amount of hinge moment change on controls for different flight modes;
- Inertial characteristics of the controls;
- Required characteristics for controls damping to provide required flutter characteristics;

3) Develop control system diagram with inclusion of actuators of feel systems, automation and steering;

4) Obtain kinematic relations between controls deviations and control sticks deviations;

5) Determine characteristics of loading the control levers;

6) Perform calculations of characteristics and parameters of servo drive of control surfaces;

7) Develop the concept and perform calculations of the hydraulic power booster capacities, including various cases of failure;

8) Select principal or block diagrams, and laws of automatic system operation. Based on the amount of problems solved on helicopter, select the optimum gear ratios from autopilot in stabilization mode;

9) Develop arrangement of levers and other control elements in the cabin;

10) Coordinate placement of control system main units with structural elements and other helicopter systems.

Only then you can proceed to development of constructive elements of a helicopter control system (for example, mechanical linkage elements, artificial feel mechanisms, installation of hydraulic system, etc.).

Final step in the development of control systems is production of stand units for comprehensive perfection of control system, implementation of the relevant improvements, training the helicopter crew for the first flight and testing of piloting techniques.

The stands and the first manufactured helicopter are used to finalize the control system and to determine frequency characteristics of the system and individual units, to eliminate possible fluctuations, and to conduct research under various failures.

Final finishing of the helicopter control system is executed on the basis of flight tests.

Fig. 8.15 shows typical control system used in the Russian single-rotor helicopters.

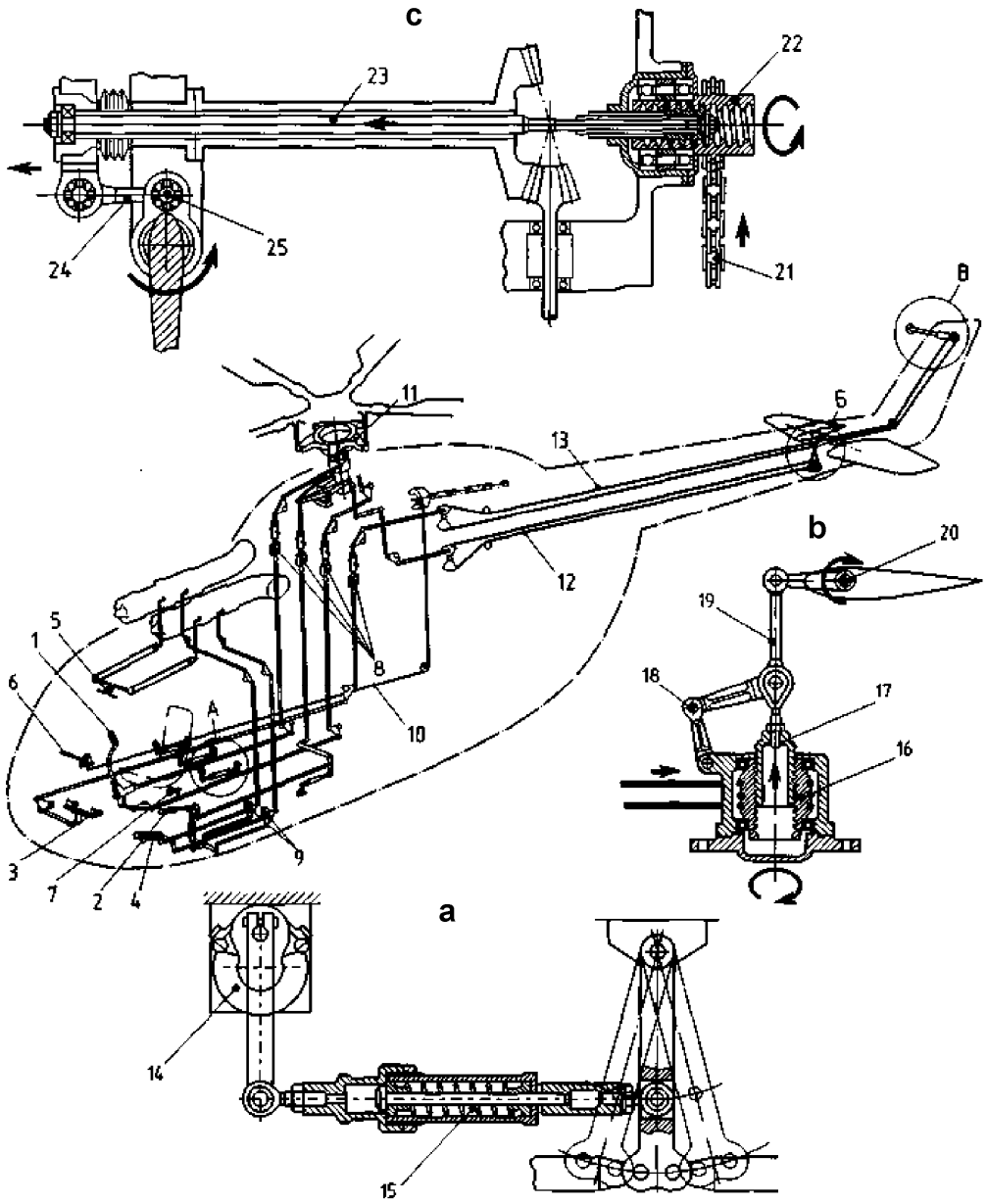


Fig. 8.15. Typical Control System of Single-Rotor Helicopter:
1 – cyclic pitch control stick; *2* – collective pitch control lever; *3* – pedal; *4* – engine throttle levers; *5* – engine stop lever; *6* – main rotor brake lever; *7* – fire cock handles; *8* – hydraulic boosters; *9* – differential mechanism for engine control; *10* – cable control linkage of main rotor brake; *11* – swash plate assembly; *12* – stabilizer control linkage; *13* – tail rotor control linkage; *14* – electromechanical coupling; *15* – spring mechanism; *16* – rotating drum with cable secured on it; *17* – translational rod; *18* – torque-link; *19* – rod; *20* – axle about which the stabilizer is deflected; *21* – roller chain; *22* – sprocket rotating body; *23* – piston making translational motion; *24* – rod; *25* – blade axis of rotation; *A* – attachment fitting of load feed mechanism in control system; *B* – control screw mechanism of stabilizer setting angle; *C* – control of blade setting angle of tail rotor blades

Some Fields of Control Systems Development

Todays fly-by-wire control system (FBWCS) come to replace mechanical systems.

Creating such systems opens up new possibilities for improving quality characteristics of helicopter stability and controllability.

In general, today's helicopter control system must include: fly-by-wire control system (FBWCS), mechanical part of the control system (MCS) and system improving stability and controlability (CISC or ACS - automatic control system) that is part of flight and navigation complex.

Even now it is possible to use single multiple redundant automatic control system (AFCS) that permits the helicopter flying in different modes, up to the automatic approach. This unified control system can be called integrated control system.

FBWCS systems have already been installed on the S-70 helicopters, Black Hawk, RAH-66 "ANSAT" and others

8.5. FRAMED UNITS

Fuselage is the helicopter body and accommodates crew, passengers, cargo and equipment; connects main rotor, wing, control surfaces, landing gear and power plant (Fig. 8.16). Fuselage geometrical parameters and configuration are defined by the helicopter purpose and its aerodynamic configuration.

Helicopter ability to perform any movement in space was the reason for its use as a crane helicopter, capable not only to transport cargo, but also to provide erection work on installation of the cargo on a given object. The operating procedure of helicopter-crane defines the shape of its fuselage. The fuselage has become a power beam, designed for securing the fuselage cabin and components of external rope suspension or freight containers (Fig. 8.17).

Helicopter of the flying boat type has fuselage made with special shape of its ventral part – boat. It must meet the requirements of fluid dynamics – has minimum load at landing a helicopter on the water and needs minimum rotor thrust in the vertical take-off (Fig 8.18.). The boat front portion must have devices (bilges), excluding formation of splashes in the pilot's field of view and engine air intakes. Amphibian helicopter must have the required stability, buoyancy and be provided with wheeled landing gear for landing on the ground.

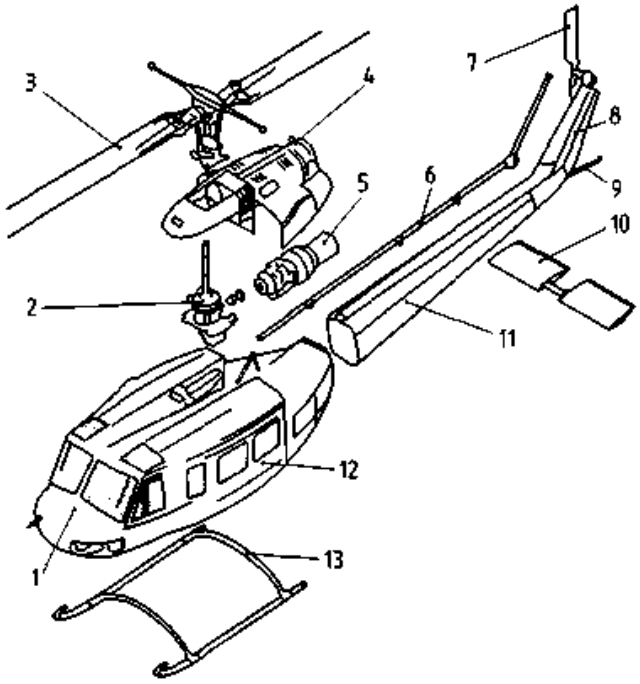


Fig. 8.16. Single-rotor Helicopter Arrangement
with Main Rotor Mechanical Drive System:
1 – flight compartment; *2* – main reducer; *3* – main rotor; *4* – engine fairing;
5 – engine; *6* – tail rotor shaft; *7* – tail rotor; *8* – fin; *9* – tail bump; *10* – stabilizer;
11 – tail boom; *12* – passenger (cargo) cabin; *13* – landing gear

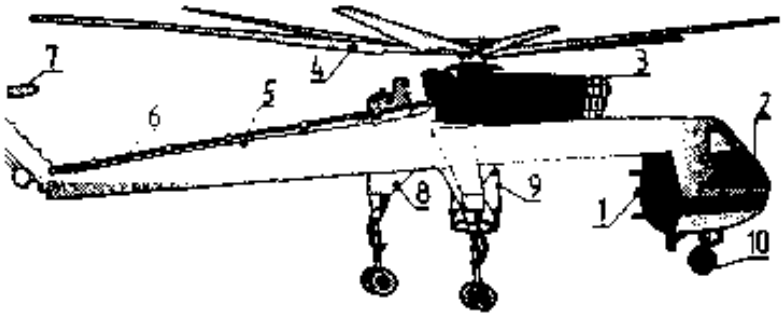


Fig. 8.17. Fuselage of Helicopter-Crane S-64A:
1 – operator cabin; *2* – pilot cabin; *3* – power plant; *4* – main rotor,
5 – fuselage load-carrying beam; *6* – tail rotor shaft;
7 – tail rotor; *8* – main landing gear legs; *9* – cargo external sling; *10* – nose LG

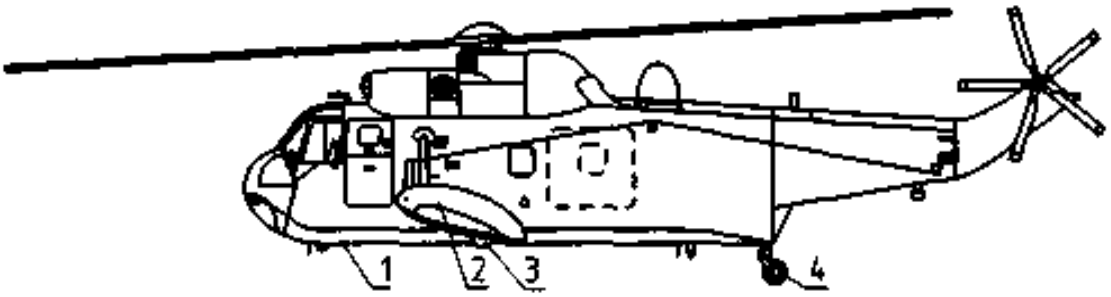


Fig. 8.18. S-61 Amphibian Helicopter Fuselage:
1 – fuselage boat bottom; *2* – float fairing;
3 – main landing gear (retracted into floats); *4* – rear landing gear

For shipment cargo inside fuselage special transport helicopters are developed (Fig. 8.19). The design of such helicopter fuselage provides loading and unloading over cargo ramp through the cargo hatch inside the cargo compartment. Cargo compartment floor has load-bearing structure that provides perception of loads from the wheeled and tracked vehicles, containers, etc., and their fixing in position. These helicopters can transport loads with dimensions exceeding the dimensions of the cargo compartment, on the external rope sling. Such helicopters are provided with ability to perform erection operations in hovering.



Fig. 8.19. Mi-26 Helicopter

Ka-26, Ka-126, Ra-226 multi-purpose helicopters are so called flying landing gear. The helicopter fuselage (Fig. 8.20) is a platform, which is used for attachment of cockpit, main gearbox with coaxial rotors, engines and landing gear, as well as for installation of various modules of proper purpose: passenger cabin, equipment for agro-chemical works and other devices. The fuselage is designed as a rectangular platform, which carries all the units and equipment necessary for helicopter mission.

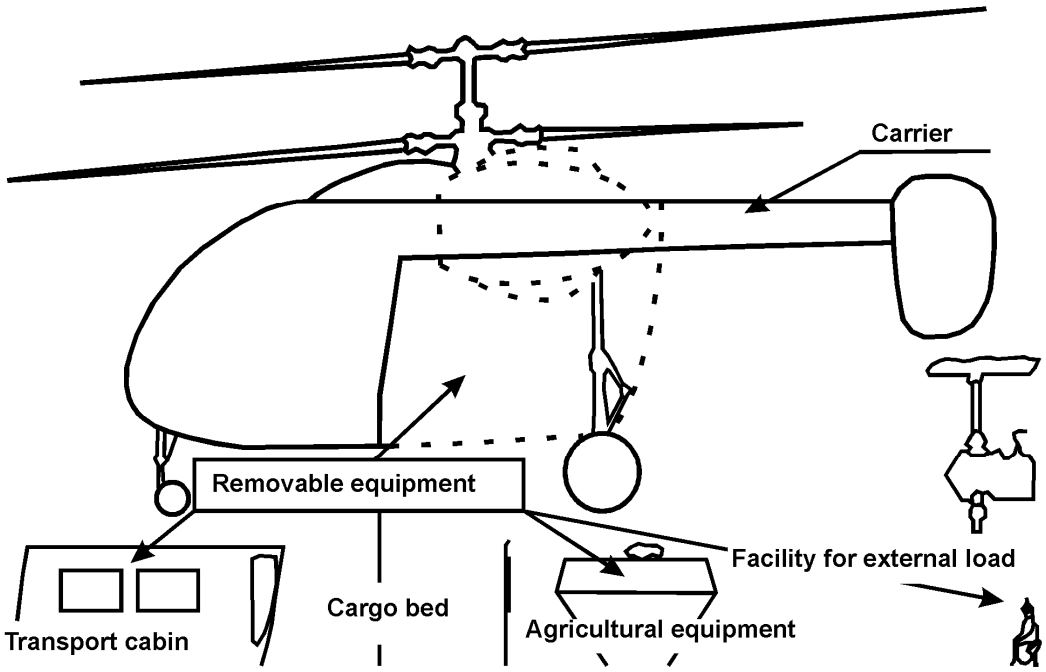


Fig. 8.20. Fuselage of Multi-Purpose Helicopter Ka-26

The fuselage design of military helicopters has its specificity on ensuring survivability of framework strong components, crew survival, evacuation of the helicopter crew during the emergency landing. It is used safely damageable cabin floor panels, devices, absorbing impact energy of the helicopter.

Light helicopter fuselage (Fig. 8.21) is a cab, which accommodates pilot and passengers. Its primary structure is a platform, which is used to attach seats, elements of the control system, main gearbox with rotor, power plant, boom with tail rotor and skid landing gear.

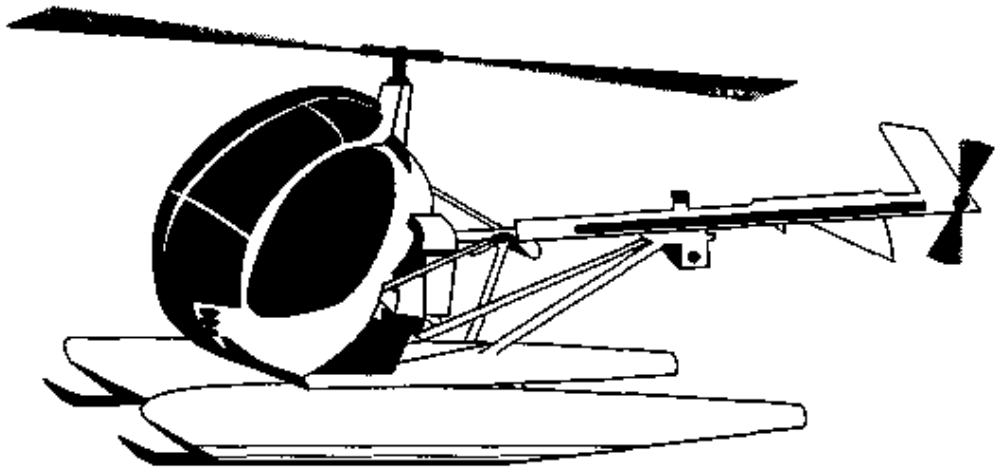


Fig. 8.21. Fuselage of Schweizer (Hughes) 300C Light Helicopter

Design of the passenger helicopter fuselage is largely defined by the level of sanitary and technical requirements applicable to interior of the passenger cabin, requirements to emergency exits for passengers, heat and sound insulation and ventilation.

Helicopter construction uses the experience gained in aircraft development of load-carrying structures of frame units. Helicopter fuselages are produced on the basis of beam, truss or mixed schemes. Fuselage structure strong elements such as spars, stringers, frames (normal, strong, joining), skin have the same function as those on the fuselage of the airplane.

Methods of optimization of frame units compressed and stretched panels have a common criterion for aviation – the minimum weight of structure while meeting the durability, stiffness, survivability, resource, technology, operational efficiency requirements, etc. Methods of designing of frame units are detailed in [2].

In helicopter engineering the beam-type fuselage is widely used. The beam structure uses internal volume of the fuselage as much as possible, all the requirements of aerodynamics and technology are provided.

Beam fuselages are divided into two types - spar and manoblock.

A variant of *fuselage monoblock structure* is represented by monocoque scheme. A monocoque is made of homogeneous material (aluminum alloys, composites) and provides for skin and frames. All the forces and moments are taken by skin. Most often the monocoque is applied for helicopter tail booms. Multilayer monocoque has sandwich panels with thin layers of carriers, reinforced by honeycomb filler. Structure of sandwich panels is very flexible and defined by the materials of the outer and inner layers, by the filler type, method of connection of the outer layers with the filler.

Possibility of using thin skins, well supported with honeycomb filler having low density, makes honeycomb structures a reserve for reducing the fuselage mass. High specific strength and resistance to vibration and acoustic loads define the ever-growing use of honeycomb structures as load-bearing elements of helicopter fuselages of any weight category.

In the truss-type fuselage the strong elements are represented by truss caps: spars, supports and bracing in the vertical and horizontal planes. Truss structure is used as the main strong element of the light helicopter fuselage and tail boom (Fig. 8.22).

The problem of a choice of rational fuselage load-bearing structure is solved by indirect methods involving weight statistics, parametric relationships and information about LBS of previous designs. In most cases, the fuselage structure type is selected based on the requirements set for the helicopter, conditions of operation and production capabilities. The problem is reduced to finding the best variant among the specified construction type.

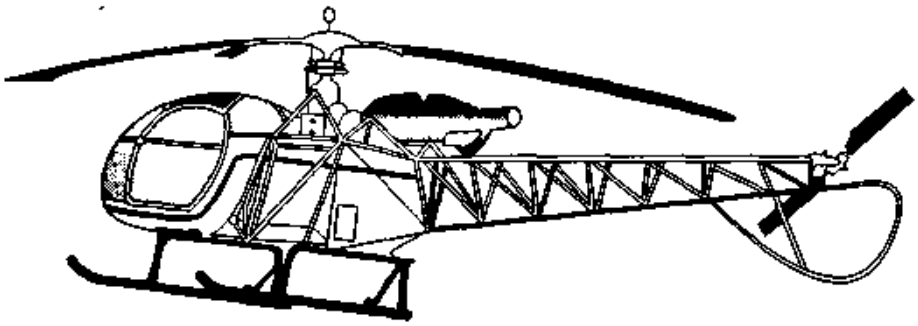


Fig. 8.22. Fuselage of AS-315 «Lama» Helicopter with Truss Tail Boom

Aerodynamic asymmetry of the rotor with oscillating blades is a source of helicopter vibrations. Vibrations are transmitted to the fuselage through the main gearbox mounting components. Elimination or attenuation of vibrations has the task of:

ensuring comfort of crew and passengers;

reducting fatigue loads on structure;

protecting against damage to sensitive electronic equipment placed onboard the helicopter.

It is impossible to eliminate vibrations by appropriate selection of parameters of blade, rotor hub, rotor blade control system, MR shaft parameters and its material, application of dampers on the rotor hub and in attachment points of the main reducer to fuselage elements because the fuselage structure reaction changes under various flight conditions. Therefore, construction of fuselage load-bearing elements is done with taking into account requirements of fatigue resistance [20]. Concentrators of forces and stresses in the joints are subject to elimination, fretting-corrosion is not allowed in strong joints, crack resistant (viscous) material to be applied.

In practice of Russian helicopter industry the bonding-welded connection of fuselage skin to frame elements is widely used (helicopters Mi-2, Mi-24, Mi-26, and others.). This connection reduces stress concentration around the welding spot due to its partial relieving by adhesive bonding. In addition, presence of adhesive layer between skin and stringer flange eliminates chemical corrosion caused by condensed moisture.

Fiber composite materials are widely used in serial and prototype production of helicopters. In some experimental helicopters fuselage is completely made of sandwich panels with the use of elements produced by winding glass, graphite and organic filaments in various combinations (Fig. 8.23).

An advantage of composite materials compared with orthotropic ones is possibility of positioning the reinforcing material in the direction of the applied load, efficiently using material mass and changing rigidity of the strong element with the purpose to prevent resonant modes of oscillation.

Composite materials have high fatigue resistance and dynamic elastic hysteresis loop, as a result high-frequency oscillations are damped by structural elements.

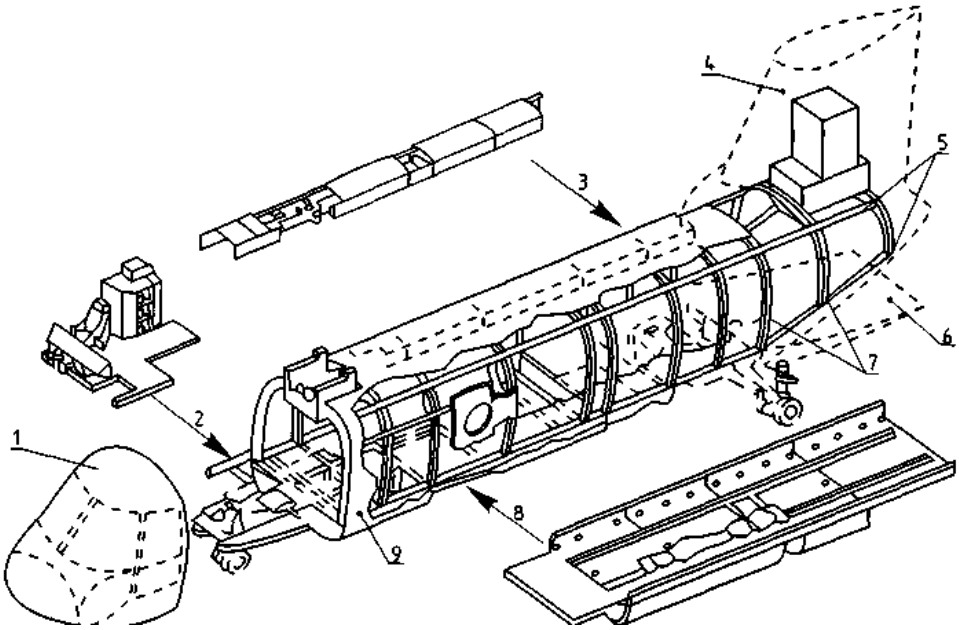


Fig. 8.23. Fuselage of Boeing-360 Experimental Helicopter:
1 – flight compartment; *2* – MR front reducer; *3* – transmission shaft fairing;
4 – fin-pylon; *5* – fuselage longitudinal elements;
6 – cargo ramp; *7* – frames; *8* – fuselage lower panel; *9* – three-layered panels

Wing, horizontal and vertical tail unit of a helicopter. Helicopter wing can be installed for the following tasks:

Reducing loads acting upon main rotor of single-rotor helicopter in order to increase flight speed. At the same time it can serve as a wing mast for hanging cargo on pylons;

Securing and unloading of main rotor, attachment of landing gear and placement of fuel tanks on rotary-wing rotorcraft of side-by-side configuration (Fig 8.24.);

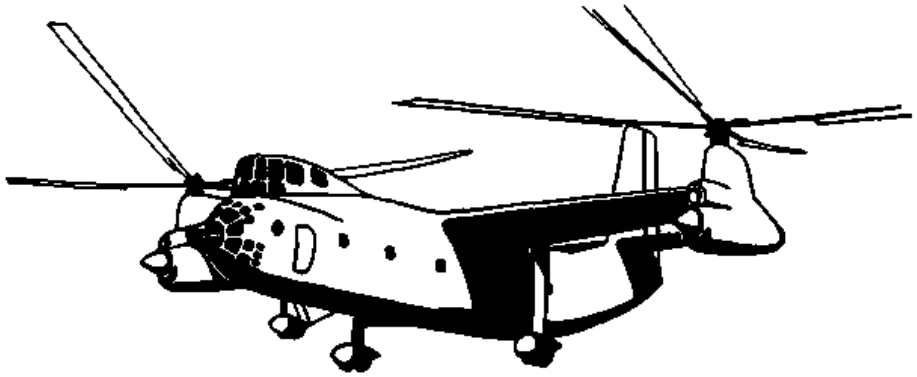


Fig. 8.24. Side-by-Side Rotorcraft Ka-22

Installation of rotor and generation of lift in level flight at the convertible aircraft (Fig. 8.25).

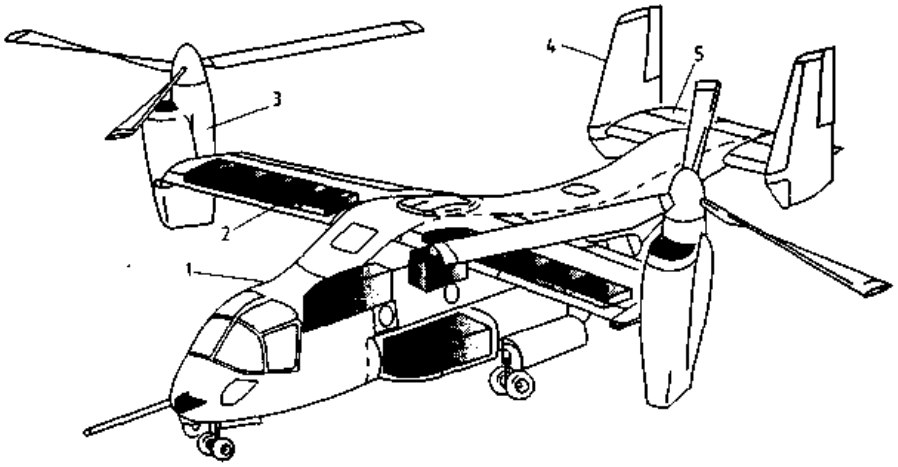


Fig. 8.25. Convertible Tilt-Rotor Aircraft of Side-by-Side Configuration V–22 «Osprey»:
1 – fuselage; *2* – wing; *3* – tilted rotors with power plants;
4, 5 – vertical and horizontal tail units

Depending on the wing purpose the magnitude and nature of the load on its strong elements change.

The nature of loading the wing panels of convertible rotary wing aircraft of side-by-side configuration (e.g., experimental aircraft V-22) when being operated as helicopter and under transient conditions necessitated to prevent arising of aeroelastic vibrations of various forms in the wing - rotor systems. One of the possible rational solutions to this complex problem is to use fiber composite materials in the zones of the wing box.

All the principles of development of load-bearing structure of cantilever wing are fully applicable to the design of stabilizer and fin.

Stabilizer works similar to a wing.

In the process of calculating the longitudinal balancing of the helicopter with the extreme front and rear CG position a decision is made whether to make changes in stabilizer setting angle in flight or on the ground.

Fin is a cantilever beam loading the tail boom with twisting and bending. Single-rotor helicopter fin is made arrow shaped for layout reasons.

Tail rotor gearbox housing is mounted at the fin top. The tail rotor shaft passes inside the fin beam. For its installation and control the fin beam is provided with hatches or detachable panels, so the fin LBS type can be: single-spar with rear wall or double-spar. Fin beam is attached to the strong elements of the tail boom similar to joining the wing with wing center section.

If the tail rotor shaft for reasons of layout can be placed in the fin nose, then fin nose does not take loads and is made in a form of fairing, and fin LBS is executed of the torsion box type.

Tail unit of coaxial helicopter consists of a horizontal unit (stabilizer and elevator) and vertical unit (fin with two rudders) and is intended to improve characteristics of longitudinal and directional stability, as well as for helicopter directional control with the help of rudders on the flight modes with forward speed. The efficiency of rudders increases with increase of speed of horizontal flight. To obtain high characteristics of helicopter directional stability on the flight modes with forward speed the vertical unit fins are set at an angle to the helicopter longitudinal axis with tails outwards.

8.6. LANDING GEAR

Helicopter landing gear under the scheme and the design is very similar to aircraft landing gear and performs similar function: it provides parking and movement of the helicopter on ground during run-up before take-off, run after landing, taxiing and towing. The landing gear receives load and dissipates most of helicopter kinetic energy when landing.

The following requirements are set to helicopter landing gears:

- possibility of free, stable and controlled movement of the helicopter on ground during run-up, taxiing and run;
- absorption and scattering of energy of hits when landing and moving on ground with soft damping of helicopter under relatively small g-forces;
- damping of "ground resonance" oscillations (spontaneously occurring oscillations of the helicopter on ground with increasing amplitude);
- minimum drag (if landing gear is not retracted during flight).

Helicopter landing gear can be wheeled, skid, float, boat (Fig. 8.26). Some helicopters used combo landing gear. For example, an amphibious helicopter landing gear is a boat joined with wheel landing gear, and wheeled landing gear of the ship-based helicopter is equipped with balloons intended to perform an emergency landing on water. On most helicopters wheeled and skid landing gears are used. Wheeled landing gear is used for all weight classes of helicopters. Skid landing gear has a simple structure, less weight and aerodynamic drag, but impossibility of helicopter landing with run, takeoff with run and taxiing to takeoff makes appropriate its usage only for small helicopters.

Wheeled landing gear. Helicopter with wheeled landing gear in the parking lot has three (three-wheeled landing gear) or four (four-wheel landing gear) points of

support. For three-wheeled landing gear two support points are arranged symmetrically relative to the longitudinal axis of the helicopter fuselage near its center of mass and they are named main landing gear. The third point of support is located at the axis of the helicopter fuselage front of the main LG – it is the nose landing gear or behind main LG – it is the tail landing gear.

Four-wheel landing gears, unlike the three-wheel LG are equipped with two nose LG. Single-rotor helicopter at the end of the tail boom has safety support with damper in order to protect against impact of the intermediate reducer and tail rotor on ground when helicopter lands in autorotation.

Most helicopters use landing gears with nose wheel, as this scheme provides safer and easier helicopter landing in poor visibility conditions, good directional stability during the run-up and run.

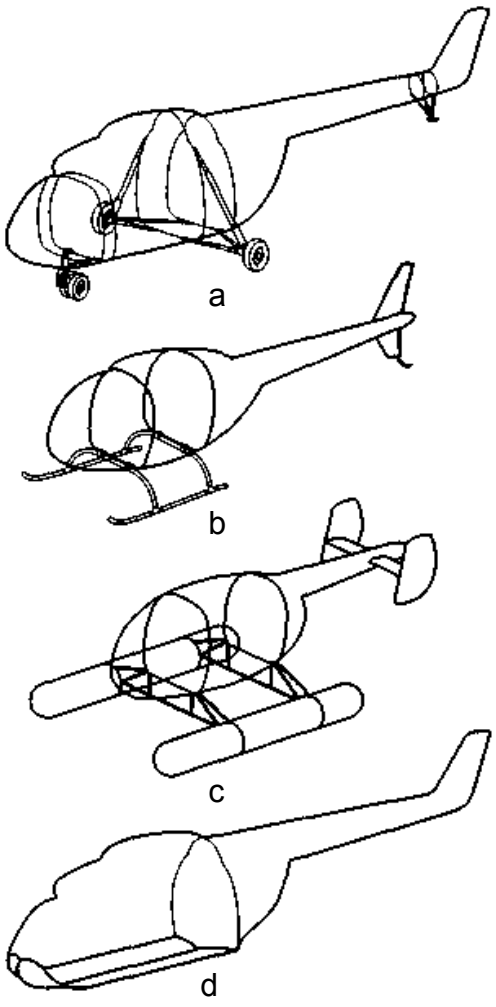


Fig. 8.26. LG Classification:
a – wheeled; *b* – skid; *c* – float; *d* – boat

Wheeled tail-wheel landing gear is often used on military helicopters, because the nose wheel prevents placement and operation of the weapon mounted in the helicopter nose. Four-wheel landing gear has to be used when the helicopter center of mass is quite high off the ground.

Basic parameters of LG with nose-wheel (Fig. 8.27):

- distance from the nose wheel to the helicopter center of mass a;
- distance from the MLG wheels to the helicopter center of mass b;
- wheel base *c;*
- wheel track B*;*
- capsizing angle θ;
- LG height h;
- anti-nosing-over angle γ.

The distance from MLG wheels to the helicopter center of mass is chosen based on the fact that these wheels shall take about 85 ... 90% of the helicopter weight. This load balancing provides the helicopter longitudinal stability and directional control when maneuvering. In addition, the value of b should prevent the helicopter nosing-over when loading cargo through the rear hatch, standing on the main wheels and moving on the ground. If the base c is short, then during taxiing the helicopter will strongly oscillate in the longitudinal plane. If the base c is too long, then the nose landing gear would take very small load and run after landing should be unstable.

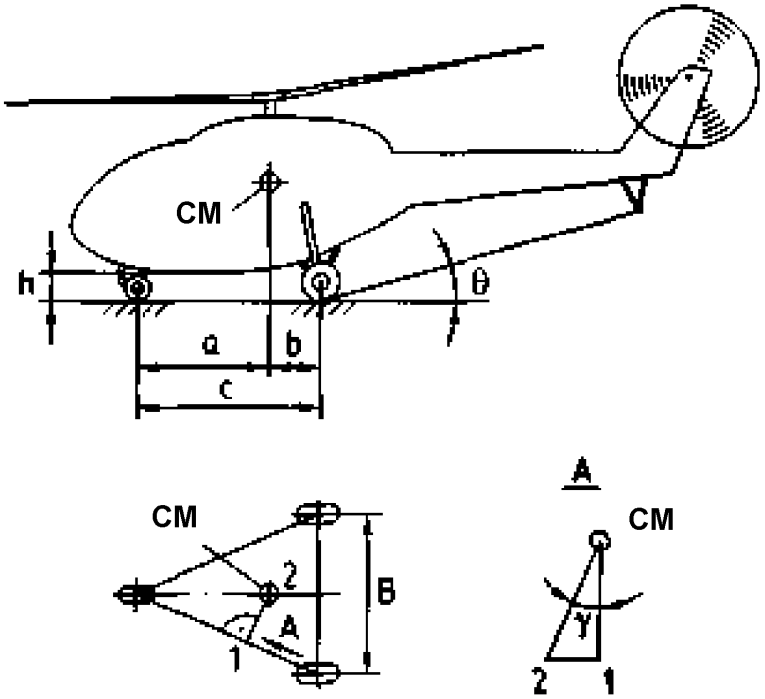


Fig. 8.27. Nose-wheel LG Basic Parameters

The minimum required wheel track B is determined based on the helicopter lateral stability. When center of mass is located too high then helicopter becomes sensitive to shocks on wheels when moving due to increasing yaw moment. Small track does not provide helicopter lateral stability. Capsizing angle θ is determined from the conditions of safety landing in autorotation.

Typically, anti-nosing-over angle γ = 35...40°. High center of mass position in order to limit the track B and to keep values of γ within these limits you have to use two nose legs. For this reason, four-wheel landing gear is used on the majority of coaxial-rotor helicopters. Landing gear height h must be at least 200 mm to allow helicopter movement over rough terrain.

Wheeled landing gear can be retractable and non-retractable. Non-retractable LG has less weight. Its weight is 2.5 ... 2.8% of the helicopter take-off weight. Retractable landing gear weight exceeds 3% of the helicopter take-off weight.

The wheeled LG design depends on the wheel position relative to the leg attachment fitting on fuselage, place of retraction, length of LG leg and retraction kinematic scheme. Landing gear may be of truss, beam or levered design.

Many helicopters (Mi-2, Mi-6, Mi-8) [4] with non-retractable landing gear use three-bar (pyramidal) truss scheme (Fig. 8.28, a). In this scheme beam (4) and rod (3) are hinged attached to the strong members in the bottom fuselage part in fittings (1) and (2). They take lateral and frontal loads of the wheel. The helicopter kinetic energy during landing is taken by shock strut (8), built in the third rod of truss (7). While compression of shock struts wheel axle (6) moves along a circular arc. The wheel in non-compressed state should have camber angle φ to provide that at the end of compression the wheel had maximum contact area with the ground. It must not exceed the value, which makes possible tire failure in the process of shock-strut compression using lateral force when landing gear track changes.

Pyramidal-parallelogram truss scheme (8.28, b) best meets operating conditions of the shipborne helicopters (Ka-25, Ka-27, Ka-32) [4]. It allows to get large wheel travel in the vertical direction required for damping helicopter vibrations at pitch and rolling motion of the ship. Track of wheels at the same time does not change much.

Beam design of the main LG (Fig. 8.29, a) is used, if the helicopter airframe layout allows not to use the LG truss design.

To reduce the size of the shock strut under limited height of the landing gear leg the levered wheel suspension is used. One of the main advantages of the levered suspension of wheels lies in mitigation of frontal impact. Levered strut has several kinematic schemes.

The scheme is often used for the main legs of non-retractable landing gear with tail wheel on military helicopters. To ensure the required track the lever-to-strut suspension pivot is displaced laterally relative to the lever attachment fitting to the fuselage strong member.

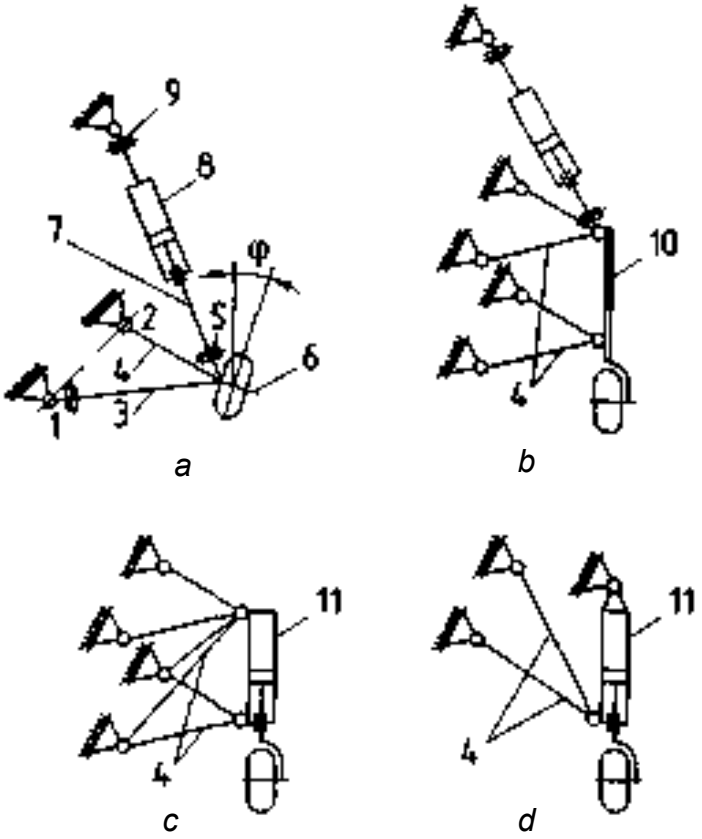


Fig. 8.28. Load-Bearing Structure of Truss-Type LG:
a – three-rod piramidal; *b* – piramidal-parallelogram; *c* – parallelogram with vertical strut;
d – piramidal with vertical strut; *1, 2* – fuselage attachment fittings; *3* – beam;
4, 7 – truss bars; *5, 9* – cardan joints; *6* – wheel axle; *8* – shock absorber;
10 – MLG leg; *11* – MLG leg with shock strut

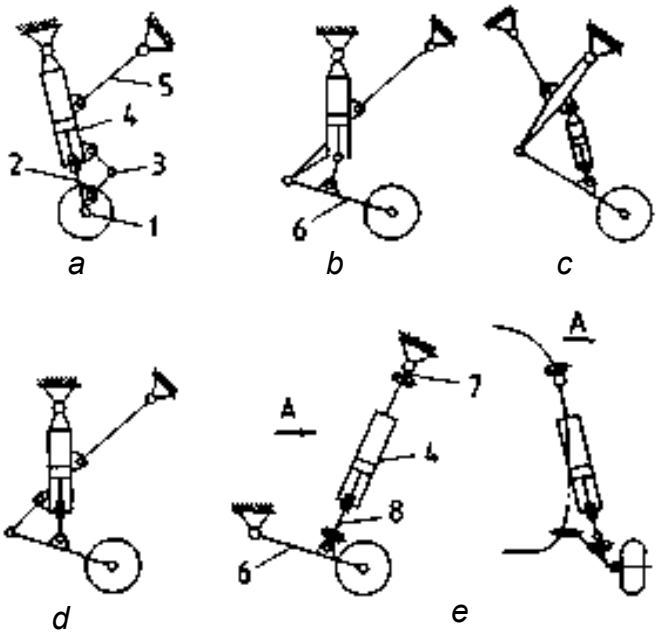


Fig. 8.29. Load-Bearing Structures of Beam-Type and Levered-Type Landing Gear:
a – beam type; *b, c, d* – levered; *1* – wheel axle; *2* – shock strut piston;
3 – torque link; *4* – shock absorber; *5* – brace (or LG retracting cylinder);
6 – lever; *7, 8* – cardan joints

During helicopter landing its landing gear must absorb kinetic energy, which is possessed by the helicopter because of presence of horizontal V_x and vertical V_y speed components. Kinetic energy is defined by the horizontal speed V_x and absorbed mainly by brakes of landing gear wheels. The kinetic energy due to the presence of the vertical velocity is absorbed by wheel tires and shock struts.

Shock struts used for this purpose are similar to aircraft landing gear shock struts in design and properties.

Helicopter wheel tires are selected by the value of the parking load P_{st} using aircraft wheels catalog. Thus consider a helicopter landing mild conditions (much smaller landing speeds). Design parking load on the wheel is taken by 30% less than indicated in the catalog. To allow helicopter to move forward on a dirt landing site, the depth of the wheel track must not exceed 6 ... 8 cm.

Semi-bottle tires and high pressure tires are often used on helicopters. Select the tire type depending on the helicopter operating conditions. Helicopter with low-pressure tires when driving on not-dense loose soil instead of taxiing can make liftoff maneuvers.

Therefore, wheel tires of semi-bottle type with a pressure of 0.5 ... 0.6 MPa are often used on helicopters. Wheels with high-pressure tires, as well as the arched wheels have high rigidity and are used on the shipborne helicopters.

In addition to the absorption and scattering of the helicopter kinetic energy during landing helicopter LG are involved in damping of ground resonance. The source of these fluctuations is unbalanced centrifugal force of the rotor. It can arise when the main rotor hub has drag hinges allowing blades perform angular movements in the plane of rotation.

To exclude origination of ground resonance it is necessary to provide sufficient vibration damping both of the helicopter on landing gear, and the rotor blades relative to the drag hinge. Possibilities of damping the rotor blades are limited, since the blade damper operates in forward flight and load the blade root with variable bending moment. Excessive damping of the blades, while maintaining strength conditions requires extra weight of the blades and rotor hub. Therefore, the exclusion of helicopter ground resonance is mainly provided by selection of the damping properties of the landing gear shock struts.

Theory and calculation of ground resonance are given in [3]. Fluctuations in helicopter ground resonance occurs mainly in the transverse plane due to the fact that the helicopter fuselage length is substantially greater than width and, therefore, the helicopter moment of inertia about the transverse axis is significantly greater than moment of inertia about the longitudinal axis. In the event of ground resonance two forms of transverse vibrations are of practical importance.

When helicopter vibrate with the first form of oscillations its center of mass displaces laterally and rotate the helicopter at a certain angle with respect to the center

of mass. This happens when stopping the helicopter rotor after landing or at the beginning of the rotor spin when the rotor takeoff thrust is close to zero, its speed of rotation is low and the load on the landing gear shock struts are practically equal to parking values. In this situation the whole gear depreciation system (shock struts with tires) works, which can be used effectively to eliminate ground resonance.

The helicopter vibrations with the second form, originate relative to its center of mass without displacement. They arise prior to helicopter lift-off or at the initial moment of its landing, when the main rotor thrust is close to the force of gravity acting on the helicopter. In these modes, the shock strut is excluded from the process of depreciating the helicopter vibrations and acts as a rigid rod. Since the damping properties of the tires are low, then total damping of landing gear in these modes is close to zero. Therefore, the helicopter oscillation frequency for the second form is considerably greater than on the first.

Radical way to eliminate ground resonance could be selection of landing gear stiffness characteristics, under which helicopter natural vibrations with the first and second forms (where possible ground resonance) exceed the operating range of rotor rotation frequency. However, in the actual design of the landing gear it is not possible due to tire low elasticity. There is a tendency to transfer to helicopter low frequency vibrations with the first form to prevent resonance. This requires shock struts with reduced rigidity, which leads to excessively large strokes of the shock strut and worsens its damping properties when the helicopter landing. However, to prevent resonance on the second form, there is a tendency to increase the tire stiffness and provide sufficient damping of the shock strut at the very beginning of its stroke.

This contradiction is resolved in two ways: by application of double-chamber shock struts in the construction of main landing gear; by application of shock strut with a special valve, which provides the desired change of the shock strut stiffness characteristics during its compression.

In the double-chamber shock strut (Fig. 8.30, a) one chamber (of low pressure) is designed for damping helicopter vibrations at low loads on the landing gear when the rotor thrust is approximately equal to the helicopter weight, and the second chamber (of high pressure) is designed to absorb high loads on the landing gear in the helicopter landing. The two-chamber shock strut is long, it is convenient to use when there is no limitation on strut dimensions.

The second method uses a single shock strut, which accommodates special spring-loaded valve (Fig. 8.30, b). It opens only when the compression force in the shock absorber exceeds some critical value P_{cr}. When the load on the shock absorber is less than P_{cr} then holes in the shock strut piston rod are open. The hole sizes are chosen from the condition of ground resonance. For loads greater P_{cr} valve opens, and

it opens additional holes of larger diameter, the size of which is selected from the conditions of the helicopter landing.

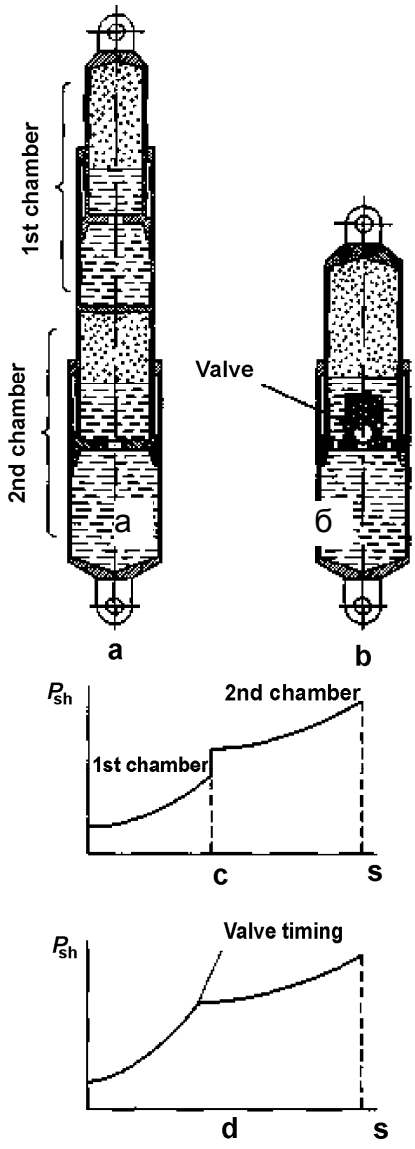


Fig. 30. Helicopter Shock Struts:
a – double-chamber; *b* – single-chamber with valve;
c – typical diagram of static compression of double-chamber shock strut;
d – typical diagram of static compression of single-chamber shock strut with valve;
p_{sh} – pressure in shock strut chamber; *S* – shock strut piston stroke

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Chapter 9. POWER PLANTS OF HELICOPTERS

9.1. CLASSIFICATION OF HELICOPTER ENGINES

Helicopter features against an airplane by capability of a main rotor to create lift and thrust necessary for its flight. Main rotor rotation requires mechanical energy. This energy is produced by the power plant, which includes one or more engines and systems to ensure their work. The primary source of energy is a chemical reaction of fuel oxidation.

Thus, the helicopter engine is a heat machine that transforms chemical energy of fuel combustion into mechanical energy.

Fuel combustion requires an oxidant. As the helicopters is an aircraft operated in the lower atmosphere, so oxygen can be used as an oxidizer, which content in the air is about 20 %. Therefore, only supply of fuel is necessary on board.

Two classes of internal combustion engines are used for helicopters: piston and turbine.

Piston helicopter engines have a star arrangement of air-cooled cylinders or in-line arrangement with water cooling.

Generally, these are four-stroke spark ignition engines. Aviation kerosene is used as a fuel. These engines provide the best performance in terms of efficiency and weight in comparison with the turbine engine with a power of less than 300 ... 600 kW, so they are placed on the light helicopters.

Medium and heavy helicopters are equipped with turbo-shaft engines, which are the subject of further analysis.

Gas turbine (GTE) engine means heat engine of internal combustion, where the air, which oxygen is used as an oxidizing agent, is pre-compressed by the compressor, driven by the turbine.

Turboshaft engine (TShE) is a gas turbine engine, in which a large part of the thermal useful energy obtained during execution of working process, is transformed by the gas turbine into mechanical work on the output shaft.

9.2. ENGINE GENERAL STRUCTURE

Currently, all helicopter turbine engines are designed with free turbine. The free turbine is used to drive the output shaft. It is not connected mechanically with the other part of the turbine, which drives the compressor. TShEs with free turbine have a number of advantages over the TShE of single-shaft structure: quick start, smaller power required of starting unit, independent control of rotation frequency of a helicopter main rotor and the generator rotor.

Conventional turboshaft engine consists of an air inlet, an air intake, a combustion chamber, a compressor turbine, a free turbine and an exhaust unit (Fig. 9.1).

The **air intake** is designed to feed the engine input with specified amount of air, ordering its flow at the compressor inlet and preliminary pressure increasing under the influence of the dynamic pressure. At low speeds specific for helicopters this increase in pressure is small.

The **compressor** provides the required pressure and flow rate of air used.

The **combustion chamber** is intended to implement the process of supplying heat to the working fluid. The gas is created as a result of fuel (working medium) combustion with high energy.

The **turbine** converts energy of working medium into mechanical energy.

The **compressor turbine** transmits this energy to the compressor; and the free turbine transmits it to the output shaft.

The **output shaft** is connected to the gearbox of the helicopter.

The **exhaust unit** is intended for discharging the exhaust gas from the turbine.

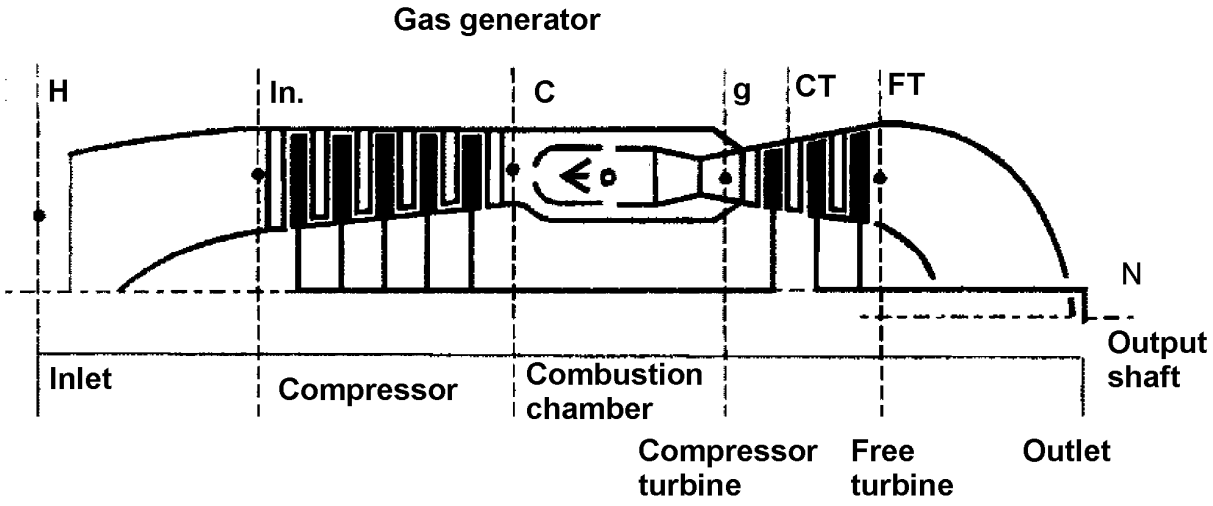


Fig. 9.1. Helicopter Turboshaft Engine Layout

The inlet unit, compressor, combustion chamber and compressor turbine form a **gas generator**.

Thus, the useful work is done by the working medium on the free turbine. The working medium in the inlet unit and the compressor is the air, and in the combustion chamber, turbine and exhaust unit is a mixture of air with the products of combustion.

The gas generator is used for preparation of the working medium to useful work on the free turbine.

9.3. ENGINE KEY DATA AND SPECIFIC PARAMETERS

Turboshaft engines have the following key parameters:

N_e – effective power on the output shaft, kW;

G_{fuel} – fuel consumption, kg/h;

G_{air} – air flow at the compressor inlet, kg/s;

m_{eng} – engine weight, kg;

D, L – overall dimensions of the engine, m;

F_{in} – inlet sectional area, m^2;

τ – service life, h;

S – cost.

Basic data can not be used as criteria for comparing perfection of different engines. For this purpose, the relative and specific parameters are used:

– specific fuel consumption $C_e = \frac{G_{fuel}}{N_e}, \frac{kg}{kW \cdot h}$;

– specific gravity $\gamma_{eng} = \frac{m_{eng}}{N_e}, \frac{kg}{kW}$;

– specific power $N_{sp} = \frac{N_e}{G_{air}}, \frac{kW}{kg/s}$;

– drag power $N_f = \frac{N_e}{F_{in}}, \frac{kW}{m^2}$.

All data and specific parameters except for weight, dimensions, cost and service life depend on flight conditions. Therefore, it is decided to compare them under two specific conditions:

– at the maximum power at the ground level at zero flight speed;

– in cruising flight.

9.4. BASIC LAWS OF WORKING PROCESS

Let us analyze the work of turboshaft engine as a heat engine. For this thermodynamics uses an i-s diagrams, where i is an enthalpy in J/kg; S is an entropy of the working medium. These parameters are specific ones corresponding to 1 kg of the working medium.

Note also that in each section of the engine flow passage the thermodynamic condition of working medium is characterized by temperature and pressure, which have static and total values (the last ones match to stagnated flow). The engine working medium temperature is associated with the enthalpy by the expression $i = C_p T$. Accordingly, if the working medium velocity is c, then $i^* = i+(c^2/2)$; $p^* = p+\rho(c^2/2)$.

Fig. 9.2 shows the TShE ideal i-s workflow diagram obtained under the assumption that the efficiencies of all units are equal to one.

In addition, it indicates the values of the specific works of compressor L_c, compression L_{comp}, expansion L_{exp}, compressor turbine L_{ct}, and free turbine L_{ft}, as well as amount of supplied heat Q_1 and extracted heat Q_2.

Having analyzed Fig. 9.2, we got the expression describing the free turbine work

$$L_{FT} = L_{exp} - L_{CT} - \frac{c_N^2}{2} = L_{exp} - L_C - \frac{c_N^2}{2} = L_{exp} - L_{comp} + \frac{v_f^2}{2} - \frac{c_N^2}{2},$$

where v_f – flight speed; c_N – gas output velocity.

From this expression it is clear that to improve the TSHE efficiency you need to ensure output of the working medium with speed as low as possible. Therefore, the exhaust unit channel of the helicopter engine is made as expanding (diffuser). Further it is assumed that the speed c_c is equal to the minimum possible value, t. e. $c_c = v_f$.

Then $$L_f = L_{exp} - L_{comp}.$$

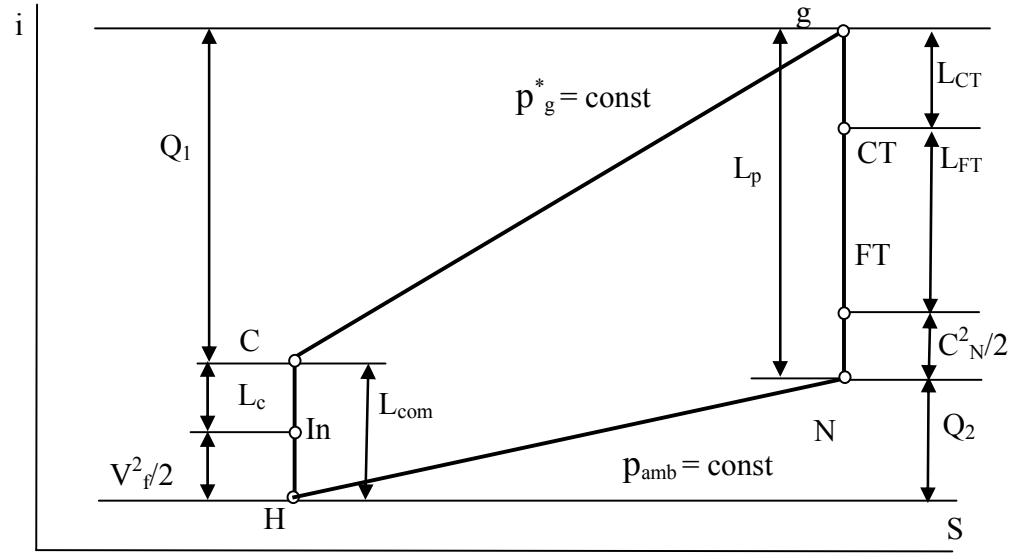


Fig. 9.2. Ideal i-s Diagram of TShE Process

Designate the engine pressure ratio as $\pi_\Sigma = \frac{p_c^*}{p_{amb}}$. Note that $\pi_\Sigma = \pi_v \pi_c^*$, where $\pi_v = \frac{p^*}{p_{amb}}; \pi_c = \frac{p_c^*}{p_{amb}^*}$. For helicopters with small flight speed values of π_v do not exceed 1.08. Then, considering the processes of expansion and compression as isentropic obtain:

$$L_p = C_p T_g^* \left(1 - \frac{1}{\pi_\Sigma^{\frac{k-1}{k}}}\right); \quad L_{comp} = C_p T_{amb} \left(\pi_\Sigma^{\frac{k-1}{k}} - 1\right);$$

$$L_{FT} = C_p T_{amb} \left(\pi_\Sigma^{\frac{k-1}{k}} - 1 \right) \left(\frac{T_g^*}{T_{amb}} \frac{1}{\pi_\Sigma^{\frac{k-1}{k}}} - 1 \right), \qquad (9.1)$$

where k – the adiabatic index adopted here the same for all sections of the flow passage.

This expression allows us to analyze the impact of various factors on the specific work of engine. Obviously, the more specific work, the more perfect engine, as each kilogram of its working medium makes greater job.

Influencing factors may be divided into two groups:

– engine cycle thermodynamic parameters T_g^* and π_c^*;

– parameters of external conditions T_{amb}, $M_п$.

Believing in the expression (9.1) values T_{amb} and π_z as constant, we find that work is a linear function of T_g^*.

Dependence on $\pi_к^*$ is more complicate: it follows from (9.1) that $L_{ft} = 0$ at $\pi_\Sigma = 1$ and at $\pi_\Sigma = \pi_{\Sigma max} = \left(\frac{T_g^*}{T_{amb}} \right)^{\frac{k}{k-1}}$. For intermediate values the work is positive and has a maximum.

Nature of function $L_{ft} = f(T_g^*, \pi_\Sigma)$ is shown in Fig. 9.3 [2].

Note that relation of the free turbine with TShE effective capacity is determined by expression $L_{FT} = \frac{N_e}{G_{air}} = N_{sp}$, i.e. L_{FT} can be regarded as a specific power. It characterizes the mass-dimensional characteristics of the engine.

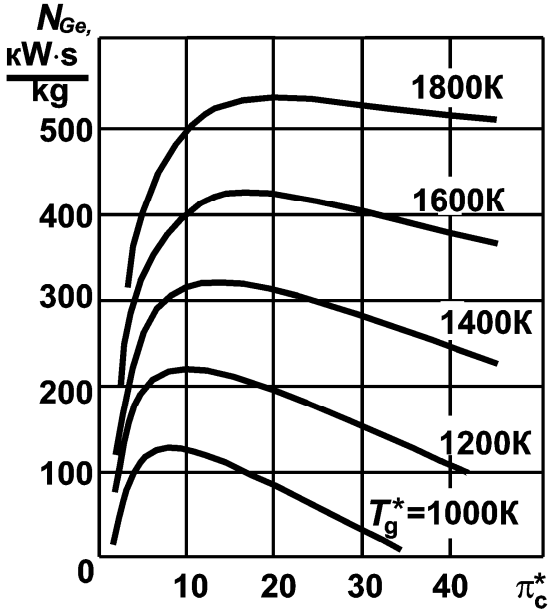


Fig. 9.3. Specific Power vs Thermodynamic Cycle Parameters

Cost efficiency of TShE is characterized by the value of fuel specific consumption $C_e = \frac{G_t}{N_e}$. Relation between C_e and parameters of the engine thermodynamic cycle T_g^* and π_c^* is shown in Fig. 9.4 [2]. It is evident that C_e reaches its minimum with $T_g^* = \text{const}$ at $\pi^*_{c.ec}$, which essentially exceeds $\pi^*_{c.opt}$.

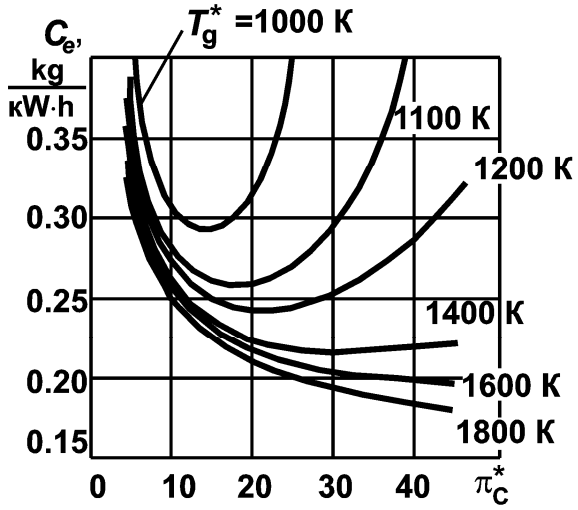


Fig. 9.4. Specific Fuel Consumption vs Thermodynamic Cycle Parameters

Thus, the turbine inlet gas temperature and total compressor pressure ratio are the main parameters of the thermodynamic cycle. *Values of* π'_Σ *are high, therefore, the entire history of GTE can be seen as the struggle for the improvement of* T_g^* *and* π_c^*. Value T_g^* is limited by capabilities of fuel used. For kerosene $T_{g.max} \approx 2600$ K. At that $\pi^*_{c.opt} > 50$, and $\pi^*_{c.ec} > 100$.

For 60 years development of gas turbine engine gas temperature before the turbine temperature has been doubled (from 1000 to 2000 K), and π_c^* has risen 10 times (from 3…4 to 30…45).

From these graphs it is evident that the larger T_g^*, the more efficient the engine, as N_{sp} increases and C_e decreases. Therefore, the design engine temperature is determined by the achieved design level of cooling the turbine parts, and first of all - its working blades. The pressure ratio in the compressor is selected based on the above dependencies $N_{sp} = f(T_g^*, \pi_c^*)$ and $C_e = f(T_g^*, \pi_c^*)$. From these graphs it is evident that at high values of T_g^* there is no sense to choose the values π_c^*, which are significantly higher than $\pi^*_{c.opt}$, as C_e decreases slightly and increase of π_c^* is achieved by increasing the number of compressor stages, weight and dimensions of the engine.

Basic data of helicopter engines having power exceeding 1,500 kW are listed in Table 9.1.

Table 9.1

Turboshaft Engines Having Power Exceeding 1500 kW

Engine	Manufacturer	Ne, kW	Ce, kg/kW·h	G_{in}, kg/s	T_g^*	π_c^*	z_c	z_f	Rotorcraft
Makila 2A	**Turbomeca**	1567	0.285		1400	12.4	3+1ct	2+2	**EC225/725 Super Puma**
RTM 322-01/9	**RR/Turbomeca**	1735	0.256	8.7	1550	16.1	3+1ct	2+2	**NH Industries NH90**
TB3-117BMA	**Motor-Sich, Ukraine**	1765	0.285	8.6	1263	9.4	12	2+2	**Mi-17, Mi-24, Ka-50**

Table 9.1 (continued)

Engine	Manufacturer	Ne, kW	Ce, kg/kW·h	G_{in}, kg/s	Tg*	π_c*	z_c	z_f	Rotorcraft
T700/T6B	**General Electric**	1775	0.263		1588	18	5 + 1ct	2+2	**NH Industries NH90**
CT7-8	**General Electric**	1879	0.274		1650	21	5+1 ct	2+2	**Sikorsky S-92**
T64-GE-16	**General Electric**	2525	0.298	12.6	1300	13	14	2+2	**Sikorsky SH/HH-53C**
T73-P-700	**Pratt Whitney**	3550	0.429	23	1140	6.75	9	2+2	**CH-54B**
T55-L-714	**Honeywell**	3629	0,304	14	1250	9.3	7+1 ct	2+2	**MH--47**
D-25B	**Perm Motors**	4050	0.390	26	1160	5.8	8	1+2	**Mi-6, Mi-10**
AE1107C	**Rolls Royce**	4586	0.249						**Bell Boeing V-22**
D-136	**Ivchenko-Progress**	8640	0.27	39.4	1478	17.6	6+6	1+1+2	**Mi-26**
D-27	**Ivchenko-Progress**	10300	0.231	37.6	1640	23	5+3(1 ct)	1+1+4	**An-70**

9.5. HELICOPTER ENGINE CHARACTERISTICS

Helicopter engine runs on a wide variety of modes within a wide zone of high altitude and climatic conditions of flight. Dependencies between the working process parameters and mode of operation & environmental conditions are called **engine performance**.

For consideration of the performance identify a free turbine with an exhaust unit and a gas generator being parts (see Fig. 9.1) of a turboshaft engine (see Fig. 9.5).

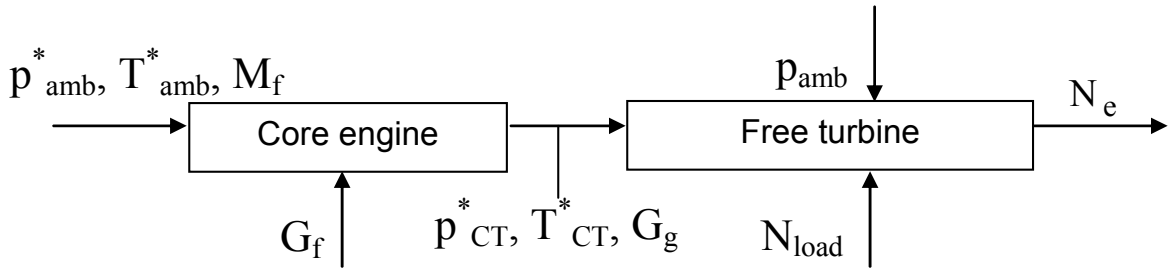


Fig.9.5. Turboshaft Engine Basic Components

9.5.1. Engine Partial-load Performance

Free turbine has a little effect upon the operation of the gas generator. Therefore, parameters in different sections of its gas path (see Fig.9.1), and the rotor speed of the gas generator are determined by external conditions (p_{amb}^*, T_{amb}^*, M_f) and fuel consumption. Thus, under constant external conditions the gas generator operation mode can be set by a single parameter (e.g., fuel consumption, rotor speed, compressor discharge pressure, turbine discharge temperature and others). This means that such gas generator has one degree of freedom. Other parameters of the gas generator are determined by the terms of the joint work of its units.

Turboshaft engine parameter dependence on the operating mode of the gas generator is called its **part-load performance** (see Fig. 9.6 [3]). In presenting the part-load performances in the technical documentation the turbocharger rotor speed is usually chosen as its argument, and for two-shaft gas generators it is rotation speed of one of the rotors.

Note that Fig. 9.6, in addition to the gas generator parameters also shows the free turbine power. Here, this dependence corresponds to constant rotation speed of the free turbine.

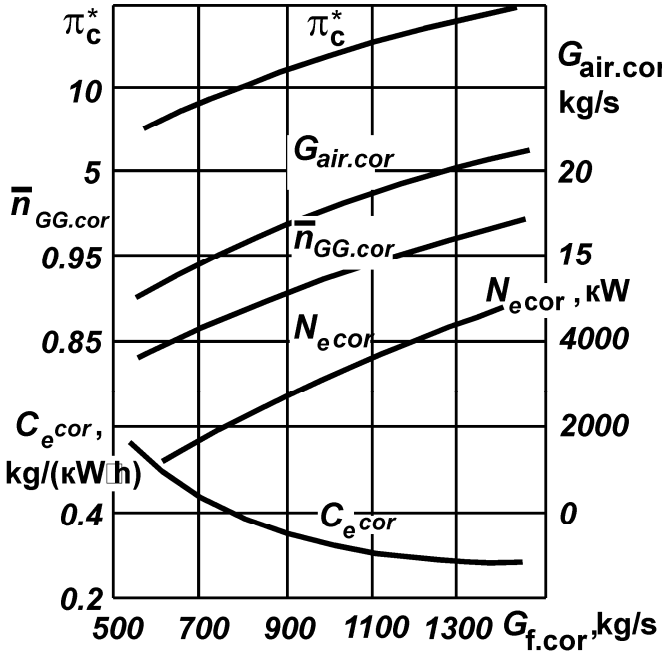


Fig. 9.6. Examples of Part-Load Performances

9.5.2. External (Loading) Engine Performance

Parameters at FT inlet are the result of the gas generator operation. As mentioned above, they only depend on ambient conditions and fuel consumption. However, FT output depends not only on the operating parameters of the working medium, but also on FT ability to convert them into mechanical power, i.e. from FT efficiency.

FT efficiency η_{FT} depends on the rotor frequency of rotation. This is explained so that under change of η_{FT} and constant value of gas consumption (which is determined by operation of the gas generator) the turbine blade airfoils angle-of-attack change: under low η_{FT} their values shift into the zone of larger positive angles, and under high η_{FT} the values shift into the zone of negative angles-of-attack. That is why values of η_{FT} and N_e at constant parameters of the gas generator depend on η_{FT}, and this dependence has a maximum.

Dependence between the output shaft power and FT rotation speed and gas generator operating mode is named as **external** (***or loading) curve***.

Fig. 9.7 [1] shows external curve of the Turbomeca helicopter engine corresponding to constant ambient conditions ($H = 0$, $M_f = 0$).

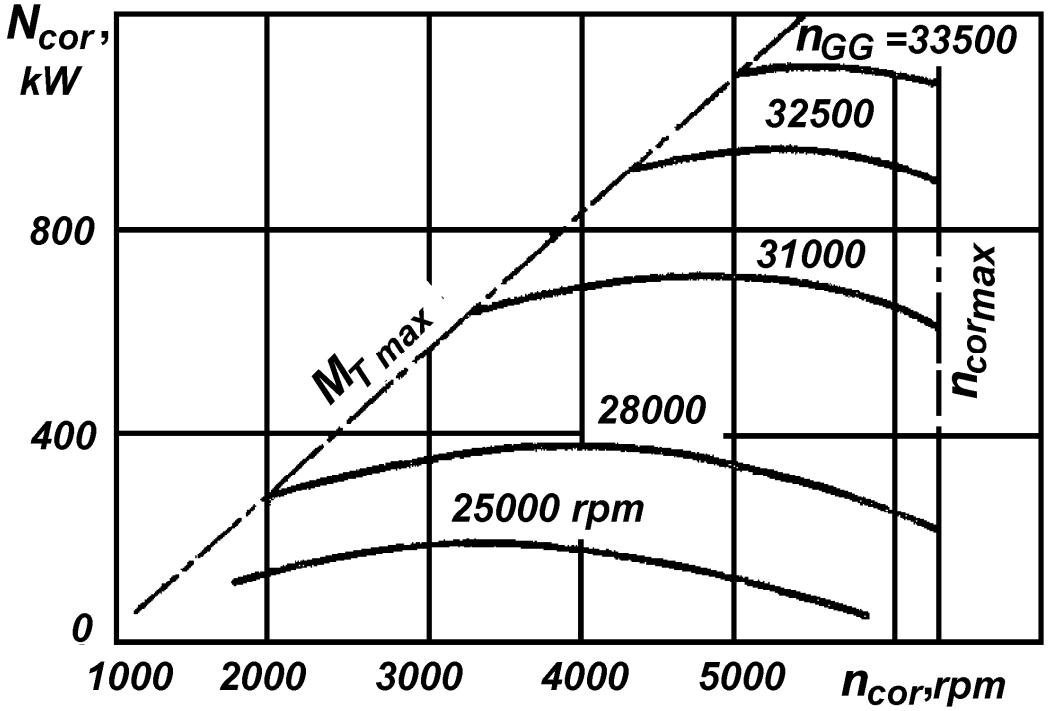


Fig. 9.7. Example of Turboshaft Engine External Performance

9.5.3. Influence of Ambient Conditions on External and Part-load Engine Performances

The above performances $n_{TC} = f(G_T)$ and $N_e = f(n_{TC}, n_{FT})$ depend on ambient parameters that is in general the following functions take place $n_{TC}=f(G_f, p_{amb}^*, T_{amb}^*, M_f)$ and $N_e=f(n_{TC}, n_{FT}, p_{amb}^*, T_{amb}^*, M_f)$.

Application of similarity theory allows us to represent the performances in a simpler form $n_{TCcor} = f(G_{T.cor})$ and $N_{ecor} = f(n_{TCcor}, n_{FTcor})$, if to use reduced values of the working parameters:

$$n_{cor} = n\sqrt{\frac{T_o}{T^*_{H}}};\quad T^*_{cor} = T^*\frac{T_o}{T^*_{amb}};\quad p^*_{cor} = \frac{p_o}{p^*_{amb}};\quad N_{e.cor} = N_e\sqrt{\frac{T_o}{T^*_{amb}}}\frac{p_o}{p^*_{amb}};$$

$$G_{T.cor} = G_T\sqrt{\frac{T_o}{T^*_{amb}}}\frac{p_o}{p^*_{amb}};\quad G_e = C_e;\quad G_{r.cor} = G_r\sqrt{\frac{T^*_{amb}}{T_o}}\frac{p_o}{p^*_{amb}}.$$

These corrected values are used in the presentation on the part-load performances in Fig. 9.6.

9.5.4. Performances Defined by Restrictions on Parameters of Engine Process

The above performances reflect the properties of the engine as being a heat engine and does not take into account the constraints on the parameters of the working process.

Under real conditions the engine is running as part of automatic control system, which controls fuel flow. The functions of the control includes execution of the main control program and a series of limitations associated with a certain level of reliability.

The trim schedule means dependency of regulated parameters on stimulus command Z (which determines the thrust rating) and the external environment.

The main trim schedule is aimed at maintaining the given power. However, power is not measured. Therefore, given the fact that between the parameters of the gas generator there is a unique relation so choose the parameter that is the most accurately measured to be the main control parameter – n_{TC}, and with constant command stimulus support n_{TC} = const.

In addition, at the established power ratings the following programs - restrictions are implemented:

$$N_{FT} \le N_{FT\,max};$$

$$N_{CT} \le n_{CT\,max};$$

$$T^*_g \le T_{g\,max}\ (\text{or } T^*_{CT} \le T_{CT\,max});$$

$$p^*_C \le p_{C\,max};$$

$$N_{CT\,cor} \le n_{CT\,cor.max};$$

$$M_{trq} \le M_{trq.max}.$$

Limiting torque is a characteristic feature of TShE engine. Given that $N_e = M_{trq}(\pi n_{FT}/30)$, and η_{FT} changes slightly, in fact this program restricts N_e.

As will be shown below, for helicopter turbine engine the output shaft power $N_{e\,max}$ is taken as maximum permissible power at the design height H_{des}. Al altitudes smaller the design value, the engine possesses a significant excess in power. Engine operation with increased power at $H < H_{des}$ is inadmissible, since there is a risk to the strength of the engine, gearbox and transmission.

Since the values of the engine parameters are associated with conditions of joint operation of units and depend on the command stimulus Z and the thermodynamic state of the air at the engine inlet, each engine parameter Y can be represented as $Y = f(Z, p^*_{amb}, T^*_{amb}, M_f)$.

Values of p^*_{amb}, T^*_{amb}, M_f are determined by high-altitude climate and speed conditions.

9.5.4.1. Climatic Performances

Climatic performances mean relations between engine parameters and the ambient temperature T_{amb} under given engine rating.

The example of a climatic performances for the «maximum» rating is shown in Fig. 9.8 [4].

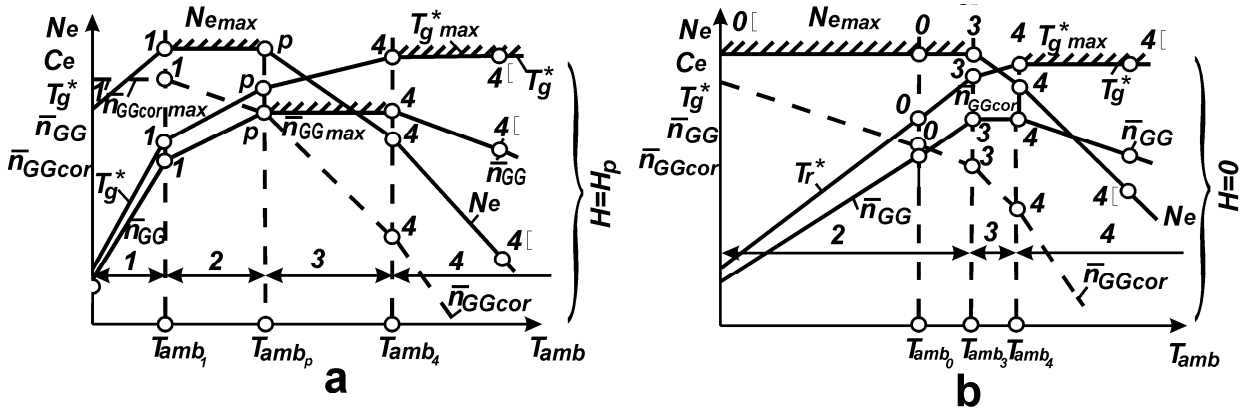


Fig. 9.8. Example of Turboshaft Engine Climatic Performances:

$a - H < H_{des}$; $b - H = 0$

The characteristics sections 1, 2, 3 and 4 correspond to the following restrictions: $1 - n_{CT.cor.max}$; $2 - N_{e\,max}$; $3 - n_{CT.max}$; $4 - T^*_{g.max}$.

9.5.4.2. Altitude Performances

Altitude performances are functions of engine parameters on flight altitude. Atmospheric air temperature and pressure are considered as equal to standard values.

The example of altitude performance for maximum rating is shown in Fig. 9.9 [4]. Highlighted sections correspond to the following limitations: 1 – $N_{e\,max}$; 2 – $n_{TC.max}$; 3 – $n_{TC.cor.max}$.

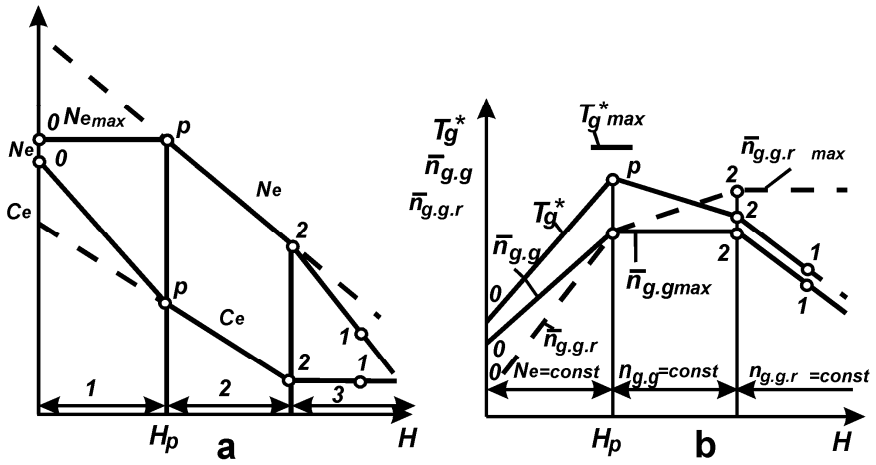


Fig. 9.9. Example of Turboshaft Engine Altitude Performances:
a – N_e, C_e; b – n_{TC}, T^*_g, $n_{TC.cor}$

9.5.4.3. Speed Performances

Heavy helicopter flight speed is usually higher than 350 km/h. The values of dynamic pressure are small and do not cause significant (over 5%) change of engine parameters. Therefore, for this class of engines the speed performances are not considered.

9.6. MATCHING OF ENGINE AND HELICOPTER

9.6.1. Joint Operation of Main Rotor and Free Turbine

Free turbine is connected to the helicopter main rotor and the steering rotor via transmission. The rotor is the main consumer of power. Therefore, the condition of the engine and rotor joint operation at steady conditions is the power balance: $N_e \eta_{tr} = N_r$,

where N_r – rotor power, η_{tr} – transmission efficiency.

N_e depends on the gas generator operating mode and n_{FT} (see Fig. 9.7). V N_e depends on the rotor frequency of rotation n_r and its blade setting angle φ_r. Considering that $n_r = i_{fl} n_{FT}$, this relation may be represented as follows: $N_r = K(\varphi_r)\, n^3_{FT}$.

In Fig. 9.10 the rotor performances are joined with the engine external performance. It is evident from the figure that each value of G_f and φ_r corresponds to certain values of n_{CT}, G_f and other gas generator parameters.

9.6.2. Provision of Helicopter Demand Power Performances

Fig. 9.10 shows the power required for the hover and horizontal helicopter flight vs $\mathbf{v}$ and H [1]. With an increase in airspeed at a predetermined altitude the required power first reduces and then increases.

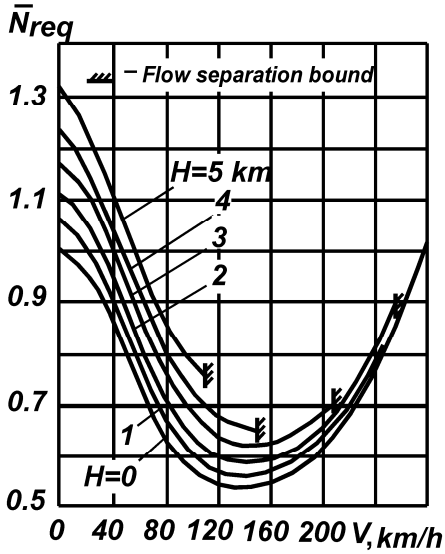


Fig. 9.10. Example of Helicopter Power Required vs Flight Altitude and Speed

The available engine power within this range of speeds virtually remains unchanged. If the engine power is selected from standard atmospheric takeoff conditions at $H = 0$, then this power exceeds the needs for flying in a given range of speeds at a given altitude. In this case the engine must be throttled. Rating of maximum flight duration corresponds to the condition of minimum power required, since in such circumstances the fuel consumption is the least (C_e increases when the engine is throttled in accordance with Fig. 9.6 is much less than reduction in the required power according to Fig. 9.11). However, with increasing flight altitude the required power increases (see Fig 9.11), and the available engine power decreases (see Fig 9.9).

Therefore, having matched the helicopter with the engine in hovering at H = 0, we can not ensure its mission in the field of high-altitude mode, corresponding to low speed flight. The same mismatch of required and available power is obtained by increasing the air temperature when N_{req} virtually is unchanged, and N_e is significantly reduced (see Fig. 9.8.).

This contradiction can be overcome by a decrease in the payload, if necessary to fly at high altitude and in hot climate. However, this method is economically inexpedient.

In order to achieve harmonization of the helicopter and engine performances within the wide range of the altitude and climatic conditions, use helicopters engines, ensuring takeoff and hovering at different altitudes and at high air temperature.

The engine that meets these requirements, is re-sized, and to ensure equality of the required and available power it must be throttled under normal atmospheric conditions.

Given this, it is advisable to limit the available maximum engine power (or torque) and thereby protect its transmission and gear against mechanical overload and to reduce their weight.

9.7. HELICOPTER GTE STRUCTURE

As an example of GTE for heavy helicopter let's consider the D-136 engine developed by Ivchenko-Progress Zaporozhye State Enterprise.

Main engine parameters are listed in Table 9.1 above.

Air intake is made via the dust protective device which is a unit of the helicopter.

The engine (see Fig. 9.11) consists of a gas generator, free turbine and exhaust unit.

The **gas generator** is of a two-stage scheme and consists of a compressor, combustor, and compressor turbine.

The Compressor is of axial type, consists of two stages - transonic low pressure compressor (LPC) and subsonic high pressure compressor (HPC). Each compressor stage is driven by own turbine stage**.**

Compressor stages are interconnected only by the gas-dynamic link – common air flow.

LPC is of axial six-stage type, consists of a rotor and stator. The first stage has inlet guide vanes (IGV). Each stage consists of working blades and straightener blade. The blades are contoured such that channels between them expand in the direction of air flow. This leads to increased pressure. The blades are mounted on the rotor, which transmits them mechanical energy for movement and keeps them from shifting radially and axially under action of centrifugal forces, inertial forces and gas. The LPC rotor is of a drum-disk design: blades are fixed on the disk with special locks and discs are interconnected with cylindrical drum spacers. The spacers and disc are joined together by electron beam welding and attached to the shaft by means of flanges.

LPC rotor is connected to the LP turbine rotor via slots.

All engine shafts have minimum number of supports equal two. This helps to reduce engine weight.

Front LPC rotor support is of angular contact elastic damping. The ball bearing is of three-point contact type, installed in a cup with an elastic element. This design reduces stiffness of the rotor and improves dynamic performance.

LPC stator consists of a front casing, intermediate casing, inlet guide vanes (IGV), guide vanes of intermediate stages and the final stage straightener blade. A spacer serving for attachment of dust-protective device (DPD) at the engine inlet is attached to the front flange of the front casing outer ring.

An annular receiver is mounted on the intermediate casing connected by holes with gas path section behind the LPC third stage.

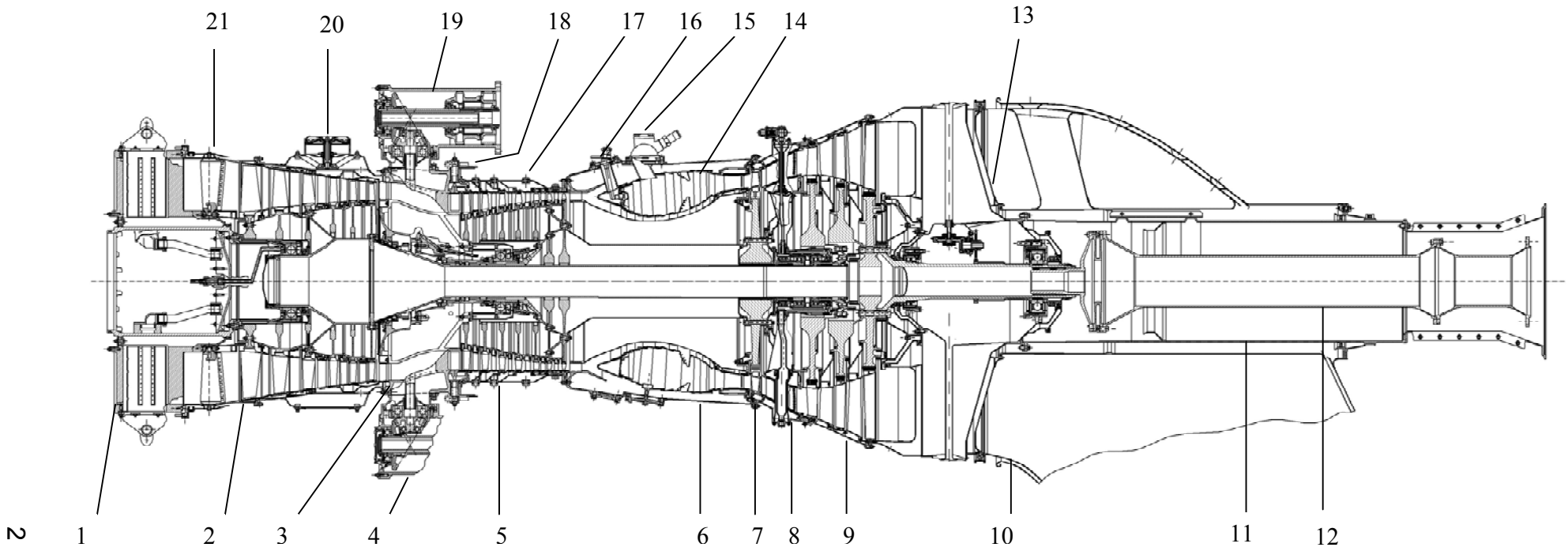


Fig. 9.11. Helicopter Turboshaft Engine with Free Turbine and Two-Stage Gas Generator:
1 – front frame; 2 – low pressure compressor; 3 – intermediate casing; 4 – lower gear box; 5 – high pressure compressor; 6 – combustion chamber; 7- high pressure turbine; 8 – intermediate pressure turbine; 9 – free turbine; 10 – exhaust unit; 11 – cone-shaped beam; 12 – output shaft; 13 – housing bearing poles of free turbine; 14 – flame tube; 15 – starting igniter; 16 – fuel nozzle; 17 – inspection window for gas path; 18 – HPC inlet guide vanes; 19 – top gear box; 20 – LPC air bypass valve; 21 – LPC inlet guide vanes

The air bypass valves are placed on the receiver to ensure stable operation of the LPC at part-load operating modes.

On LPC middle casing the inspection bosses are placed used to input optical instrument when inspecting LPC rotor blades in operation. Rear flange is used to attach the LPC middle casing to the intermediate casing.

The intermediate casing is a key load-bearing element of the engine. It accommodates the gear box and accessories as well as front engine suspension components.

HPC is of axial, seven-stage type, consists of IGV, rotor, stator, air bleed valve and HP rotor front support bearing assembly. The main structural elements of the HPC are similar to the corresponding elements of the LPC. Welded drum, discs of the last stages, spacers and shafts are interconnected by bolts; blades are connected with discs by means of shanks of "dovetail" type.

HPC rotor is connected to the HP turbine with bolts.

The front ball bearing is mounted in the elastic support with rigid travel restrictor. Rear roller bearing of high-pressure rotor is placed in the oil damper.

Blades of HPC inlet guide vanes are made swivel that allows the engine setting-up during bench tests. After setting-up the IGV blades are fixed in the selected position.

The **intermediate casing** is used for formation of a transition path from the LPC to HPC, accommodation of units and drives for them, as well as placement of the HP rotor front support and front cap of engine mount.

The annular shells forming the gas path are interconnected by hollow ribs, which accommodate service lines.

Intermediate casing consists of a body with drive boxes and a central drive. All drive units are driven by the HP rotor.

LPC is attached to the intermediate casing from the front, and HPC –from behind.

Combustion chamber is of an annular type, consists of an outer and inner casings, diffuser with a straightener of the HPC last stage, flame tube, fuel manifold with fuel injectors and starting igniters.

Air is heated in the combustion chamber by the heat, escaping in burning of atomized fuel in its flame tube continuously injected by operating nozzles. The flame tube separates the combustion zone, providing an air flow in this part sufficient for ustainable fuel combustion. The remaining air portion mixes with the combustion gases and reduces their temperature upstream the turbine.

The flame tube has a welded structure composed of individual rings having a number of holes for passage of secondary air.

The fuel injectors are centrifugal, single-channel units.

Fuel ignition in the combustion chamber during start is provided by two staring igniters mounted on its housing.

The engine **turbine** is three-shaft, reactive unit, consisting of three stages: high pressure (HPT), low pressure (LPT) and a free turbine (FT). HPT drives the HPC, LPT drives LPC and FT drives the helicopter transmission.

The roller bearings are supports of the turbine stages. All supports have oil dampers for damping vibrations arising during the engine operation.

HPT is of single-stage type, consists of a casing, nozzle unit and rotor. The vanes of the nozzle unit and the rotor blades have internal channels and cooled with air coming out of the HPC.

The blades are contoured such that channels between them are narrowed in the direction of the gas flow. This reduces pressure and temperature. Rotor blades are mounted on the rotor, which takes mechanical energy from them and prevents them from shifting in the radial and axial directions. Each groove accommodates two blades.

The nozzle unit consists of eight different sectors. Vanes are cooled, deflector-type.

HPT disc and rotor blades are cooled with air coming out of the HPC. Cooling of blades is convective.

LPT is of single-stage type, consists of casing, nozzle block and rotor. Nozzle block vanes are cooled, rotor blades are not cooled. The hollow vanes of the nozzle block accommodate rods being load-bearing members connecting supports housing with an outer turbine casing. Also all service lines pass through the vanes.

The outer casing has windows for inspection of HPT and LPT rotor blades, closed with plugs.

FT is two-stage, not cooled unit, consisting of stator, rotor and FT posts housing.

FT discs are cooled by air bleed from HPC intermediate stage. They are interconnected and connected with FT shaft by bolts.

FT shaft transmits torque to the output shaft due to spline connection, which rotates the helicopter transmission. It is placed inside the cone-shaped beam.

The support housing is connected to external casing by load-bearing posts protected from high temperatures with aerodynamic shells. The external housing has windows for rotor blades inspection.

Exhaust unit is placed behind the housing of the FT supports. Its shape provides gas output aside of the helicopter structural members.

9.8. HELICOPTER POWER PLANT SYSTEMS

The engine normal operation within the assigned lifetime in a wide range of flight conditions and operating modes is provided by the auxiliary systems: fuel, oil, electric, starting, ice protection, bleed air, fire protection, control, diagnostics [5]. Below they are considered briefly using the D-136 engine as an example.

9.8.1. Fuel Supply System

Fuel supply system feeds engines with fuel under any operation conditions. It consists of a low pressure system, high pressure system and starting system.

Low-pressure system consists of fuel tanks, booster pumps, preservation nozzle and fire hydrant located in a helicopter and power plant section: booster pump, fuel-oil unit with main fuel filter, differential pressure switch on fuel filter and piping.

High pressure system includes high pressure gear pump, fuel control, fuel manifold, nozzles and piping.

Starting system includes: fuel pilot valve, starting igniters, starting fuel nozzles and piping.

Fuel supplied from the helicopter system in the pump unit is directed by a low-pressure pump to the fuel-oil unit and the fuel filter to the high pressure pump. The pump delivers fuel to the fuel nozzles through the fuel metering controller and the fuel manifold, and also to the starting nozzles through electromagnetic starting valve.

9.8.2. Oil Supply and Breathing System

The **oil system** ensures permanent oil supply to friction surfaces in bearings of the compressor and turbine rotors and rotating parts of the central drive for their lubrication and cooling.

The oil system comprises the following components: oil tank, fuel-oil unit, air-oil heat exchanger, oil unit, air separator, safety filters of oil injectors and suction pumps, pipes, channels and nozzles, sensors, indicators and alarms for pressure, temperature and chip detectors in the oil.

The oil unit consists of suction and discharge pumps, check and reducing valves, fine oil filter with bypass valve, and differential pressure switch, air vent.

Oil from the oil tank flows by gravity into the delivery pump of the oil unit, wherefrom under pressure it is fed to a fine filter placed in the oil unit cover. The value of oil pressure at the engine inlet is maintained within a predetermined range by means of reducing valve. Having passed the fine filter, the oil via outer pipelines is supplied to LPC support and turbine supports, and through the channels made in the intermediate housing it is supplied to the HPC support, the central drive and top gear unit. The rest of the parts and components are lubricated by bubbling.

Oil is taken out from bearing cavities by the suction pumps. On the way to pumps oil washes thermal chip detectors and is filtered by the pump safety filters. Pumped oil is drained into the sump of the bottom gear unit, from where having passed a chip detector and safety filter, is pumped by main suction pump and transferred to a centrifugal air separator. The separated oil flows from the air cooling unit in fuel-oil unit. A thermal valve mounted downstream sends oil for further cooling in the air-water heat exchanger

if the oil temperature exceeds a predetermined value. The cooled oil returns to the oil tank.

The **breathing system** connects oil cavities with atmosphere. This is necessary to ensure normal operation of the oil system and seals.

All bearing oil cavities breath into the cavity of the upper gear unit. The air-oil emulsion passes from this cavity through the centrifugal breather, where the air is separated from oil. The separated oil is discharged into the lower tray of the gear unit, and the air is vented via the outer duct into the gas path of the engine exhaust unit.

9.8.3. Electrical Equipment

Electrical equipment of the engine provides operation of:

- electrical automation of starting system;
- engine status warning system;
- engine parameters measuring system;
- electronic control system.

9.8.4. Starting System

Starting system allows starting the engine on the ground and in flight.

The starting process involves:

- air starting system;
- electrical system;
- fuel igniters;
- fuel metering system;
- automatic control system.

The air starting system provides engine start from the airfield or airborne source of compressed air or one of the working engines.

It consists of:

- source of compressed air;
- air starter with air starting valve;
- pipes, check valves and shut-off valves.

The **air starter** is a single-stage turbine, driven by compressed air. It spins the high pressure rotor, which is connected via a reduction gear.

The **air starting valve** with control unit is designed to open and shut off the air to the air starter when starting the engine.

The shut-off valves are designed for instant automatic shut-off of the starter air supply channel.

The starting program is implemented by the automatic staring panel, mounted in the helicopter, and engine fuel metering automation.

9.8.5. Drain System

A **drain system** is designed to collect and drain outside the engine the following liquids: fuel drained from the combustion chamber and turbine housing under depressurization; fuel and oil out of drainage cavity of the input shaft of the fuel seal assembly; slurry discharged from the drainage seal cavities of the hydraulic pump.

A drainage system consists of tanks, pipes and ejector shrouds installed on a section of the exhaust unit.

9.8.6. Ice-protection System

Ice-protection System (IPS) is of air-heat type and used to protect LPC IGV vanes, front housing struts and pitot tube from ice accretion. The pitot tube is permanently heated with air supplied via pipeline from the HPC. The rest of the elements listed above are heated with air taken from the HPC after switching on IPS.

IPS includes a IPS valve with switching-on indicative device, temperature control, IPS valve control and open position alarm system and piping.

When hot air passes through the temperature control the air flow is controlled depending on its temperature. Downstream the temperature control-the air through the pipes flows into the front housing struts, heating their leading edges, as well as through the annular cavity in the LPC vanes and through channels in the vanes comes in the compressor gas path.

9.8.7. Fire Protection System

Fire in the engine compartment is possible due to presence of fuel and oil, possibility of their leakage and contact with the heated structural elements. Parts of titanium alloys can ignite when the rotor touches stator elements.

Fire protection system consists of detectors and fire-fighting equipment. The fire detectors contain thermocouples; by the signal of that thermocouples an electrical circuit of fire extinguishing closes. The fire-fighting equipment include: fire-extinguishing agent bottles; squibs providing fast opening of bottles; electromagnetic control valves that supply the extinguishing fluid in the compartment where the fire arose; signaling devices of the open position of control valves. To activate the fire suppression system in case of emergency during a forced landing apply special inertial sensors, triggering under certain overload.

9.8.8. Automatic Control System (ACS)

Engine ACS performs the following functions:

– dosing starting and main fuel in the engine starting mode;

– maintaining the engine operating condition in accordance with setting command;

– limitation of the parameter maximum values (speed of rotors, corrected rotor speed, gas temperature, compressor pressure, fuel consumption, torque);

– limitation of the parameter minimum values (corrected rotor speed, fuel consumption);

– limiting parameters in acceleration mode;

– limiting parameters in deceleration mode;

– synchronizing engines that operate in parallel in common power plant;

– ensuring the compressor stable operation.

Specific feature of ACS of high-power helicopter power plant is a presence of two channels of set actions. Schematic diagram of the ACS of helicopter turbine engine fuel consumption is shown in Fig. 9.12. ACS of heavy helicopter engines features with presence of two channels of setting commands:

– The first channel («pitch ») sets the angular position of the rotor blades and automatically reconfigures the stimulus command Z, which determines the engine operating mode;

– The second channel («power») corrects the engine rotation speed (changing the value of the setting command Z) at a fixed position of the helicopter rotor blades.

This system provides flexibility to manage the helicopter, but its complexity requires appropriate training of pilots.

The fuel governor in modern ACS is made on the basis of electronic computing devices with hydromechanical actuator. This makes it possible to implement complex control program and necessary logic of selecting the active control loop.

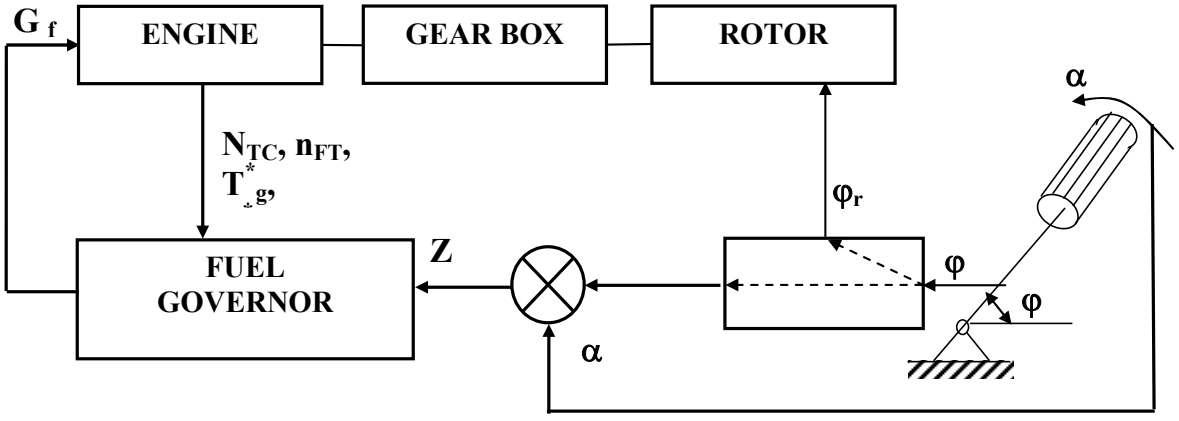


Fig. 9.12. Schematic Diagram of Helicopter Engine Automatic Control System

9.8.9. Diagnostic System

Diagnostic system solves two main tasks:

- ˜in-flight monitoring of engine and its systems and delivering information about their current status to the crew members;
- ˜diagnosing technical condition of the engine and its systems to generate recommendations for their maintenance and repair.

The composition of the onboard diagnostics system includes: measuring system with sensors, transducers and channels for information exchange; computational facilities to realize the diagnosis algorithms; indicating system, which includes monitors, signaling lamps, light and sound annunciators; recording system, including operating and fault recorders.

The results of operational recording are processed in the ground ICU-Related Diagnostics using universal computational means.

Engine diagnostic system is an important component of a complex of means for maintaining reliability and efficiency of engine operation and operation of the aircraft in a whole.

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Chapter 10. SCIENTIFIC BASIS OF DETERMINING AERODYNAMIC CHARACTERISTICS AND EXTERNAL LOADS

10.1. AERODYNAMIC ESTIMATION SYSTEM FOR SINGLE-ROTOR HELICOPTER

10.1.1. Block Diagram of Single-Rotor Helicopter Aerodynamics Estimation

Continuous development of rotorcraft (power-to-weight ratio rise, complication of carrying systems, improvement of flight performance, etc.) requires proper development and improvement of the helicopter aerodynamic estimation methods.

Helicopter aerodynamic estimation means definition of its aerodynamic characteristics on a given set of steady (stationary) flight regimes and determination by these data of the values of the power plant required power under these flight regimes. Together with the power plant available capacity under the specified atmospheric conditions the aerodynamic estimation results allow us to define the helicopter basic flight characteristics in steady flight regimes such as static and reasonable ceilings, minimum and maximum flight speed, vertical speed under various flight conditions, flight range and duration, special helicopter characteristics under critical flight regimes (including MR autorotation) etc.

The developed system of helicopter aerodynamic estimation is a complex of mathematical models and computational methods, combined in a system intended for development of a single-rotor helicopter (Fig. 10.1), which meets modern requirements. These requirements include:

1. Ability to analyze a plurality of the proposed helicopter configuration options at the initial design stage.
2. Mathematical modeling of special and critical flight regimes, extreme situations.
3. Validity of the estimation results of the helicopter basic aerodynamic flight characteristics.
4. Ability to use mathematical methods of optimization.
5. Consideration of mutual influence of the helicopter various elements.
6. Ability to change parameters of used helicopter arrangement quickly and independently compared to the physical model.
7. Modular structure of the aerodynamic estimation system, which allows for rapid modernization of individual units or replacement of the obsolete block with new one if necessary.
8. Easy and reliable operation of the aerodynamic estimation system, visual and convenient format of the results.

System of helicopter aerodynamic estimation satisfying the above requirements has been mainly developed and passed trial operation in the design of two helicopters.

Depending on the calculation required accuracy and the available amount of raw data of the projected helicopter parameters in the aerodynamic estimation system, you can use a variety of approaches.

In order to implement main flight and technical requirements for the designed helicopter during the preliminary project stage a simplified approach is used based on the method of power balance (block "Calculation of Flight Performance" at the program first level, Fig. 10.1). On the basis of this approach the basic arrangement decisions are taken, the geometric and weight parameters of the main units and assemblies are determined, as well as of the helicopter as a whole (Figs 10.2, 10.3).

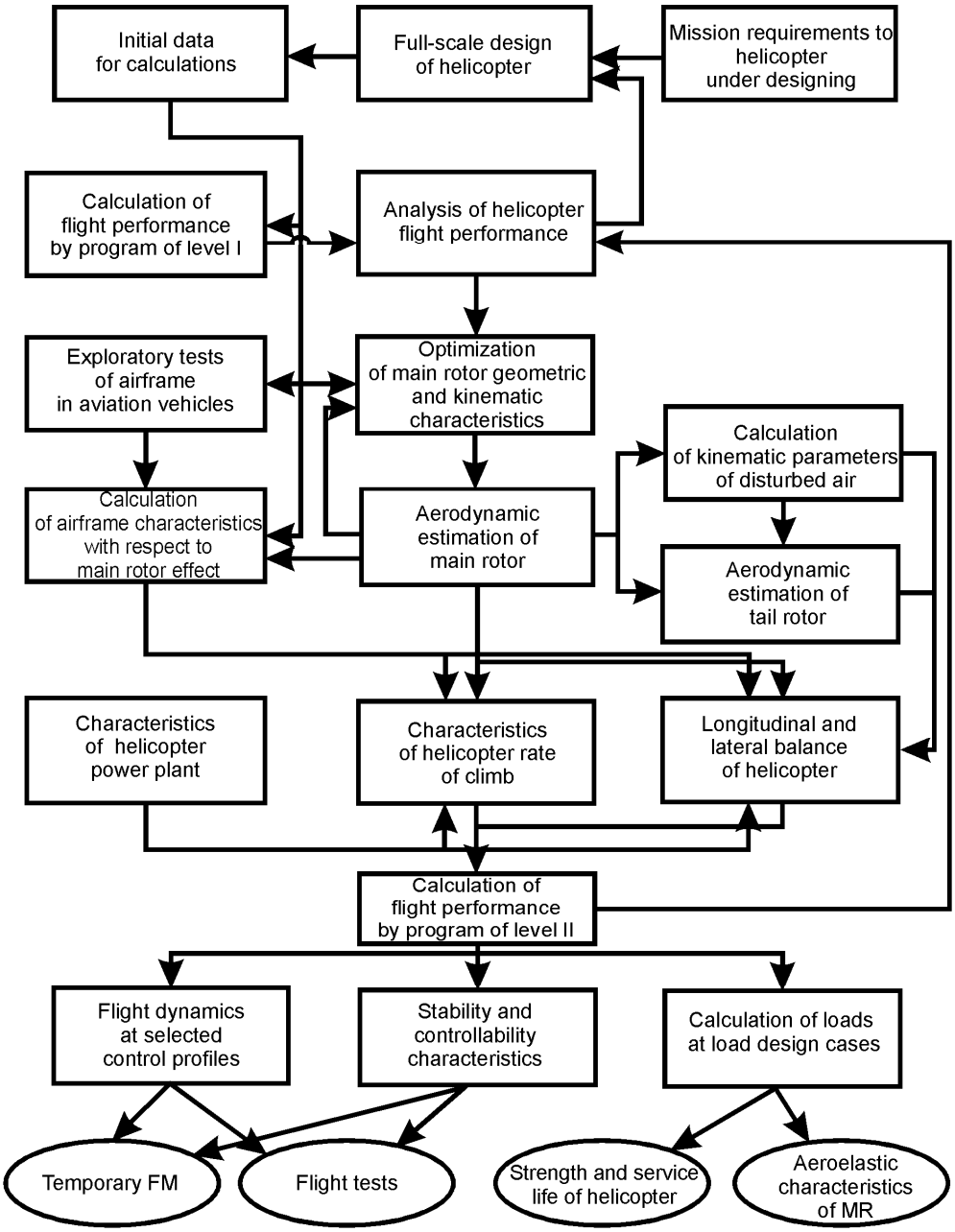


Fig. 10.1. Block Diagram of Mathematical Model
for Aerodynamic Estimation of Single-Rotor Helicopter

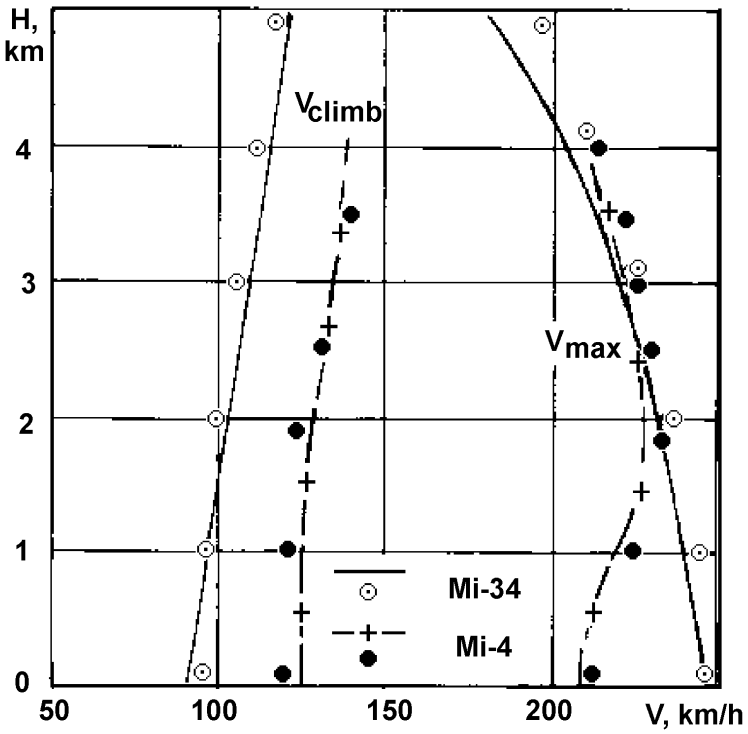


Fig. 10.2. Comparison of Design and Experimental Data with Regard to Specific Flight Velocities

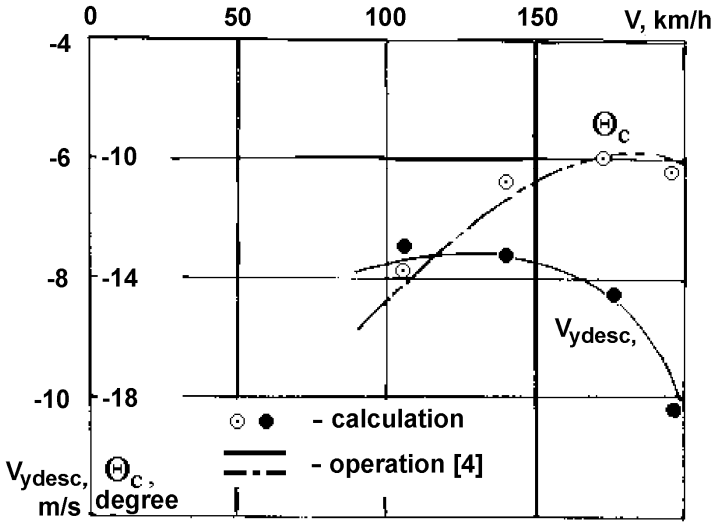


Fig. 10.3. Features of Helicopter Descending in Autorotation

At the second stage of the helicopter aerodynamic estimation the power balance method is used. Using this method allows you to optimize the carrying system at the main operating regimes of the helicopter flight (block "Optimization of main rotor geometric and kinematic characteristics", Fig. 10.4), to determine in the first approximation the required ranges of angle-of-attack and MR collective pitch, take into

account mutual interference of the individual elements of the helicopter layout (Fig. 10.5), and to determine the appropriate amount of experimental studies, etc.

At the final stage the full-balancing method is used, which requires a large amount of raw data. This method allows, along with the definition of basic flight characteristics (block "Calculation of Flight Performance by Programs of Level II") to calculate the position of balancing for all of its controls, the angles of pitch, roll and slip, angles of control surfaces setting, wing, etc. (Figs 10.6, 10.7).

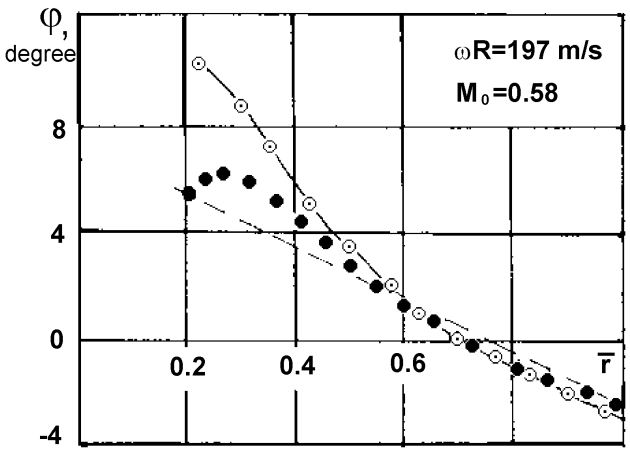


Fig. 10.4. MR Blade Optimal Twist for Two Flight Regimes

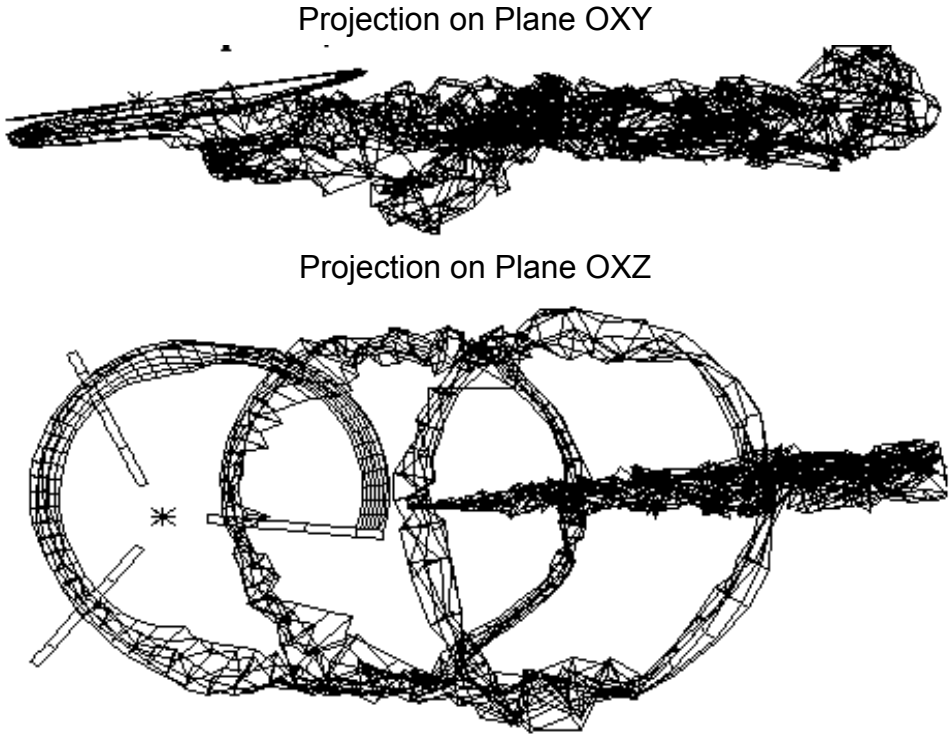


Fig. 10.5. Interaction of Vortex Structures of Main and Tail Rotors

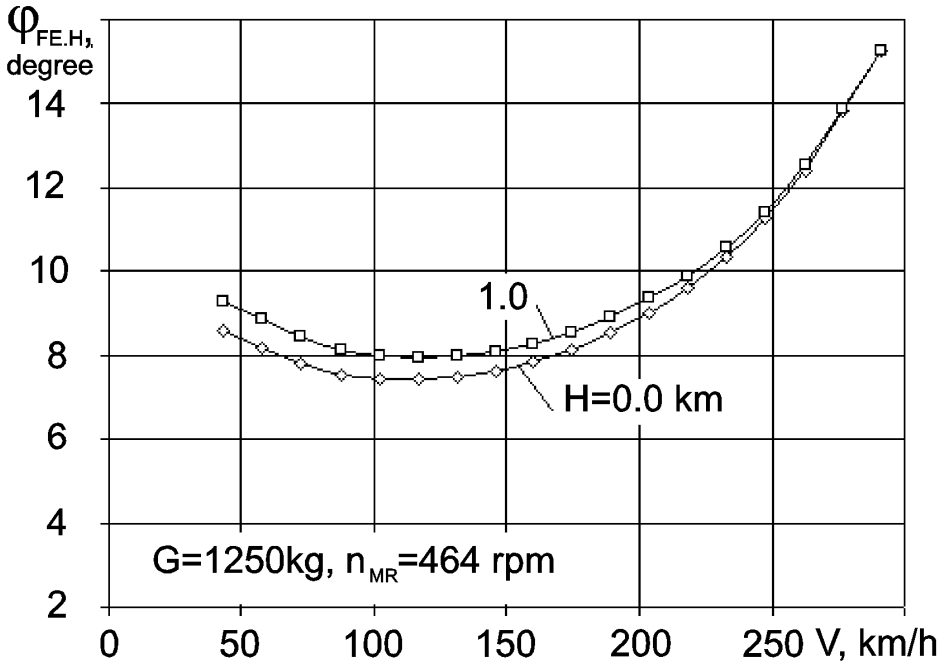


Fig. 10.6. Balancing Angles of Main Rotor Feathering Hinge in Level Flight

The above methods and approaches for helicopter aerodynamic estimations are based on different methods of determining the MR aerodynamic characteristics of different level of detail, as well as on various models of airframe aerodynamic characteristics, and accounting mutual influence of different elements (Fig. 10.8).

Together with the specified characteristics of the power plant the hourly and kilometer fuel consumption is determined (Fig. 10.9), rational cruising flight conditions are chosen. Characteristics of the helicopter hovering of both out of screen surface influence and close to the screen (Fig. 10.10) are calculated. Static ceiling is determined by the equality of the needs and available power, taking into account the recommended margins in vertical speed.

Geometric and kinematic parameters of main rotor should provide capability of steady horizontal flight at acceptable values of blade flapping angles, control margins in cyclic pitch, lack of stall at the main rotor blades within the azimuth range of 240... 300° (Figs 10.11, 10.12).

The results of the helicopter aerodynamic estimation can be used as input data for the helicopter dynamics mathematical models, characteristics of static and dynamic stability and control.

The results of aerodynamic load distribution over the blades of main and tail rotors can be used for checking strength analysis, preliminary assessment of the blade life (Fig. 10.13), etc. Distribution of aerodynamic loads over the main (tail) rotor blades allows for transition to solving aeroelastic and dynamic strength problems, avoiding self-oscillation and other dangerous resonance phenomena.

The developed system of the helicopter aerodynamic estimation is the basis for the development of rotary-wing aircraft automated design integrated system. However, even in its current form the aerodynamic estimation system allows for greater computing capacity when performing the helicopter layout parametric studies in order to optimize and meet specified tactical and technical requirements. This helps to avoid significant errors in determining the basic parameters of the projected helicopter that can lead to serious consequences both for the project and for the institution or company, executing the design process. Fig. 10.14 shows an increase in the cost of making changes to the project at its various stages, which confirms the importance and responsibility of decisions taken at the early design stages.

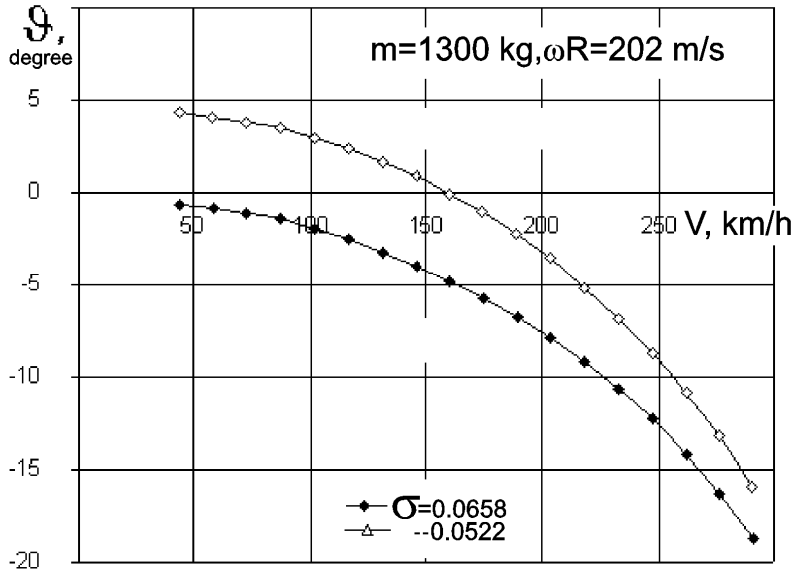


Fig. 10.7. Balancing Angles of Helicopter Pitch in Level Flight

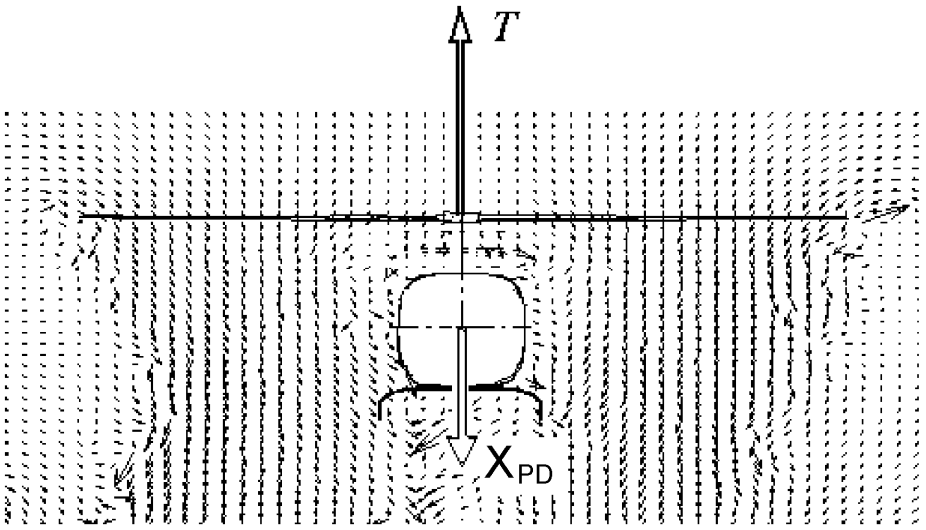
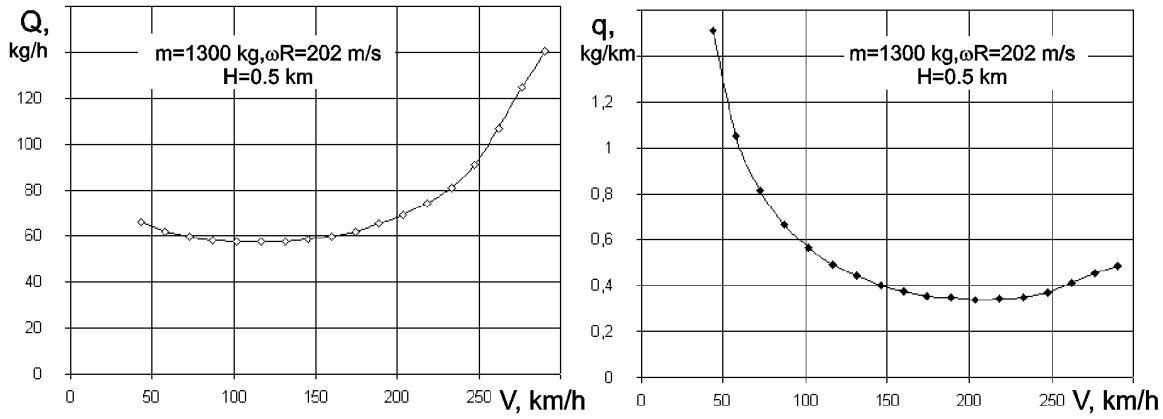


Fig. 10.8. Determination of Airframe Parasitic Drag Under Action of Vertical Flow



a b

Fig. 10.9. Hourly (a) and Kilometer (b) Fuel Consumption in Level Flight

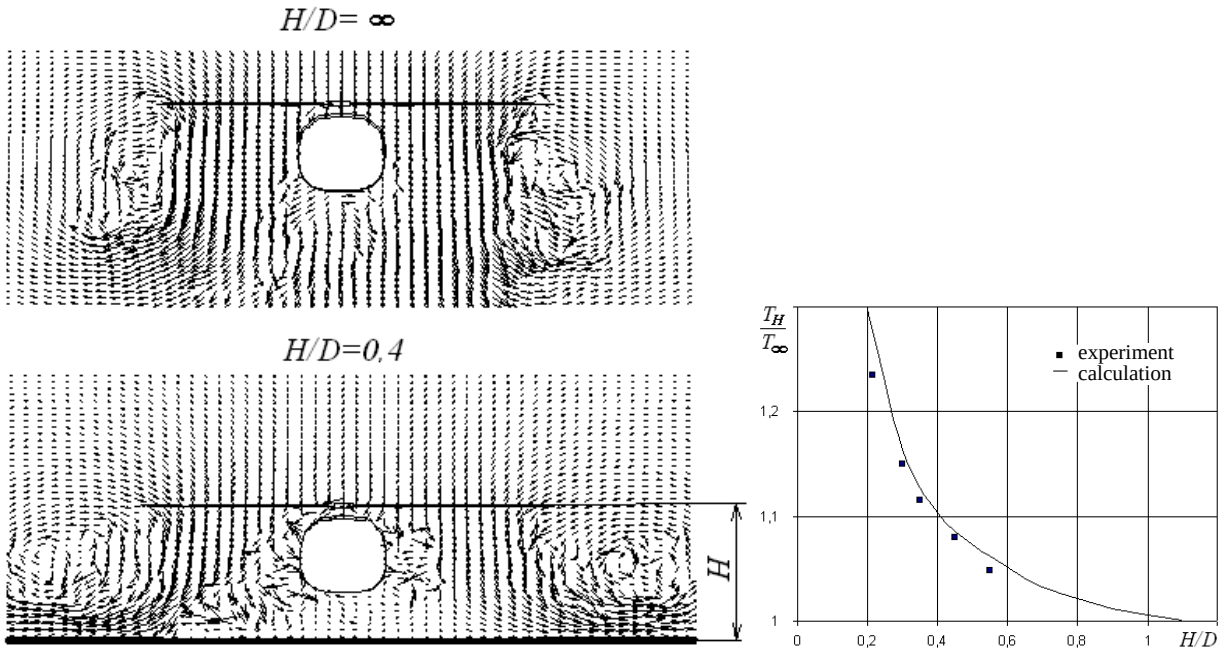
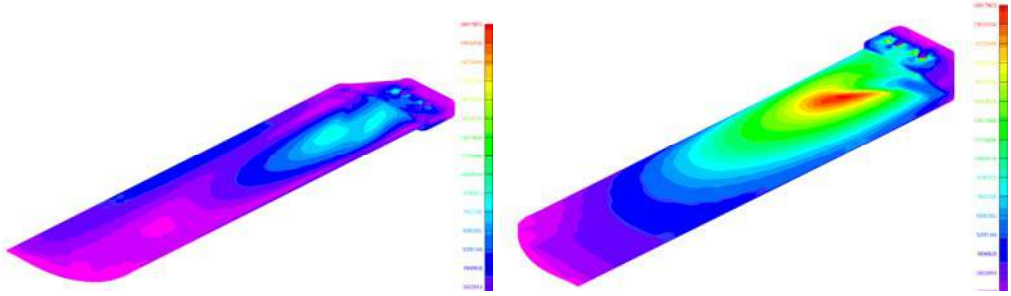


Fig. 10.10. Influence of Hovering Altitude on Main Rotor Thrust



a b

Fig. 10.11. Distribution of Mises Equivalent Stresses over Upper (a) and Lower (b) Tail Rotor Blade Surface

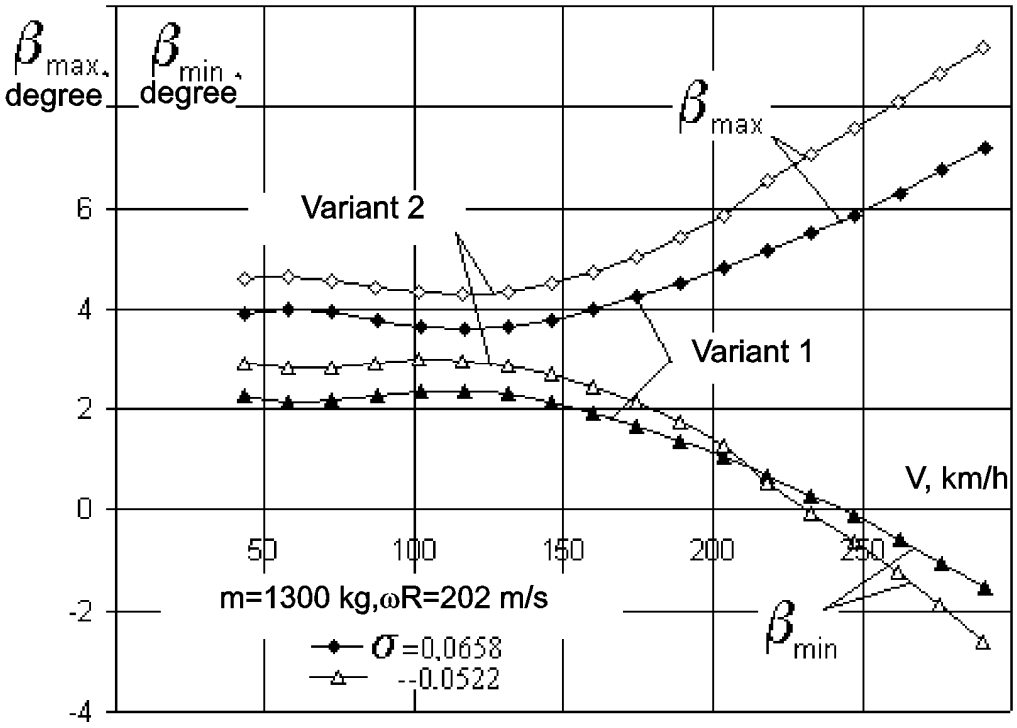


Fig. 10.12. Influence of Flight Speed on MR Blade Flapping Amplitude

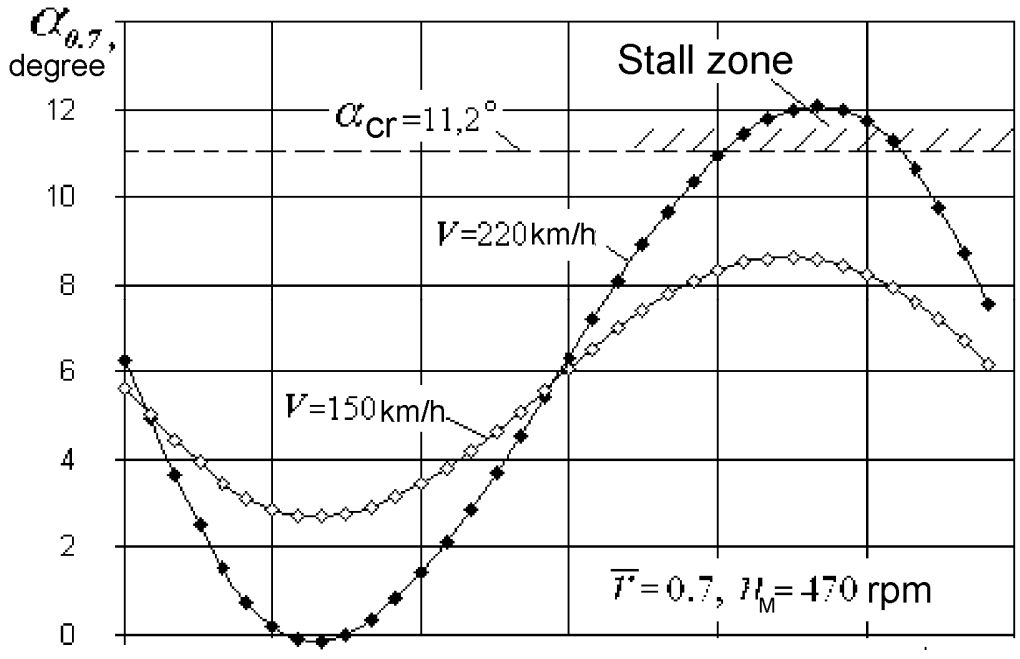


Fig. 10.13. Influence of Flight Speed on Change
of Characteristic Blade Section Angle-of-Attack. Stall

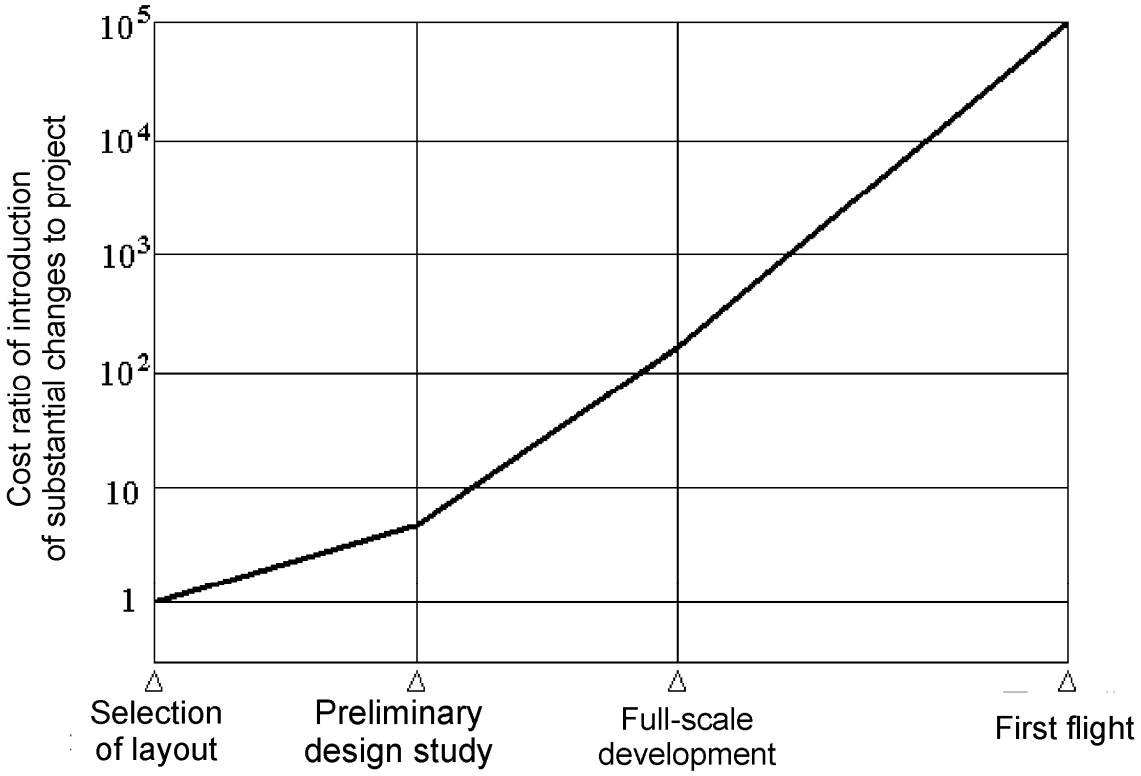


Fig. 10.14. Cost of Project Changes at Various Stages of Completion

10.1.2. Subsystem for Estimation of Main Rotor Aerodynamic Characteristics

Work conditions of the helicopter main rotor significantly differ from the wing or flow conditions about a wing or a propeller during the main flight regimes. These differences have led to origination of an individual branch of science – the main rotor theory, which considers MR flow conditions and the forces and moments acting upon it.

In helicopter steady flight the relative velocity of the air flow oncoming to different blade sections (elements), and accordingly, aerodynamic forces of these areas varies in time according to complex periodic law. Under the influence of periodic loads blade performs complex spatial motion, fluctuating in the hinges and being deformed as an elastic body. Consequently, MR theory in general is the theory of unsteady aerodynamics of curvilinear motion of carrier surface.

Unlike wing theory MR theory significantly differs due to the fact that each rotor blade not only creates its own velocity inductive field, but is forced to work in the velocity field induced by the other blades. The problem of determining the velocity field induced by all blades is the most difficult part of the main rotor theory.

The founder of the rotors theory is N.E. Zhoukovsky (1912-1918), who in his work has shown that the impact of MR blades on the environment can be replaced with the influence of the system consisting of attached and free vortices. These vortices come

from the blade and move along the flow lines in relative motion (Fig. 10.15). Validity of the rotor vortex theory has been confirmed by numerous experiments.

The MR general theory is usually divided into different models, which differ mainly in the method of modeling of free vortex wake behind the rotor.

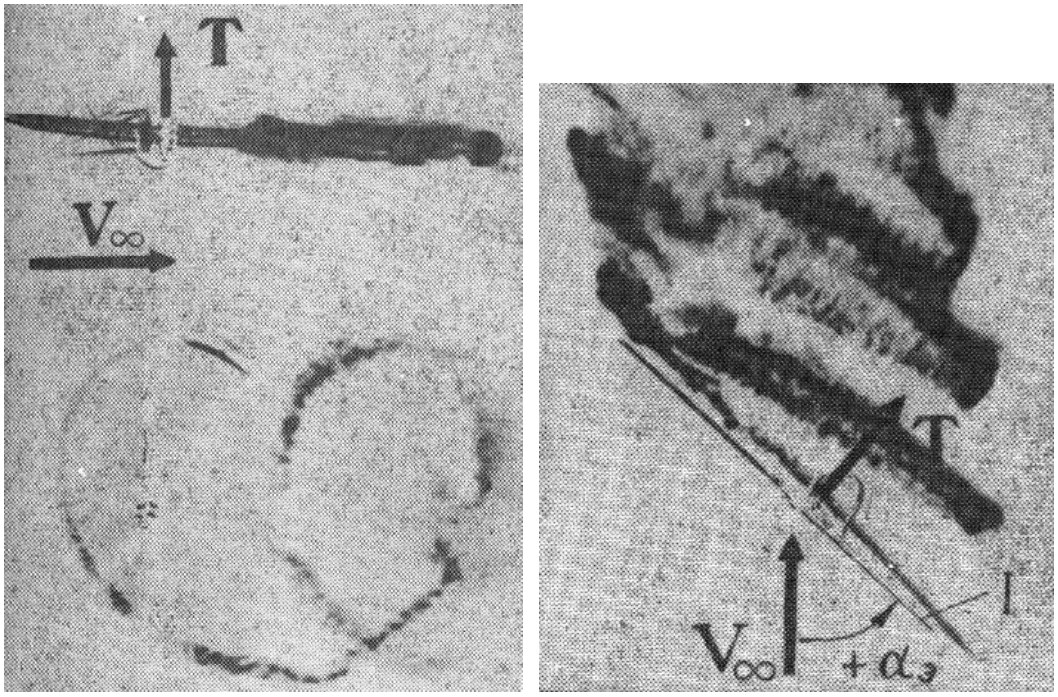


Fig. 10.15. Visualization of Main Rotor Vortex System

10.1.2.1. Mathematical Models of MR Theory

When solving the practical problems of determination of aerodynamic loads acting on the rotor and the associated inductive velocity field the mathematical models and theories are used that could be divided into linear, quasi-linear and non-linear (Fig. 10.16).

Linear models and linear methods of aerodynamic estimation suggest that the vortex system in the rotor wake is defined as a rigid system of cylindrical spiral vortices. The cylinder axis is directed along the vector of the approach flow speed and pitch of the spirals is defined by the value of the approach flow velocity.

Linear methods and approaches are implemented in the rotor vortex and disk theories. The MR disk theory replaces the rotor with the active disk, and the field of inductive velocity is averaged over time. Vortex and disk linear models make it possible to obtain exact analytical solutions for the field of inductive velocity, and accordingly, quick solution to the problem of calculating the MR aerodynamic loads.

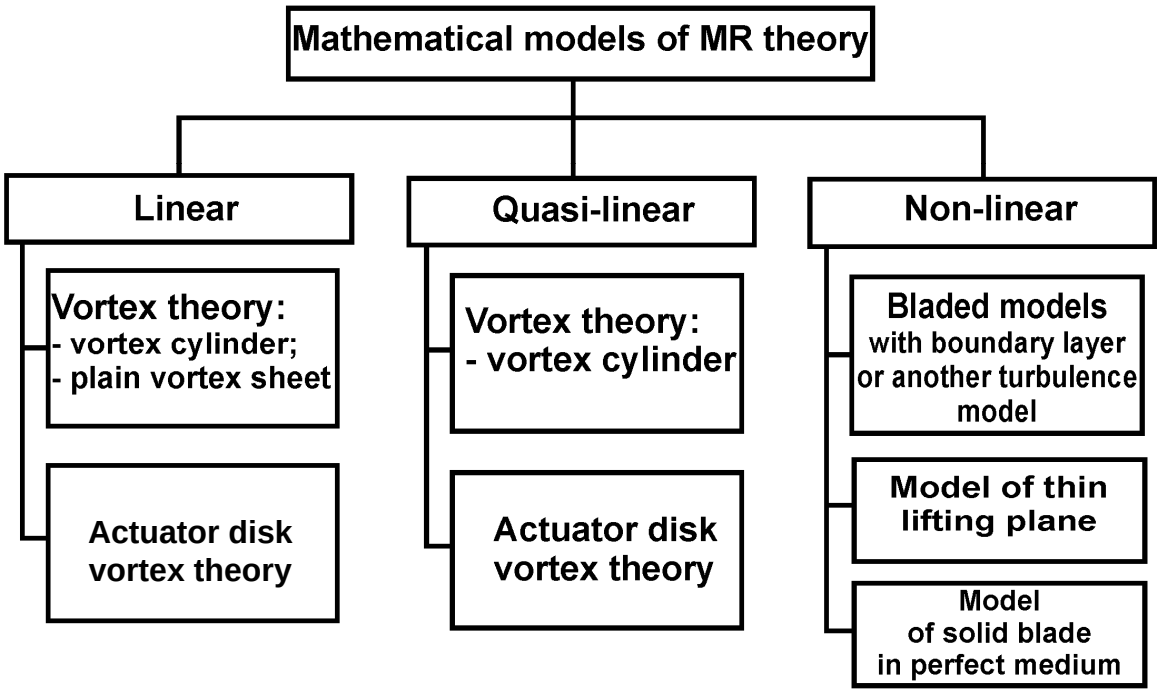


Fig. 10.16. Mathematical Models in Main Rotor Theory

Quasi-linear methods and models in contrast to the linear ones in determining the wake vortex spirals pitch account the geometric sum of the velocities of the approach flow and the vertical component of the inductive speed averaged over rotor disk. In the quasi-linear mathematical models the vortex and disk theories are usually used.

Nonlinear mathematical methods of estimating the MR aerodynamics feature in that the determination of the inductive velocity field consider their instantaneous values. The instantaneous value of the inductive velocity at an arbitrary point in space changes over time with the evolution of the vortex wake behind the rotor. The vortex wake behind the rotor moves with the local flow velocity being a sum of the approach flow velocity vector and the induced velocity vector. Thus, the wake is free and not strictly given and the wake structure (shape) is formed in the process of the flow evolution.

In the nonlinear mathematical models each blade and its vortex wake are simulated individually. Theories used in the non-linear models are called blade-type. The following representations of blade may be used in blade models :

– Bodily (volume) model;

– Thin lifting surface.

In the blade simulation there used the hydrodynamic features distributed over the blade surface in the form of a vortex layer, flow-source layer, dipoles, etc. In all the models there used the impermeability conditions, boundary condition of disturbances attenuation away from the rotor.

In contrast to the above listed approaches of nonlinear models those models occupy a special place, which use the Euler equations with the boundary layer or the Navier-Stokes equations with a particular model of turbulence. The complexity of creating turbulence model that adequately describes the compressible flow, stall and unsteady flow in the vicinity of the rotor, as well as great computer resources to meet these challenges, hindering the development of such approaches. Therefore, more simple methods are often used to solve practical problems to ensure the required accuracy.

10.1.2.2. Block Diagram of Main Rotor Subsystem Aerodynamic Estimation

The forces and moments from the main rotor are transmitted to the rotor shaft and further to the reducer. Thus the rotor shaft perceives not only the rotor torque but also other components of the moments acting on the hub from the blades. The loads acting on the MR hub may be determined only from the MR estimated aerodynamics.

Therefore let's consider in detail the structure of the "Aerodynamic estimation of main rotor" block in the block diagram given in Fig. 10.1.

Depending on the objective of the assigned task and the helicopter flight regime in the aerodynamic estimation block one or more programs are executed as shown in Fig. 10.17.

In accordance with the above block diagram calculations are carried out for MR axial regimes using the following theories:

1. Main rotor quasi-linear vortex theory.
2. Main rotor impulse theory.
3. Main rotor nonlinear vortex model.

On the regimes of forward flight the MR aerodynamic estimation subsystem uses computing programs based on the following theories and methods:

1. Linear vortex rotor theory.
2. Quasi-linear vortex theory.
3. Method of discrete vortices, sometimes referred to as the method of discrete features.

The main terms of applicable rotor theories and numerical methods are discussed in item 10.1.2.1. Further, we will consider only MR quasi-linear vortex model which provides calculation of aerodynamic and inertial loads acting on the MR blade and hub in general. In this regard you need to know the basic designations of MR geometrical and kinematic parameters that will be used hereinafter.

10.1.2.3. Main Rotor Basic Geometry and Kinematic Characteristics

When solving various problems of the main rotor aerodynamics and dynamics of its blades the *stability axes* $O_{MR}x_{MR}y_{MR}z_{MR}$ are used (Fig. 10.18).

The axes origin (point O_{MR}) is located in the point where the rotor shaft axis intersects the plane passing through the hub center and perpendicular to the shaft axis. This plane is called *MR constructive plane of rotation* and designated as CPR. The vertical axis $O_{MR}y_{MR}$ coincides with the MR axis of rotation and is directed upward in the positive direction of the rotor thrust.

The longitudinal axis $O_{MR}x_{MR}$ is perpendicular to the axis $O_{MR}y_{MR}$ and is directed forward on the flight along the incident air flow velocity vector projection V_{∞} on CPR (Fig. 10.18).

Transverse axis $O_{MR}z_{MR}$ lies in the rotor plane of rotation, is perpendicular to the axes $O_{MR}x_{MR}$ and $O_{MR}y_{MR}$ and directed towards the retreating blade. The blade is called the *upcoming* (advancing), if it moves toward the approach flow. If the movement of the blade matches the direction of the approach flow, it is called *retreating* (lagging).

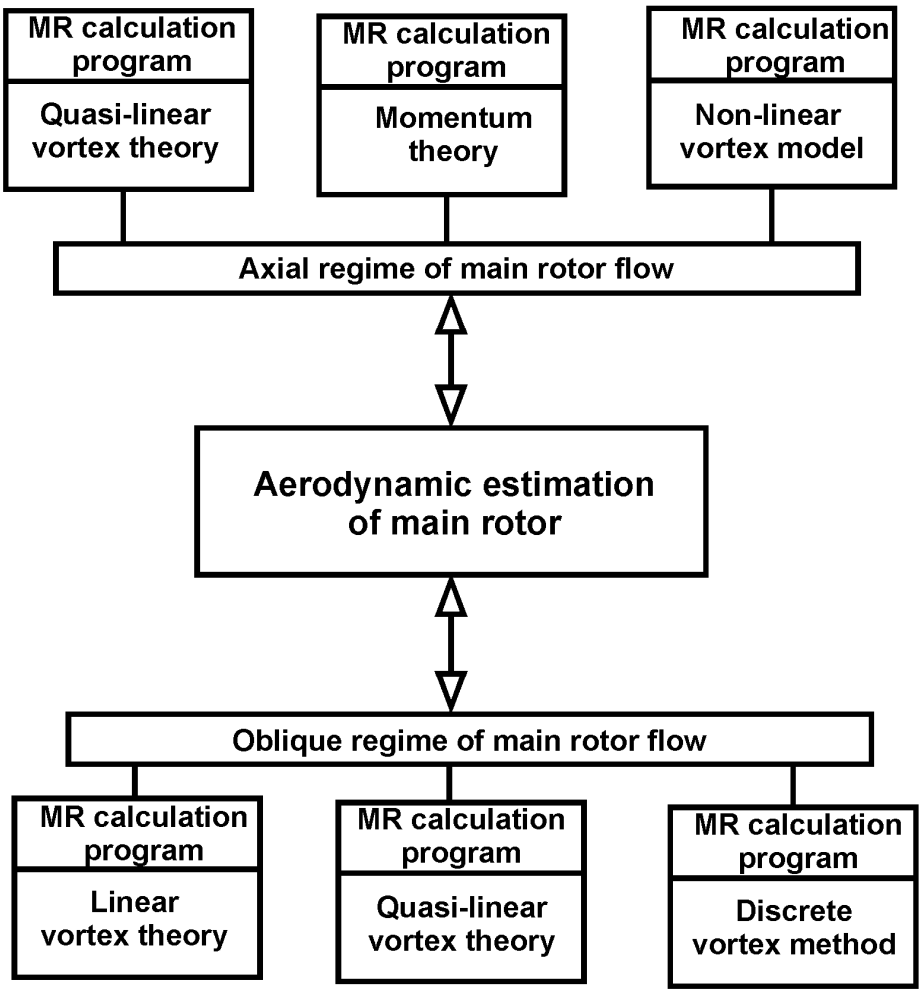


Fig. 10.17. Block Diagram of Rotor Aerodynamic Estimation Subsystem

Depending on the blade direction of rotation main rotors are divided into rotors of the right-hand and left-hand rotation. When viewed from above (from the end of the axis $O_{MR}y_{MR}$) and MR blades rotate counter-clockwise – this is the left-hand rotation rotor, if clockwise – right-hand rotation rotor. Depending on the MR rotation direction the stability axes can be right (for right-hand rotation main rotors used in the UH-1A, R.22, R.44 helicopters etc.) or the left (for left-hand rotation rotors on all helicopters of Mil company). Since most modern helicopters have MR rotating clockwise then further we will use left-handed coordinate system.

Helicopter rotor consists of a number of blades mounted on the hub in a certain way, and typically it is described by a number of geometric parameters.

1. MR *diameter* is a diameter of the circle, which is described by the moving blade tips during rotor operation in place. Often, instead of considering MR diameter its radius is used $R = D/2$ (Fig. 10.19).

The radius r of a blade arbitrary cross section determines its distance from the MR axis of rotation. In practice it is convenient to use relative radius $\bar{r} = r/R$, which ranges $0 \le \bar{r} \le 1.0$.

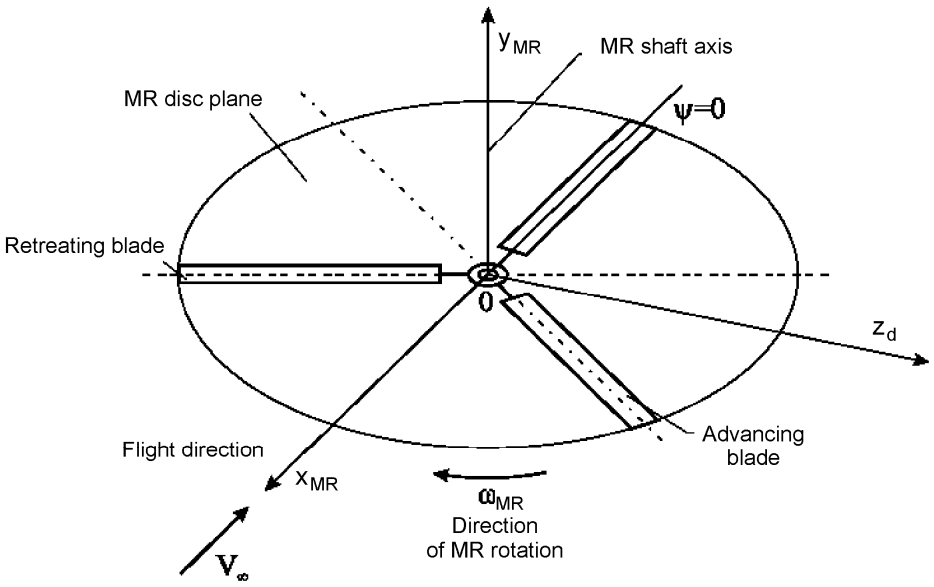


Fig. 10.18. Stability Axes of Left-hand Main Rotor

2. The *blade root section radius* r_0 is the distance from the rotor axis up to begin of the blade profiled part. Typically, relative value is used $\bar{r}_0 = r_0/R$.

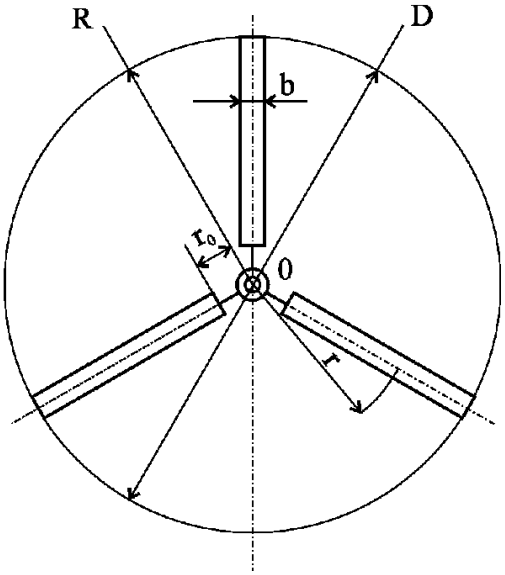


Fig. 10.19. Main Rotor Geometrical Parameters

3. *Characteristic radius* of blade. For MR blades value of the relative radius $\bar{r} = 0.7$ is called the blade characteristic radius as the blade characteristics in this section are similar to the average values for the whole blade.

4. *Disc area* F_{MR} – area of a disc having diameter D, then

$$F_{MR} = \pi D^2 / 4 = \pi R^2 .$$

5. *Number of blades* K_{bl}. Number of MR blades depends mainly on the helicopter flight weight and its purpose. For modern helicopters K_{bl} can vary from 2 to 5.

6. *Blade planform.* On modern helicopters the blades having rectangular planform are the most widely used. These blades have blade chord constant along its length and are the most reliable and efficient. They are subject to less stringent requirements for a geometric twist and airfoil thickness distribution law along the blade radius.

7. *Solidity ratio* σ_{MR}. It characterizes the degree of filling the MR disc area and is the ratio of the total area $K_{bl}F_{bl}$ of all the blades to the disc area

$$\sigma_{MR} = \frac{K_{bl}F_{bl}}{F_{MR}} = \frac{K_{bl}b}{\pi R}, \qquad (10.1)$$

here $F_{bl} = Rb$ – reference area of one blade.

8. *Relative offset* of flapping and drag hinges for articulated rotor

$$\bar{l}_{fl.h} = \frac{l_{fl.h}}{R}, \bar{l}_{d.h} = \frac{l_{d.h}}{R}, \qquad (10.2)$$

where $l_{fl.h}, l_{d.h}$ – distance from the rotor axis of rotation to the corresponding hinge (Fig. 10.20). For modern helicopters $\bar{l}_{fl.h} = 0.02...0.05, \bar{l}_{d.h} = 0.04...0.06$.

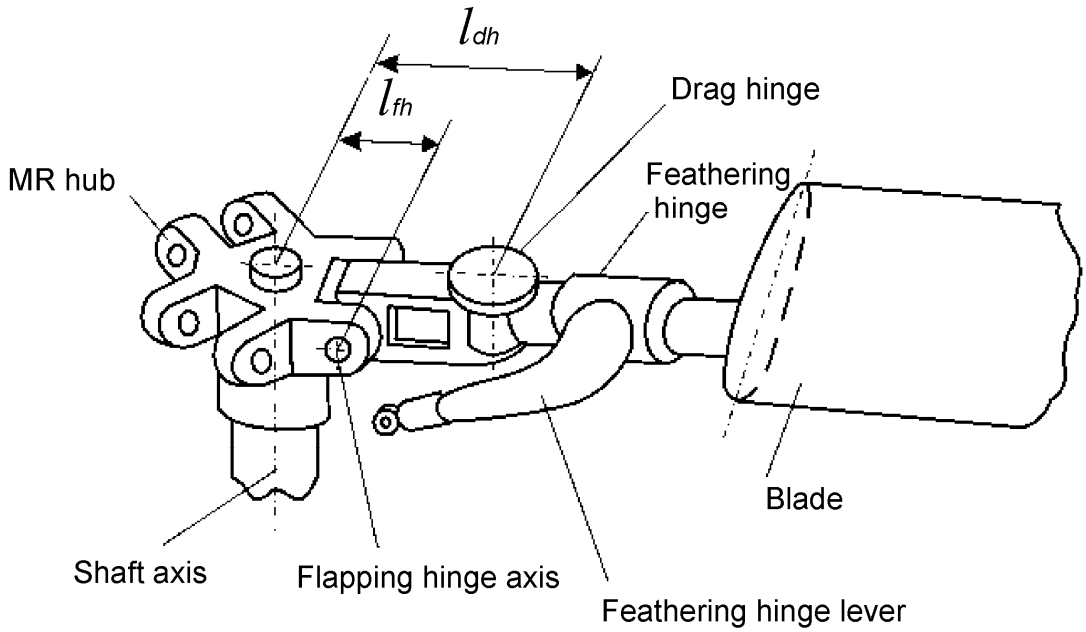


Fig. 10.20. Articulated Main Rotor Hub

9. *Blade airfoil.* An airfoil is a contour shape formed by the plane intersecting the blade perpendicularly to its longitudinal axis. Blade airfoil similar to a wing airfoil is characterized by a number of geometric parameters (Fig. 10.21):

- *Blade chord* b is the line connecting the two most distant point of the airfoil;
- *Airfoil center line* is a line drawn through the middle of the segments perpendicular to the chord and limited by the upper and lower airfoil arches. The center line of the symmetric airfoil coincides with the chord, the center line of the asymmetric airfoil is curved;
- *Thickness ratio* $\bar{c}$ is a ratio of maximum airfoil thickness c_{max} to its chord, which sometimes is given in percents;

$$\bar{c} = \frac{c_{max}}{b} \cdot 100\%; \tag{10.3}$$

- *Relative coordinate of airfoil maximum thickness* $\bar{x}_c$, determines the position of the maximum airfoil thickness on the chord

$$\bar{x}_c = \frac{x_c}{b} \cdot 100\%; \tag{10.4}$$

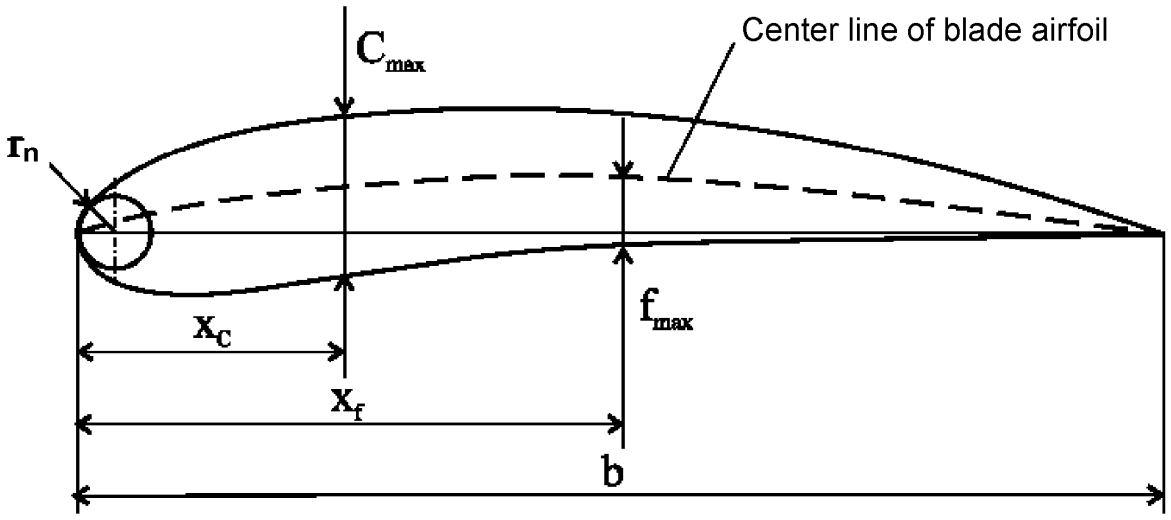


Fig. 10.21. Blade Airfoil Geometry

- *Relative maximum camber* $\bar{f}$, represents the ratio of the maximum distance between the center line and airfoil chord f_{max} (Fig. 10.21) to the chord length, which is expressed as a percentage

$$\bar{f} = \frac{f_{max}}{b} \cdot 100\%; \tag{10.5}$$

- *Relative coordinate of airfoil maximum camber*

$$\bar{x}_f = \frac{x_f}{b} \cdot 100\% ; \tag{10.6}$$

- *Relative radius of the airfoil nose camber*

$$\bar{r}_{MR} = \frac{r_{MR}}{b}. \tag{10.7}$$

Blade airfoils are combined in series; each series has certain laws of thickness variation and the curvature (concavity) along the profile chord. For blades of modern light helicopters mainly symmetric or asymmetric airfoils with small curvature and well-rounded nose are used. The airfoil thickness ratio $\bar{c}$ = 6…16% and usually it increases towards the root.

10. *Blade section angle* φ, the angle between the blade chord in this section and rotor CPR (Fig. 10.22, a). Blade section angle at the characteristic radius $\bar{r}$ = 0.7 at the neutral position of controls is called the angle of MR blade setting or collective pitch, and is denoted φ_0.

11. *Blade geometrical twist*. It is necessary for a more uniform distribution of aerodynamic load spanwise. For this purpose, the blade section angles are reduced to

its end, i.e. sections are constructively turned relative to each other at an angle $\Delta\varphi$ (Fig. 10.22, b). Blade twist $\Delta\bar{\varphi}_{bl}$ is defined as the difference between the section angles at the blade tip $\left(\bar{r}=1\right)$ and root $\left(\bar{r}=\bar{r}_0\right)$ radiuses $\Delta\bar{\varphi}_{bl}=\varphi_{\bar{r}=1}-\varphi_{r_0}$. Modern helicopters mostly have geometrical twist $\Delta\bar{\varphi}_{bl}$ of (7…9°) at most.

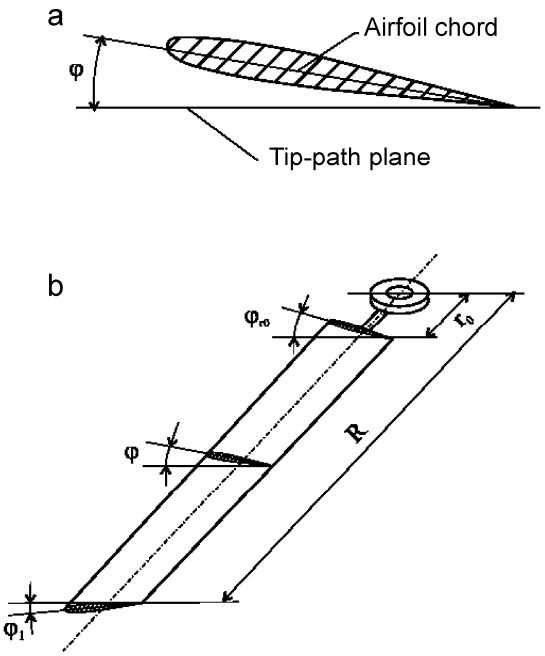


Fig. 10.22. Blade Section Angle

MR kinematic characteristics determine the nature of translational, rotational and vibrational motion of the blades and rotor in general. The kinematic characteristics of helicopter main rotor include the following parameters:

1. *Rotational frequency*, which represents the number of rotor revolutions per minute (n_{MR}, rpm), sometimes revolutions per second are used (n_s, rps). The angular velocity of MR rotation is related to its rotational speed by the known relation $\omega_{MR}=2\pi n_s=\pi n_{MR}/30$.

2. Blade *tip speed* $\omega_{MR}R$ and *circumferential Mach number* $M_0=\omega_{MR}R/a$ (a – speed of sound at the given atmospheric conditions). MR blade tip speed $\omega_{MR}R$ (or M_0) has significant influence on the helicopter flight characteristics. Circumferential speed $\omega_{MR}\cdot R$ of modern helicopters is about 180…230 m/s.

3. *Air velocity of the approach flow* V. Based on the known principle of reversibility it does not consider the main rotor motion with velocity V (helicopter flight speed) relative to the stationary air, but the air stream that approaches to the rotating

rotor with speed V. For the convenience of the flight regime setting the helicopter dimensionless parameter of the approaching flow rate $\overline{V} = V / \omega_{MR} R$ is used.

4. *Main rotor angle-of-attack* α_{MR}. MR angle-of-attack means the angle enclosed between the vector of the approach flow speed V and CPR (Fig. 10.23, a). The value of the rotor angle-of-attack α_{MR} is considered positive $(\alpha_{MR} > 0)$, if the air flow is approaching to the main rotor from below, and α_{MR} is negative $(\alpha_{MR} < 0)$, if the air flow is approaching to the main rotor from above.

Depending on MR orientation relative to the vector of the approaching flow rate V (i.e. MR angle-of-attack), there are two basic regimes of the rotor operation - axial flow and oblique flow.

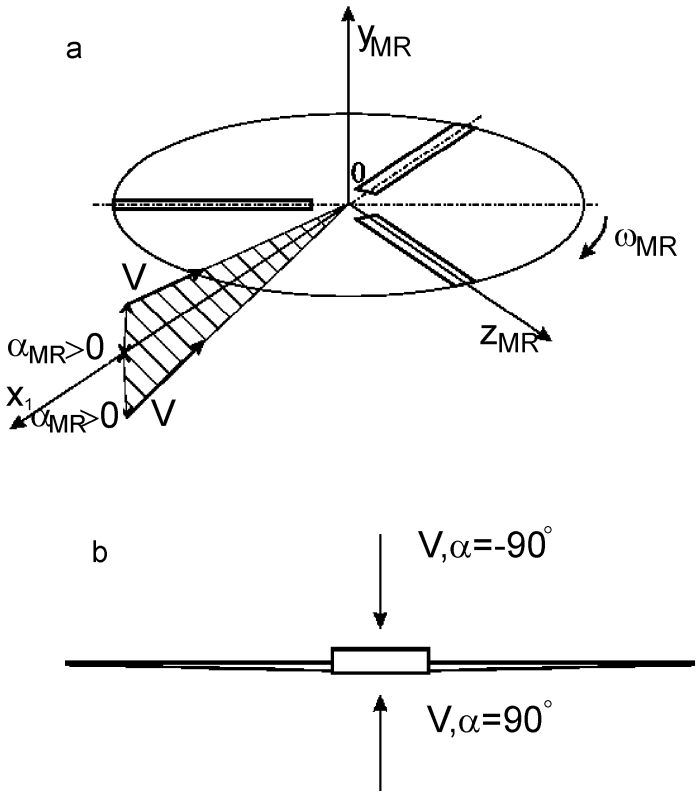


Fig. 10.23. Helicopter Rotor Angle-of-Attack

Under the axial flow conditions the air flow approaches perpendicularly to MR plane of rotation (Fig. 10.23,b). This regime is observed in hover, during vertical climb (α_{MR} = -90°) or vertical descend (α_{MR} = 90°).

Oblique flow conditions correspond to such flight conditions, when the air flow approaches to MR at a certain angle ($\alpha \neq \pm 90°$). Oblique flow regime corresponds to helicopter horizontal flight, climb $(\alpha_{MR} < 0)$ or descend $(\alpha_{MR} > 0)$.

5. *Characteristic of rotor regime* μ. It represents the ratio of the velocity vector V projection on MR plane of rotation $V_x = V \cdot \cos\alpha_{MR}$ (Fig. 10.24) to blade tip speed:

$$\mu = \frac{V \cdot \cos\alpha_{MR}}{\omega_{MR} R}. \qquad (10.8)$$

Characteristic of rotor regime varies within the wide range. With that μ=0 corresponds to axial flow about main rotor. Under small angles of attack (α_{MR} </10…14°/) it can be assumed that $\cos\alpha_{MR} \approx 1.0$, then $\mu \approx \overline{V}$.

6. *The blade azimuth* ψ, which is measured in the MR direction of rotation between the longitudinal blade axis and blade initial position above the tail boom axis (Fig. 10.25). Blade azimuth ψ defines blade position on MR disc. Blade is named *advancing* at $0 < \psi < 180^\circ$ (in left part of a left-hand rotation rotor disc) and *retreating* at $180^\circ < \psi < 360^\circ$ (in the right part of the disc, Fig. 10.18).

7. *Blade flapping angle* β_{bl} of main rotor determines the angular movement of the blades in flapping hinge regarding the rotor plane of rotation and is considered positive ($\beta_{bl} > 0$) when the blade moves upward with respect to CPR (Fig. 10.26).

8. *Blade swinging angle* ξ determines the blade angular movement relative to the drag hinge axis. It is measured from the line passing through the rotor axis and flapping hinge axis and the blade longitudinal axis when β=0 (Fig. 10.27). Swinging angle ξ is positive when blade deflects opposite to MR rotation.

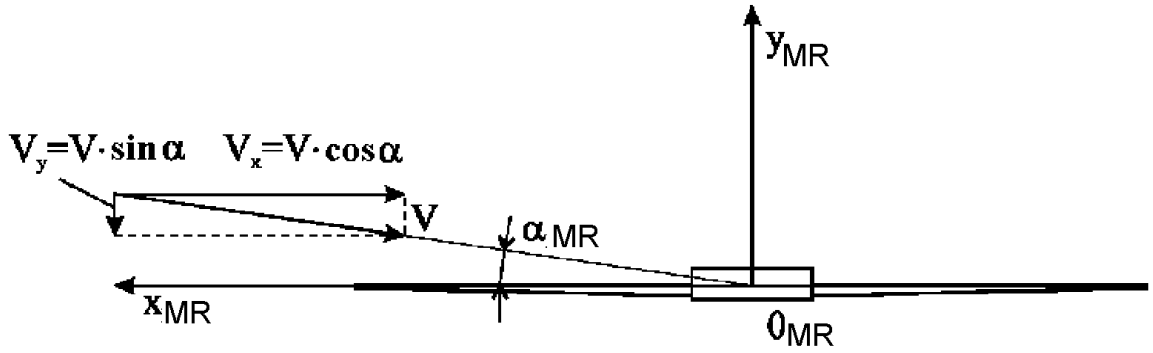


Fig. 10.24. Components of Approaching Flow Speed Vector

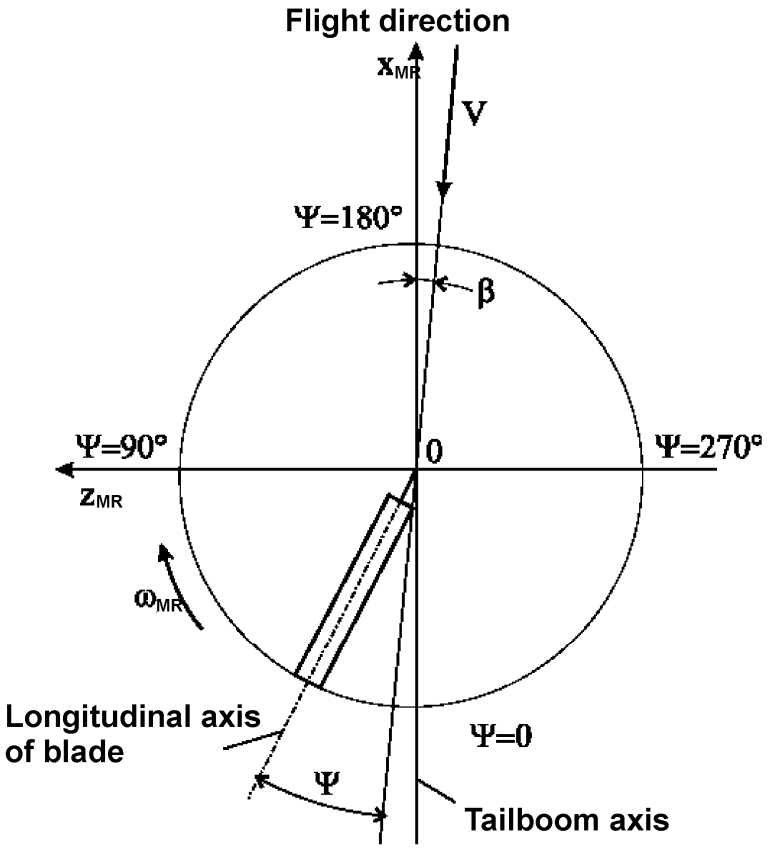


Fig. 10.25. Blade Azimuth Angle

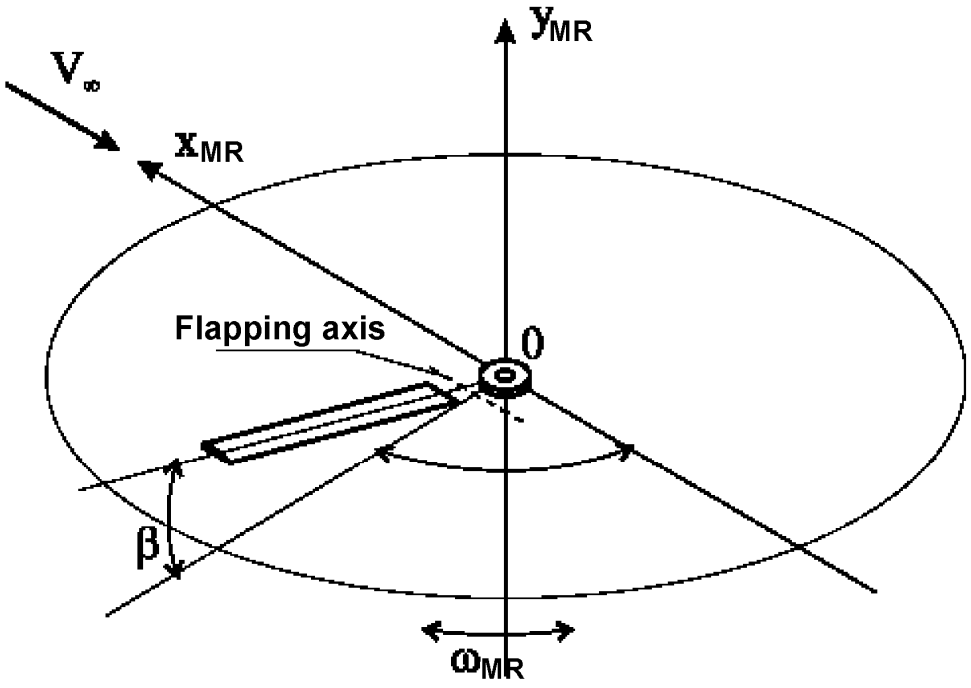


Fig. 10.26. Blade Flapping Angle

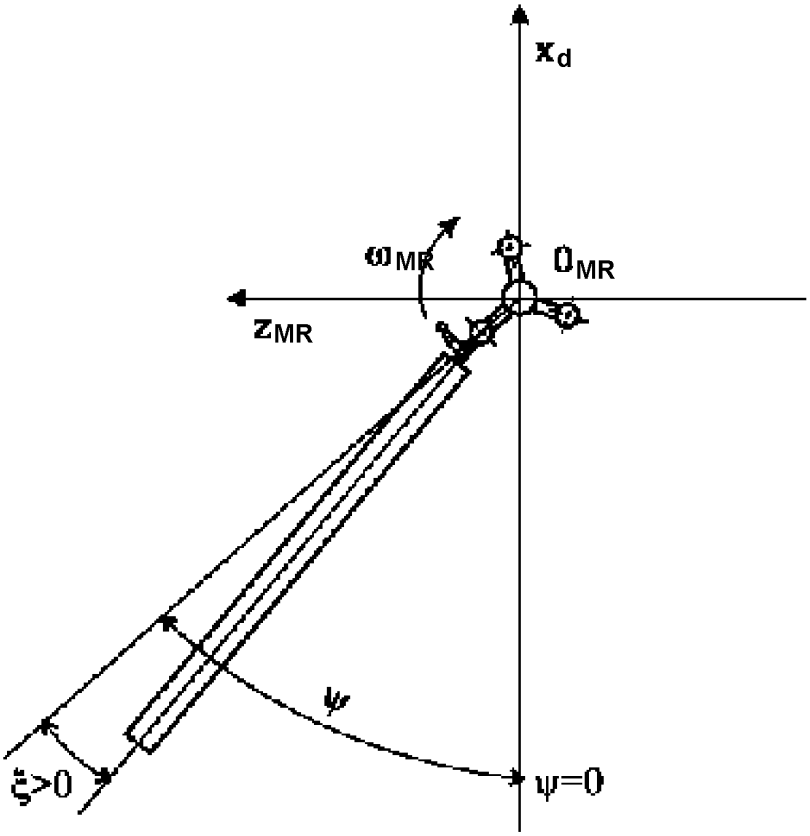


Fig. 10.27. Blade Swinging Angle

10.1.3. Rotor Aerodynamic Characteristics Design Procedure

Design procedure for rotor aerodynamic characteristics with forward speed is based on the classic MR theory developed by M.L. Mil and M.N. Tishchenko. MR aerodynamic characteristics are calculated together with a numerical integration of the blade flapping equation and taking into account distribution of the inductive velocity field over MR disk. The proposed method of calculation provides a wide range of MR aerodynamic characteristics with highly reliable results at the lowest consumption of the computer resources.

10.1.3.1. Background and Basic Assumptions of Computing Method

The proposed computing method takes into consideration the following geometric and kinematic parameters of the rotor:

- Blade geometry twist $\Delta\varphi = f(\bar{r})$ (Fig. 10.22, b);
- Blade planform, that is given by the relevant law of chord change along the blade radius $\bar{b} = b/R = f(\bar{r})$;

- MR number of blades K_{bl};
- Blade profile layout, which suggests presence of three sections, consisting of airfoils of different series (Fig. 10.28);

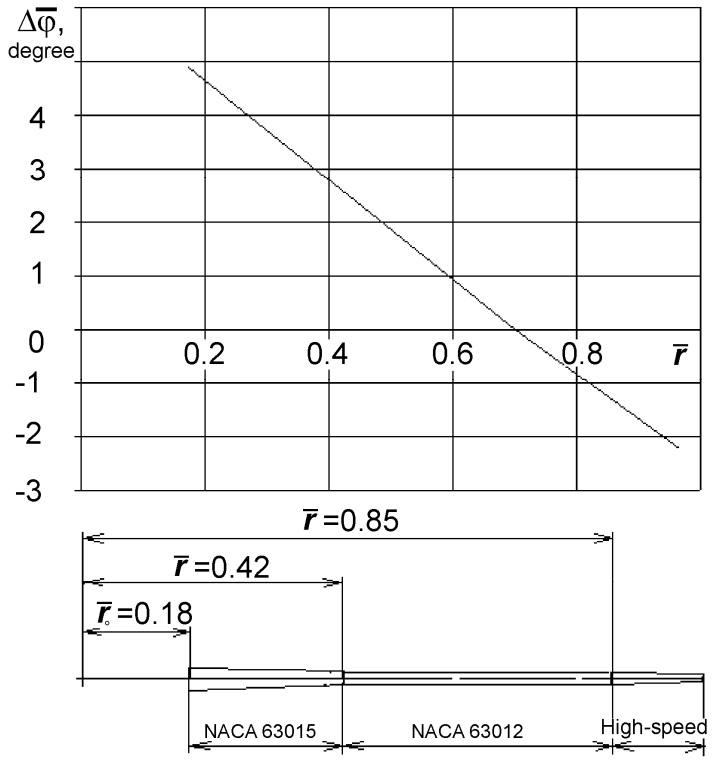


Fig. 10.28. Airfoils and Twist of Main Rotor Blades

- Main rotor radius *R*;
- Mass-inertial characteristics of the blade, which are determined using well-known expressions:

$$m_{bl} = \int_{r_0}^{R} dm_{bl} \cdot dr, \qquad J_{fl.h} = \int_{l_{fl.h.}}^{R} dm_{bl} \cdot r^2, \qquad S_{fl.h} = \int_{l_{fl.h}}^{R} dm_{bl} \cdot r,$$

$$J_0 = \int_{0}^{l_{fl.h}} dm_{bl} \cdot r^2, \qquad S_0 = \int_{0}^{l_{fl.h}} dm_{bl} \cdot r, \qquad (10.9)$$

where dm_{bl} – actual blade weight per unit length;

J_0, S_0 – inertial characteristics of the blade sleeve;

- The flapping hinge stiffness, which is characterized by the derivative $M^{\beta}_{fl.h}$ and determines the moment acting on the blade when it is deflected by the unit angle from the neutral position. In accordance with this definition, the value of the blade elastic moment is found from the expression

$$M_{ctl} = -M^{\beta}_{fl.h} \cdot \beta; \quad (10.10)$$

- The value of the blade flapping compensator coefficient k, the function of which is to change the blade setting angle by the amount of

$$\Delta\varphi_{\beta} = -k \cdot \beta. \quad (10.11)$$

When calculating the rotor aerodynamic characteristics we will suggest that the rotor is under conditions of steady horizontal flight with a constant speed. This regime is characterized with the following parameters:

- Atmospheric conditions (air density ρ, air kinematic viscosity ν, acoustic speed a);
- Flight speed V;
- Rotor number of revolutions n_{rot} or blade tip speed ωR;
- Main rotor angle-of-attack α_{rot}, that is the angle between the velocity vector and rotor constructive plane of rotation (CPR) (Fig. 10.29);
- Main rotor collective pitch φ_0, which corresponds to the blade section angle with respect to CPR at zero flapping angle. It is considered positive when leading edge of the blade section flaps upward (Ref. Fig. 10.22);
- Components of the rotor cyclic pitch θ_1 and θ_2, whereby cyclic change of the blade setting angle with regard to its azimuth position is determined by the expression

$$\Delta\varphi_c = \theta_1 \cdot \cos\psi + \theta_2 \cdot \sin\psi, \quad (10.12)$$

Where θ_1 and θ_2 are blade setting angles relatively to CPR at azimuth ψ=0 and ψ=π/2 correspondingly, then under action of the cyclic control and presence of flapping regulator the blade section angle change is described by the equation

$$\varphi = \varphi_0 + \Delta\varphi + \Delta\varphi_{\beta} + \Delta\varphi_y = \varphi_0 + \Delta\varphi - k\beta + \theta_1 \cdot \cos\psi + \theta_2 \cdot \sin\psi. \quad (10.13).$$

The main criteria of similarity, which, when fulfilled, make possible to generalize calculation results for the range of flight regimes, are the following:

- Similarity by air viscosity;
- Similarity by compressibility;
- Dynamic similarity;
- Kinematic similarity.

The first criterion requires equality of Reynolds numbers in the blade sections in simulation and real flight to ensure similarity

$$Re = \frac{Ubr}{\nu} = \frac{a}{\nu} bM. \quad (10.14)$$

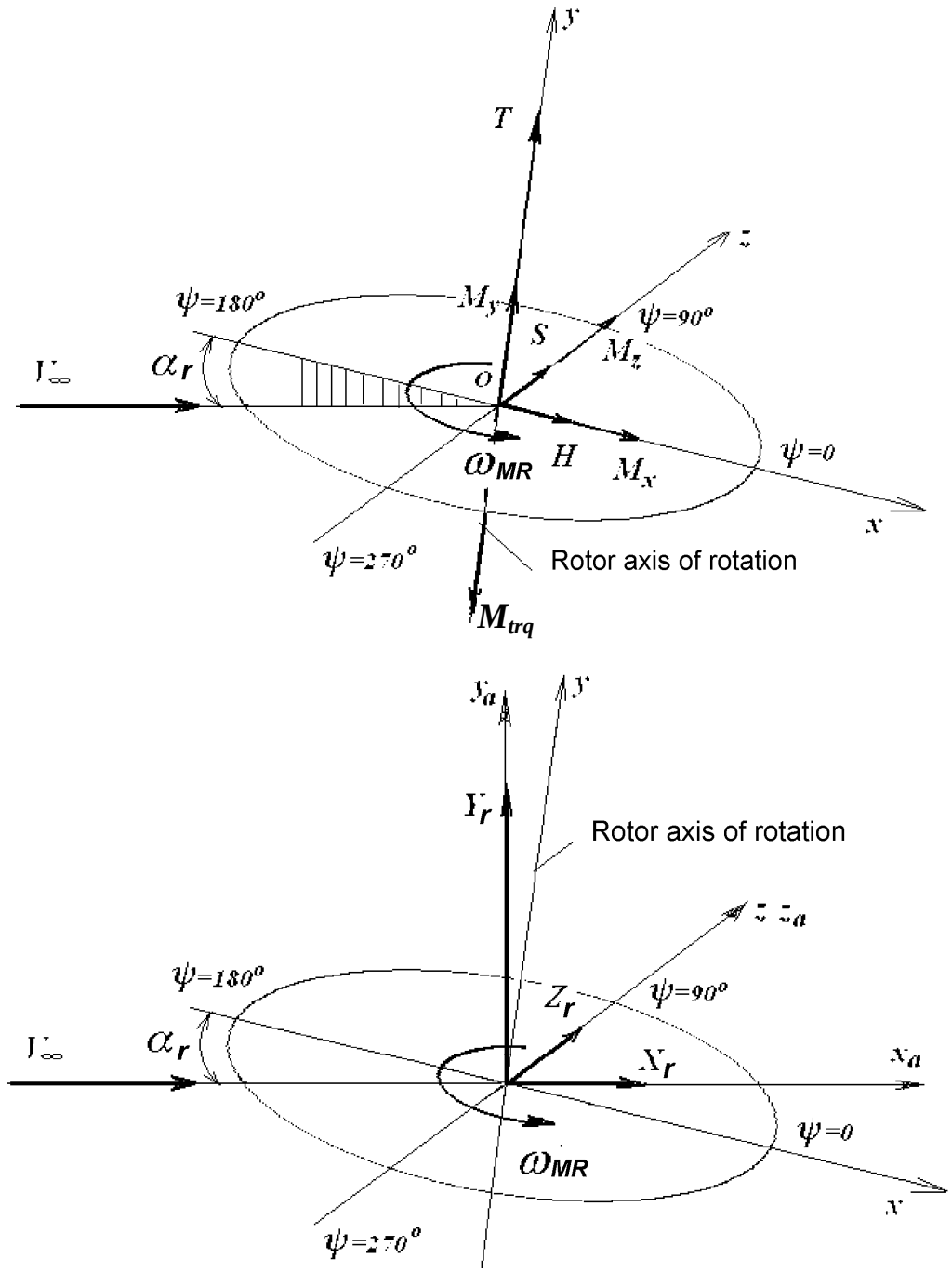


Fig. 10.29. Determination of MR Aerodynamic Forces and Moments:
a – in stability axes; b – in wind axes

The second criterion of similarity is provided by the equality of Mach numbers in calculations and in physical object. Average value of the blade tip Mach number is taken as the similarity parameter of MR flow regimes. The first and second MR flow regimes similarity criteria in the proposed numerical method are provided by the appropriate setting of the airfoil aerodynamic performances $C_{y_a}(\alpha, M, Re)$, $C_{x_a}(\alpha, M, Re)$ (Fig. 10.30).

$$M_R = \frac{\omega R}{a}. \qquad (10.15)$$

Main parameter to ensure dynamic similarity is the blade weight characteristic

$$\gamma = \frac{C_y^{\alpha} \cdot \rho \cdot b_{07} \cdot R^4}{2(J_{fl.h} - 2S_{fl.h} \cdot l_{fl.h} + m_{bl} l_{fl.h}^2)}, \qquad (10.16)$$

where $l_{fl.h}$ – flapping hinge offset;

b_{07} – blade chord at the characteristic radius $\bar{r}$=0.7.

Kinematic similarity of MR flow regimes requires equality of the blade section angles-of-attack and is ensured by the equality of the relative flight speeds $\overline{V}$, rotor angles-of-attack α_r, collective pitch φ_0 and cyclic pitch $\Delta\varphi_{cyc}$. Meeting of the first two criteria can be replaced by one

$$\mu = \overline{V} \cdot \cos\alpha_r, \qquad (10.17).$$

where μ – rotor operation regime coefficient. The problem of MR aerodynamic calculation is reduced to determining the arising thereon forces and moments for given input data and the following basic assumptions:

1. MR blade is absolutely rigid in bending and torsion and has two degrees of freedom – it turns in the flapping and feathering hinges.
2. The blade hinge moment does not affect total moments on MR hub.
3. The component of the hinge moment arisen due to elastic deformations of the blade is not considered.
4. The aerodynamic characteristics of blade cross section airfoils are independent of the stream slip angle along the blade and influence of centrifugal forces on the boundary layer.
5. The unsteady flow about blade sections is not considered, i.e. stationary hypothesis is accepted when determining the aerodynamic characteristics of the airfoils.
6. Stiffness of the control linkage is not considered.

Neglect of the control linkage elasticity and rigidity on the one hand, and the unsteady flow about the blade sections - on the other hand, have the opposite effect on the rotor characteristics and, above all, on its carrying properties. If the blade torsional stiffness limits the rotor load carrying capacity due to the reduction of the blade section angles, the unsteady flow about the blade sections leads to a positive effect of “delayed stall”. The experience of flight tests, as well as calculations taking into account the control linkage elasticity and without it have revealed that the rotor maximum carrying capacity using the assumptions adopted above, is close to the real, and hence, the above factors roughly compensate each other.

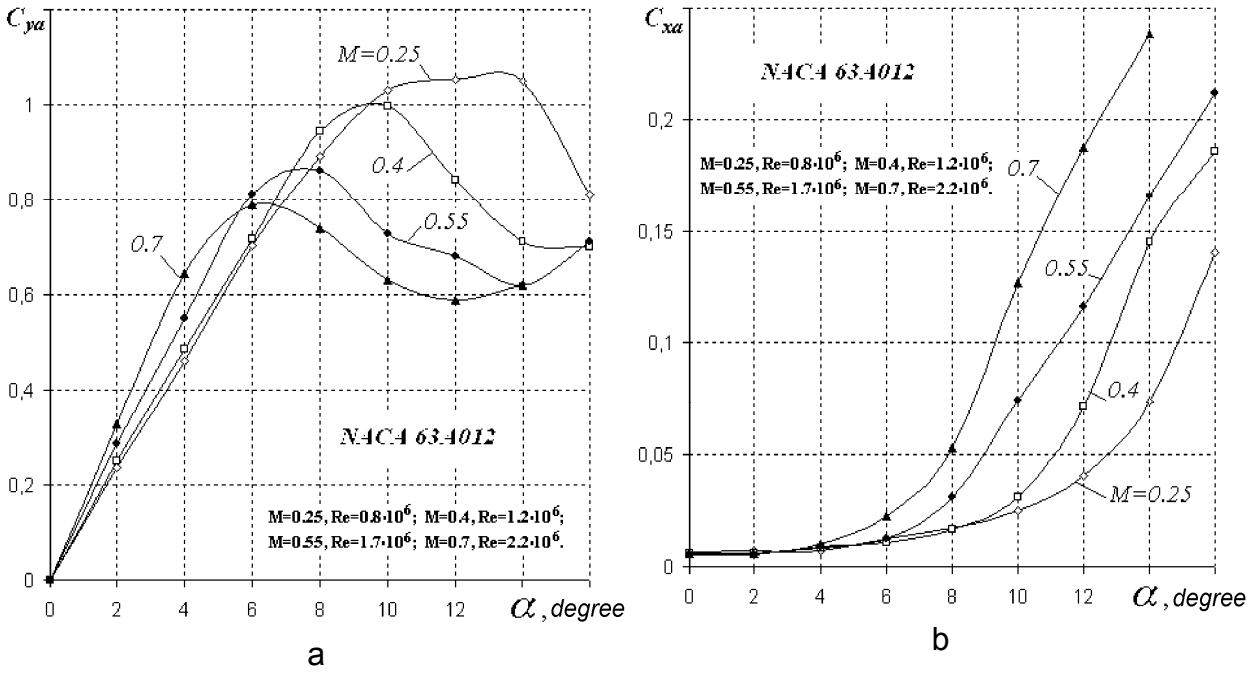


Fig. 10.30 Airfoil Aerodynamic Characteristics in Relation to
Mach and Reynolds Numbers:
a – lift coefficient vs angle-of-attack;
b – drag coefficient vs angle-of-attack

10.1.3.2. Main Rotor Force and Moment Coefficients

To determine the aerodynamic forces and moments acting on MR in forward flight, typically the coordinate system associated with the rotor is used (Fig. 10.29, a).

The coordinate system point of origin is located in the center of the hub (the point of intersection of CPR with axis of rotation). The O_x and O_z axes lie in CPR and are directed to the sides of azimuths ψ=0° and 90° respectively, axis O_y is directed upward along the axis of rotation. It should be noted that the axis O_z is directed towards advancing blade: for right rotating rotor – to the right (Fig 10.29, a), for left rotating rotor – to the left.

In the adopted coordinate system the rotor full aerodynamic force may be decomposed into the following components: rotor thrust T, longitudinal force H and lateral force S. Positive direction of forces is consistent with the positive direction of the O_{xyz} coordinate system axes.

The rotor full aerodynamic torque M may be also decomposed into components along the axes of the O_{xyz} coordinate system. Moment M_x is called the roll moment, moment with respect to the axis O_y is the moment of rotational drag M_y, the moment

with respect to the axis O_z is called pitching moment or longitudinal moment M_z. In accordance with sign rule the moment is considered positive if its vector coincides with the positive direction of O_{xyz} axes. When rotor rotates with a constant angular velocity the following equality is true

$$M_y - M_k = 0, \tag{10.18}$$

here M_k – torque to be applied to MR shaft to provide that condition $\omega_r = \text{const}$ is true.

Equation (10.18) shows that calculation of rotational drag moment is equivalent to calculation of required torque.

The values of the aerodynamic forces and moments acting on MR are calculated using dimensionless aerodynamic coefficients by the known formulas:

$$T = C_t \frac{\rho \cdot (\omega R)^2}{2} \cdot F_{MR}, \tag{10.19}$$

$$H = C_h \frac{\rho \cdot (\omega R)^2}{2} \cdot F_{MR}, \tag{10.20}$$

$$S = C_s \frac{\rho \cdot (\omega R)^2}{2} \cdot F_{MR}, \tag{10.21}$$

$$M_x = m_x \frac{\rho \cdot (\omega R)^2}{2} \cdot R \cdot F_{MR}, \tag{10.22}$$

$$M_y = M_k = m_k \frac{\rho \cdot (\omega R)^2}{2} \cdot R \cdot F_{MR}, \tag{10.23}$$

$$M_z = m_z \frac{\rho \cdot (\omega R)^2}{2} \cdot R \cdot F_{MR}, \tag{10.24}$$

where C_t, C_h, C_s – aerodynamic coefficients of thrust, longitudinal and lateral forces correspondingly.

m_x, m_k, m_z – dimensionless coefficients of the roll, torque and pitch moments correspondingly;

$F_н = \pi R^2$ – MR disc area.

In some cases, such as problems of the helicopter dynamics the MR aerodynamic forces can be decomposed along the wind axes $O_{x_a y_a z_a}$ (Fig. 10.29, b). The component of the total aerodynamic force directed along the axis O_{y_a} is called rotor lifting force $Y_в$. Component $X_в$ directed along the axis O_{x_a}, is called the MR drag force or propulsive

force. Component along the axis $O_{x_a y_a}$ is the rotor transverse force $Z_в$. MR aerodynamic forces also can be recorded using the dimensionless aerodynamic coefficients

$$Y_в = C_{y_в} \frac{\rho(\omega R)^2}{2} \cdot F_{MR}, \quad (10.25)$$

$$X_в = C_{x_в} \frac{\rho(\omega R)^2}{2} \cdot F_{MR}, \quad (10.26)$$

$$Z_в = S = C_s \frac{\rho(\omega R)^2}{2} \cdot F_{MR}. \quad (10.27)$$

As it follows from Fig. 10.29, wind axes $O_{x_a y_a z_a}$ are turned by the MR angle-of-attack α_{MR} relatively to body axes (assuming the rotor has no slip angle). Then, by using the turning formulas we obtain

$$Y_r = T \cdot \cos\alpha_r - H \cdot \sin\alpha_r, \quad (10.28)$$

$$X_r = T \cdot \sin\alpha_r + H \cdot \cos\alpha_r, \quad (10.29)$$

Or without units of measurement:

$$C_{y_r} = C_t \cdot \cos\alpha_r - C_h \cdot \sin\alpha_r, \quad (10.30)$$

$$C_{x_r} = C_t \cdot \sin\alpha_r - C_h \cdot \cos\alpha_r, \quad (10.31)$$

Here C_{y_r} and C_{x_r} – coefficients of lift and drag accordingly.

Power on MR shaft, needed for implementation of given flight conditions is determined from the equation

$$N_{MR} = M_k \cdot \omega = m_k \frac{\rho(\omega R)^3}{2} \cdot F_{MR}. \quad (10.32)$$

This aerodynamic estimation method allows to calculate MR dimensionless aerodynamic coefficients of the second type, differing from the introduced above in the fact that they are also divided by the rotor solidity ratio

$$t = \frac{C_t}{\sigma}, \quad h = \frac{C_h}{\sigma}, \quad S = \frac{C_s}{\sigma}, \quad t_y = \frac{C_y}{\sigma}, \quad t_x = \frac{C_x}{\sigma}. \quad (10.33)$$

The dimensionless aerodynamic coefficients in this form are also widely used in the literature (e.g., in papers written by M.L. Mil, Vildgrube etc.).

10.1.3.3. Determination of Inductive Velocity Field

When calculating the rotor aerodynamic characteristics the vector of mean induced velocity v_i can be decomposed along the body axes to the components:

- axial ν_t (along axis O_y);
- longitudinal ν_r (along axis O_z);
- side ν_s (along axis O_z).

Components of inductive speed are directed opposite to positive direction of forces T, H, S.

Inductive speed is determined according to the pulse theorem

$$\bar{\nu}_t = \frac{C_t}{4B^2\bar{V}_\Sigma}, \tag{10.34}$$

where $\bar{\nu}_{T_0} = \nu_{T_0}/\omega R$ – dimensionless value of mean inductive velocity over MR disc;

B – end loss rate;

$\bar{V}_\Sigma = \sqrt{(\mu - \bar{\nu}_r)^2 + (\bar{V}\sin\alpha_r - \bar{\nu}_t)^2 + \bar{\nu}_s^2}$ – total flow relative velocity through CPR.

Based on the expression (10.34) it can be written according to the pulse theorem

$$\frac{\bar{\nu}_{T_0}}{C_t} = \frac{\bar{\nu}_r}{Cr} = \frac{\bar{\nu}_s}{C_s}. \tag{10.35}$$

Equations (10.34) - (10.35) for fixed values of aerodynamic coefficients C_t, C_h, C_s form a system of nonlinear equations solvable for the unknown inductive velocities using method of successive approximations. For the initial approximation it is taken

$$\bar{\nu}_{T_0} = \begin{cases} \dfrac{C_t}{4B^2\bar{V}}, \text{at } V \neq 0, \\ \operatorname{sign} C_t \cdot \dfrac{\sqrt{C_t}}{2B}. \end{cases} \tag{10.36}$$

The process is repeated until the specified accuracy is reached

$$\left| \frac{C_t}{4B^2\bar{V}_\Sigma^i} - \bar{\nu}_{T_0}^i \right| \le \varepsilon, \text{ where } \varepsilon = 10^{-6} \ldots 10^{-8}.$$

Longitudinal ν_r and lateral ν_s components of inductive speed are considered constant over MR disc. Change of axial component relies longitudinally skewed according to the second Glauert hypothesis and is determined by the expression

$$\bar{v}_T = \bar{v}_{T_0}[\sqrt{1.5\bar{r}}f_1(\mu) + (1 + K_\mu \cdot \bar{r} \cdot \cos\psi) \cdot f_2(\mu), \tag{10.37}$$

$$K_\mu = \frac{4}{3}\frac{\mu - \nu_r}{\mu - \nu_r - 1.2\lambda}, \tag{10.38}$$

where f1(μ) and f2(μ) – empirical coefficients, which take into account the uneven distribution of the inductive velocities depending on the rotor operating regime coefficient;

$\lambda = \overline{V} \cdot \sin\alpha - \bar{\nu}_{T_0}$ – inflow ratio;

ψ – blade azimuth position on MR disc.

Such representation of the inductive speed field, as it has been shown by studies, significantly improves the convergence process of the successive approximations and provides faster convergence of the results with the experimental data.

10.1.3.4. Algorithm for Calculating MR Aerodynamic Forces and Moments

If you know the inductive velocity distribution field over main rotor disc then you can determine the aerodynamic forces acting on each rotor blade and MR as a whole. When calculating the MR blade aerodynamic loads use the blade element theory. The blade element theory is based on the hypothesis of flat sections, which suggests that flow about each blade section has the parameters of the air flow (airfoil angle-of-attack, flow velocity, etc.), corresponding to the parameters of flow about wing of infinite span, i.e. airfoil. This approach allows to determine the aerodynamic forces acting on the blade element (section) and then to find the aerodynamic forces and moments of the blade by integration over the blade length (Fig. 10.31).

To determine the aerodynamic forces acting on the selected blade element dr, let's find components W_x and W_y of the resulting flow velocity r in an arbitrary cross-section and in certain azimuth position. For this purpose let's consider the velocity polygon at an arbitrary radius r (Fig. 10.32).

The components of the dimensionless velocities are found using relations

$$\overline{W}_x = \frac{W_x}{\omega R} = (\bar{r} - \bar{l}_{fl.h}) \cdot \cos\beta + \bar{l}_{fl.h} \cdot \cos\beta_0 + \\ + (\mu - \bar{\nu}_r) \cdot \sin\psi + \bar{\nu}_s \cdot \cos\psi; \tag{10.39}$$

$$\overline{W}_y = \frac{W_y}{\omega R} = (\overline{V} \cdot \sin\alpha_r - \bar{\nu}_T) \cdot \cos\beta - (\mu - \bar{\nu}_r) \cdot \cos\psi \cdot \sin\beta + \\ + \bar{\nu}_s \cdot \sin\psi \cdot \sin\beta - (\bar{r} - \bar{l}_{fl.h}) \cdot \frac{d\beta}{dt}. \tag{10.40}$$

It must be borne in mind that in simulation of the rotor with elastic blades being attached using stiff butt the offset of equivalent hinge $l_{fl.h}^{eq}$ may be greater than the radius of the root section r_0. Consequently, there is a blade portion $r < l_{fl.h}^{eq}$, where for equations (10.39) - (10.40) the sleeve setting angle β_0 is taken as angle β.

By the calculated values $\overline{W}_x$ and $\overline{W}_y$ the flow angle is determined

$$\Phi = \operatorname{arctg}\frac{\overline{W}_x}{\overline{W}_y} + \begin{cases} 0, \text{at}\, \overline{W}_x > 0 \\ \pi, \text{at}\, \overline{W}_x < 0. \end{cases} \tag{10.41}$$

Relative resultant velocity in section $\bar{r}$

$$\overline{W} = \sqrt{\overline{W}_x^2 + \overline{W}_y^2}\ . \tag{10.42}$$

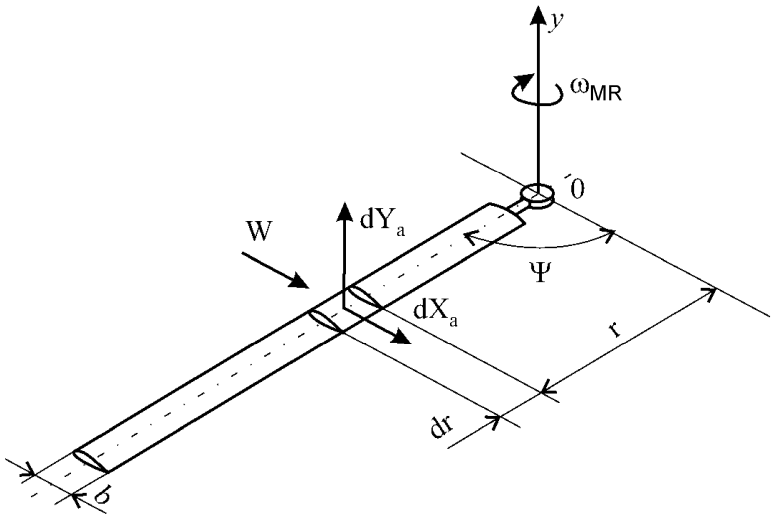


Fig. 10.31. Aerodynamic Forces Acting on MR Blade Element

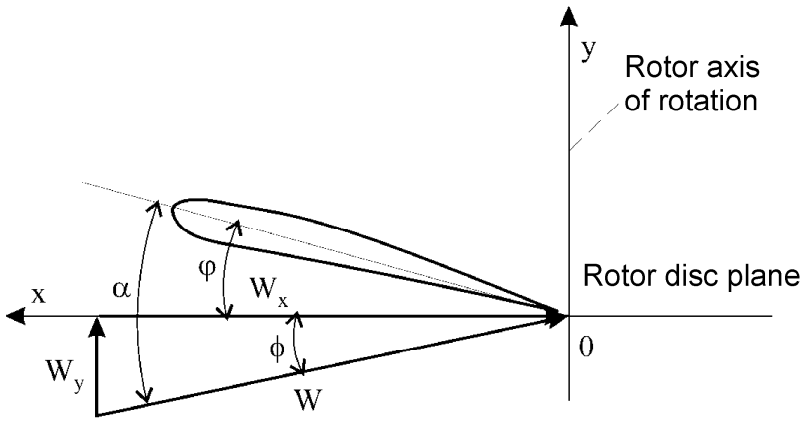


Fig. 10.32. Velocity Polygon in Blade Section

Knowing magnitudes of the resultant velocity $\overline{W}$ and inflow angle, let's find the angle-of-attack and Mach number in this section

$$\alpha = \varphi + \Phi\,, \tag{10.43}$$

$$M = M_R \cdot \overline{W} \tag{10.44}$$

where φ – section angle (Fig. 10.32);

M_R – blade tip Mach number.

Also the Reynolds number may be determined

$$\mathrm{Re} = \frac{W \cdot b}{\nu} = \frac{a}{\nu} b \cdot M . \tag{10.45}$$

Local angles-of-attack and Mach numbers M define lift coefficient C_{y_a} and profile drag C_{x_a}. For calculation of the rotor aerodynamic forces and moments let's move on to body axes (Fig. 10.33)

$$C_y = C_{y_a} \cdot \cos\Phi + C_{x_a} \cdot \sin\Phi , \tag{10.46}$$

$$C_x = C_{x_a} \cdot \cos\Phi - C_{y_a} \cdot \sin\Phi , \tag{10.47}$$

The trigonometric functions of inflow angle are found from the relation

$$\cos\Phi = \frac{\overline{W}_x}{\overline{W}}; \qquad \sin\Phi = \frac{\overline{W}_y}{\overline{W}} . \tag{10.48}$$

Distributed (per unit length) values of the elementary thrust and rotation drag are found from the expressions

$$dT = dT_1 \cdot \cos\beta = C_y \frac{\rho W^2}{2} b \cdot dr \cdot \cos\beta , \tag{10.49}$$

$$dQ = C_x \frac{\rho W^2}{2} b \cdot dr . \tag{10.50}$$

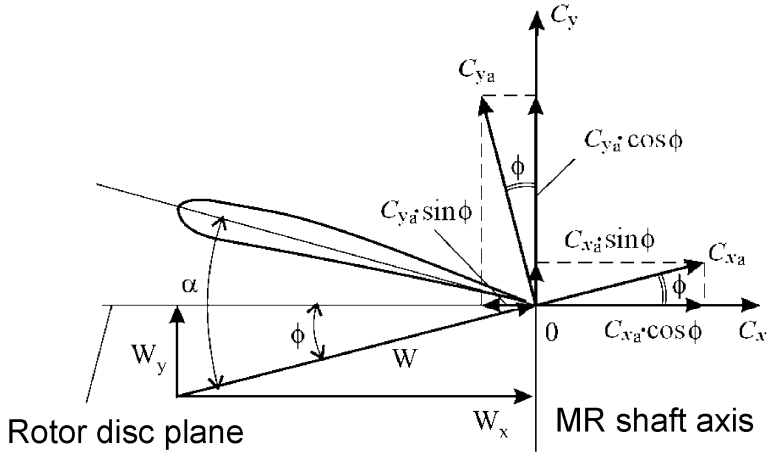


Fig. 10.33. To Definition of Blade Element Thrust and Rotation Drag

Having divided the right parts of equations (10.49) - (10.50) by $\rho(\omega R)^2 \cdot F_H / 2$ we shall obtain the dimensionless aerodynamic force coefficients of the dedicated blade element

$$dC_{T_{bl}} = C_y \frac{\overline{W}^2 \cdot \bar{b} \cdot d\bar{r}}{\pi} \cdot \cos\beta , \tag{10.51}$$

$$dC_{Q_{bl}} = C_x \frac{\overline{W}^2 \cdot \bar{b} \cdot d\bar{r}}{\pi}. \qquad (10.52)$$

Elementary coefficients of the second kind are calculated by multiplying equations (10.51) - (10.52) by the number of blades and by dividing by the rotor solidity ratio:

$$dt = C_y \cdot \overline{W}^2 \frac{\bar{b}}{\bar{b}_7} \cdot \cos\beta \cdot d\bar{r}, \qquad (10.53)$$

$$dq = C_x \cdot \overline{W}^2 \frac{\bar{b}}{\bar{b}_7} \cdot d\bar{r}. \qquad (10.54)$$

Elementary coefficients of the longitudinal and lateral forces are found as the sum of the projections on the axes Ox and Oz

$$dh = -dt \cdot \cos\psi \cdot \sin\beta + dq \cdot \sin\psi, \qquad (10.55)$$

$$ds = -dt \cdot \sin\psi \cdot \sin\beta - dq \cdot \cos\psi. \qquad (10.56)$$

To determine the thrust moment elementary coefficients relatively to the hub center the following conditions must be met:

- If $r > l_{fl.h.}$ apply this force to the hinge $dm_T = dt \cdot l_{fl.h} \cdot \cos\beta$;

- If $r < l_{fl.h.}$ then $dm_T = dt \cdot r$.

To get the full moment with respect to the rotor center add to the air component of the aerodynamic forces the moments generated by centrifugal forces of rotation and flapping, inertia forces of flapping, blade weight, etc.

$$\begin{aligned} m_\psi = \int_{r_0}^{1} dm_T - \frac{2}{\pi R^5 \rho}\Big\{ & l_{fl.h}(S_{fl.h} - m_{bl} \cdot l_{fl.h}) \cdot \Big[\cos\beta \cdot \sin\beta_0 + \\ & + \frac{d^2\beta}{d\psi^2} \cdot \cos(\beta - \beta_0) - \left(\frac{d\beta}{d\psi}\right)^2 \cdot \sin(\beta - \beta_0)\Big] + \\ & + (m_{bl} \cdot l_{fl.h}^2 + J_0) \cdot \sin\beta_0 \cdot \cos\beta_0 + \\ & + \Big[g(m_{bl} \cdot l_{fl.h} - S_0) \cdot \cos\beta_0 - M_{fl.h}^{\beta}(\beta - \beta_0)\Big] \cdot \frac{1}{\omega^2}\Big\}. \end{aligned} \qquad (10.57)$$

Vector m_ψ lies in CPR and is perpendicular to the blade longitudinal axis. Elementary pitch and roll moments at this azimuth are determined by the relations

$$m_{z\psi} = -m_\psi \cdot \cos\psi, \quad m_{x\psi} = -m_\psi \cdot \sin\psi. \qquad (10.58)$$

Properly the elementary force dQ_{Bl} also creates a moment, however, the amount of force dQ is much less than dT, and the arm does not exceed $l_{fl.h} \cdot \sin\beta$, so the moments of the forces caused by rotation drag are not taken into account.

Elementary moment coefficient of rotation drag (torque) in this section

$$d\left(m_k/\sigma\right) = dq\left[\left(\bar{r} - \bar{l}_{fl.h}\right)\cos\beta + \bar{l}_{fl.h}\cos\beta_0\right]. \quad (10.59)$$

Considering expressions (10.46), (10.47), (10.51), (10.52) coefficient dq can be expressed as

$$dq = C_x \frac{\overline{W}^3 \bar{b}}{\overline{W}_x \bar{b}_7} d\bar{r} - dt \frac{\overline{W}_y}{\overline{W}_x}. \quad (10.60)$$

From formula (10.39) we can determine the arm $(\bar{r} - \bar{l}_{fl.h})$

$$(\bar{r} - \bar{l}_{fl.h}) \cdot \cos\beta + \bar{l}_{fl.h} \cdot \cos\beta_0 - \overline{W}_x - (\mu - \bar{\nu}_r) \cdot \sin\psi - \bar{\nu}_s \cdot \cos\psi. \quad (10.61)$$

Substituting expressions (10.60) and (10.61) in (10.59) after a series of transformations, we obtain the following expression:

$$d\left(\frac{mk}{\sigma}\right) = C_{x_r} \overline{W}^3 \bar{b} d\bar{r} + dt\bar{\nu}_t + dh\bar{\nu}_r + ds\bar{\nu}_s - dt_x \overline{V} + dt(\bar{r} - \bar{l}_{fl.h}) \frac{d\beta}{d\psi}. \quad (10.62)$$

In this expression, the first term defines:

- profile losses

$$dm_p = C_x \cdot \overline{W}^3 \frac{\bar{b}}{\bar{b}_7} \cdot d\bar{r} \cdot \sigma;$$

- the following three components are inductive losses

$$dm_i = dC_T \cdot \bar{\nu}_T + dC_r \cdot \bar{\nu}_r + dC_s \cdot \bar{\nu}_s;$$

- the fifth member is propulsive losses

$$dm_{prop} = dC_{x_r} \cdot \overline{V};$$

- the last member means losses for flapping

$$dm_\beta = \sigma \cdot dt(\bar{r} - \bar{l}_{fl.h}) \cdot \frac{d\beta}{d\psi}.$$

Integral force and moment coefficients on the rotor are found by expressions

$$t = \frac{1}{2\pi}\int_0^{2\pi} d\psi \int_{\bar{r}_0}^{1} dt,$$

$$h = \frac{1}{2\pi}\int_0^{2\pi} d\psi \int_{\bar{r}_0}^{1} dh, \qquad (10.63)$$

$$S = \frac{1}{2\pi}\int_0^{2\pi} d\psi \int_{\bar{r}_0}^{1} ds,$$

$$m_k = \frac{1}{2\pi}\int_0^{2\pi} d\psi \int_{\bar{r}_0}^{1} dm_k = \frac{\sigma}{2\pi}\int_0^{2\pi} d\psi \int_{\bar{r}_0}^{1} dq\left[(\bar{r} - \bar{l}_{fl.h})\cdot\cos\beta + \bar{l}_{fl.h}\cdot\cos\beta_0\right].$$

10.1.3.5. Calculation of Blade Flapping Motion

As follows from the expressions (10.39) - (10.40) for calculating a relative air flow speed components W_x, W_y in this section we need to know flapping angle and its rate of change $d\beta/dt$ calculated at every azimuth. These parameters, as well as the flapping acceleration $d^2\beta/dt^2$ are needed to calculate the MR aerodynamic forces and moments.

These characteristics of the blade flapping are determined using the blade balance equations with respect to the flapping hinge, called the equation of the blade flapping motion.

Equation of blade flapping can be obtained based on the equation of blade balance relative to the flapping hinge. Forces acting on the blade element dr in the plane of flapping are shown in Fig. 10.34. This blade element is under action of the following forces:

- blade thrust dT;
- centrifugal force $dN = dm\cdot\omega_r^2\cdot r$;
- blade flapping inertia force $dp_{ir} = -dm\cdot r\cdot\ddot{\beta}$ ($\ddot{\beta} = d^2\beta/dt^2$ - blade flapping angular acceleration);
- weight of the selected blade element $dG_{bl} = g dm_{bl}$.

Since any hinge receives and transmits only forces, but cannot transmit moments, the sum of the moments of all forces about the axis of flapping hinge must be zero:

$$M_{fl.h} = \int_{l_{ш}}^{R} (dT(r - l_{fl.h}) - dN(r - l_{fl.h})\cdot\sin\beta -$$
$$-dP_{ir}(r - l_{fl.h}) - dG(r - l_{fl.h})\cdot\cos\beta = 0. \qquad (10.64)$$

Integrating this expression and taking into account expression (10.9) for the inertial-mass characteristics of the blades, we obtain the equation of the blade flapping in the integral form

$$\int_{l_{fl.h}}^{R} dT_{bl}(r - l_{fl.h}) - g \cdot \cos\beta(S_{fl.h} - m_{bl} \cdot l_{fl.h}) - M_{ctl} =$$

$$= (J_{fl.h} - 2S_{fl.h} + m_{bl} \cdot l_{fl.h}^2) \cdot \omega^2 \frac{d^2\beta}{dt^2} + \left[J_{fl.h} \cdot \cos\beta - \right.$$

$$\left. -S_{fl.h} \cdot l_{fl.h}(2\cos\beta - \cos\beta_0) - m_{bl} \cdot l_{fl.h}^2(\cos\beta_0 - \cos\beta)\right] \cdot \omega^2 \cdot \sin\beta. \quad (10.65)$$

The terms on the left side of the equation are the moments of thrust forces, the blade weight and the elastic forces $M_{el} = -M_{el}^{\beta}(\beta - \beta_0)$ about the axis of flapping hinge. The terms on the right side of the equation are moments caused by the flapping inertia forces and centrifugal forces with respect to flapping hinge.

In this formulation of the problem equation (10.65) can be solved only numerically as the blade aerodynamic characteristics depend on its flapping motion, and blade flapping motion can be described if the blade aerodynamic forces are determined. In other words, the problem of calculating the blade aerodynamic characteristics considering its flapping is nonlinear. The developed program implements computational process of integrating the flapping equation based on Euler's method with recalculation. This method is described in detail in the book “Helicopters. Calculation and Design” written by M.L. Mil. Also other numerical methods can be used for integration of the flapping equation.

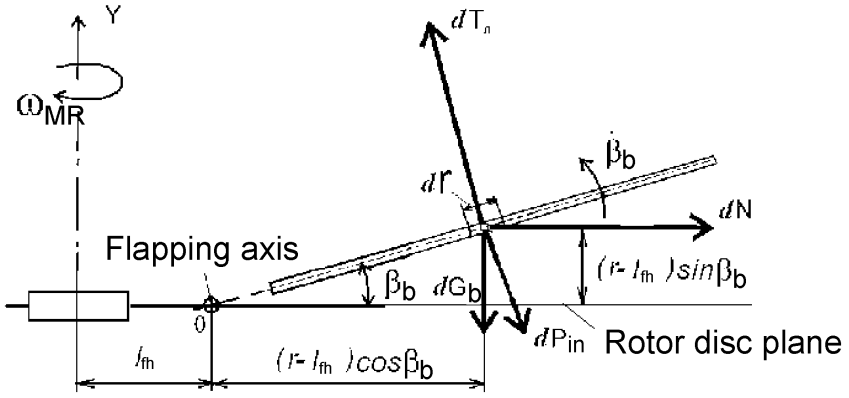


Fig. 10.34 Forces Acting on Blade Element in Plane of Flapping

After obtaining the sustainable blade flapping motion the harmonic analysis of flapping angle depending on the azimuth is carried out. This dependence is approximated by a trigonometric sum in the form

$$\beta = a_0 - a_1 \cdot \cos\psi - b_1 \cdot \sin\psi - a_2 \cdot \cos 2\psi - b_2 \cdot \sin 2\psi \,. \qquad (10.66)$$

The expansion factor is determined by the formulas:

$$a_0 = \frac{1}{2\pi} \int_0^{2\pi} \beta \cdot d\psi \,,$$

$$a_k = -\frac{1}{\pi} \int_0^{2\pi} \beta \cdot \cos(k\psi) \cdot d\psi \,, \qquad (10.67)$$

$$b_k = -\frac{1}{\pi} \int_0^{2\pi} \beta \cdot \sin(k\psi) \cdot d\psi \,,$$

where k – a harmonic number.

Methodological studies have shown that satisfactory estimation accuracy of MR aerodynamic characteristics is provided with integration step in azimuth $|\Delta\varphi| \le 12 \ldots 15^0$.

10.1.3.6. Validation of Results Accuracy. Capabilities of Developed Numerical Method

The developed method of calculating MR aerodynamic characteristics allows to access and analyze large amounts of information concerning calculated total and distributed rotor parameters. In developing the work program a large volume of methodological research was done, which has greatly improved the accuracy of the results.

Figs 10.35 - 10.37 show the results of test problems aimed to determine total distribution of MR aerodynamic characteristics of real helicopters. Taking into account the assumptions and simplifications, this technique enables determination of a wide range of MR aerodynamic characteristics with sufficient accuracy. Features of the developed calculation methods are illustrated with the calculated data shown in Figs 10.38 - 10.41.

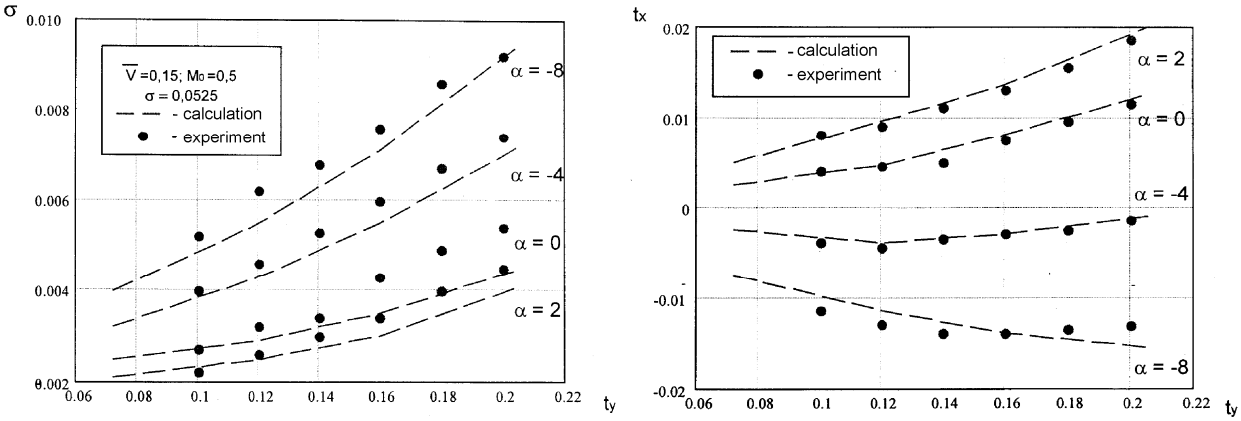


Fig. 10.35. Polar Curves of Mi-4 Helicopter Main Rotor in Level Flight

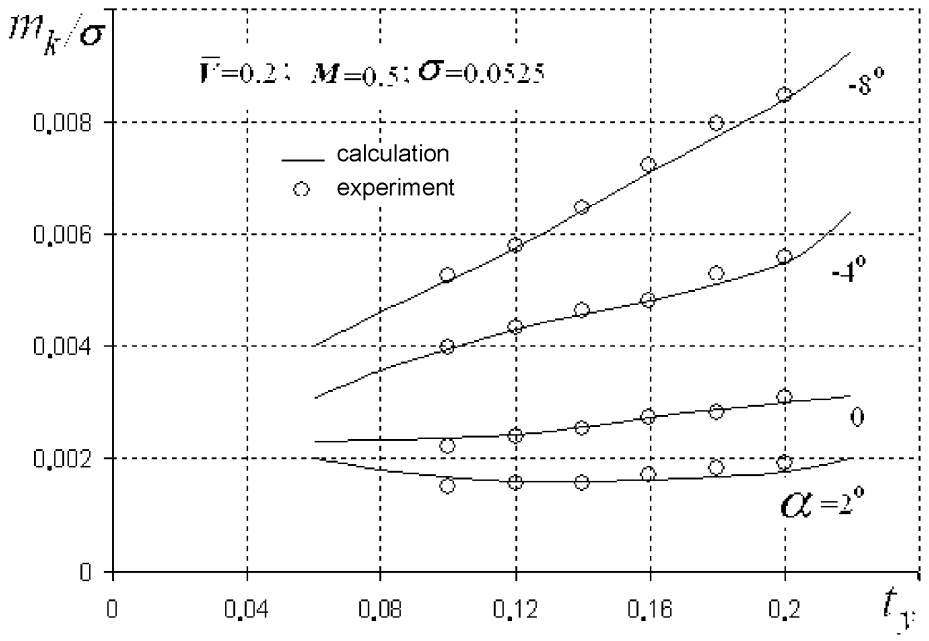


Fig. 10.36. Polar Curves of Mi-4 Helicopter Main Rotor in Forward Flight

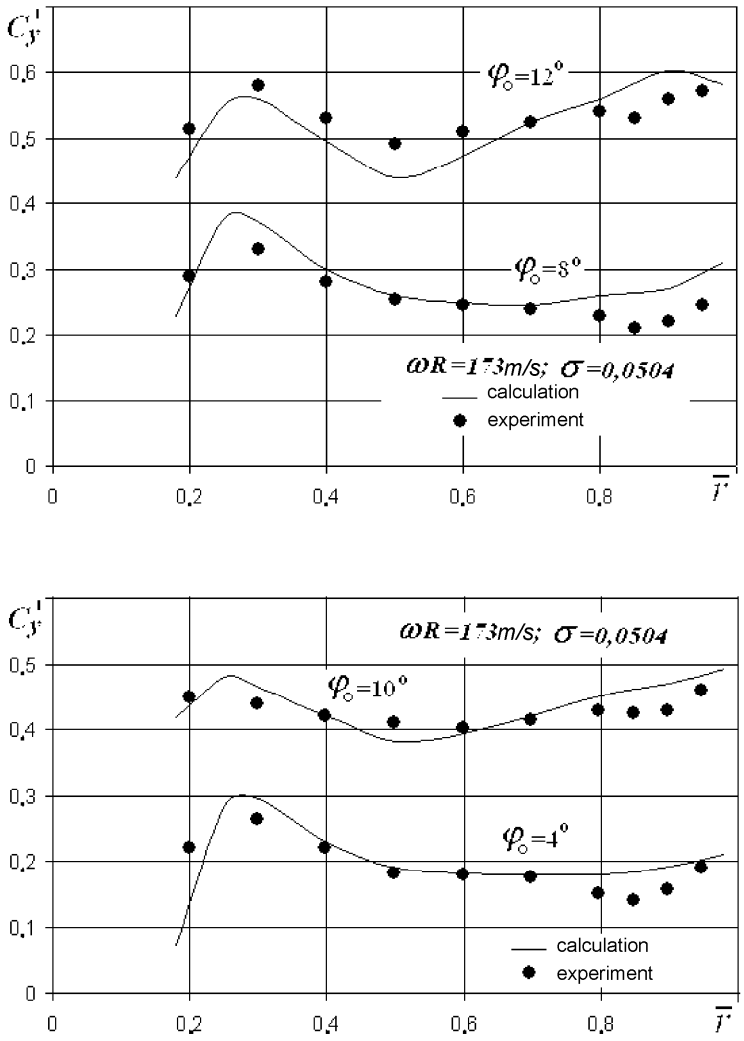


Fig. 10.37. Distribution of Lift Coefficient for Main Rotor Blade Cross-Sections

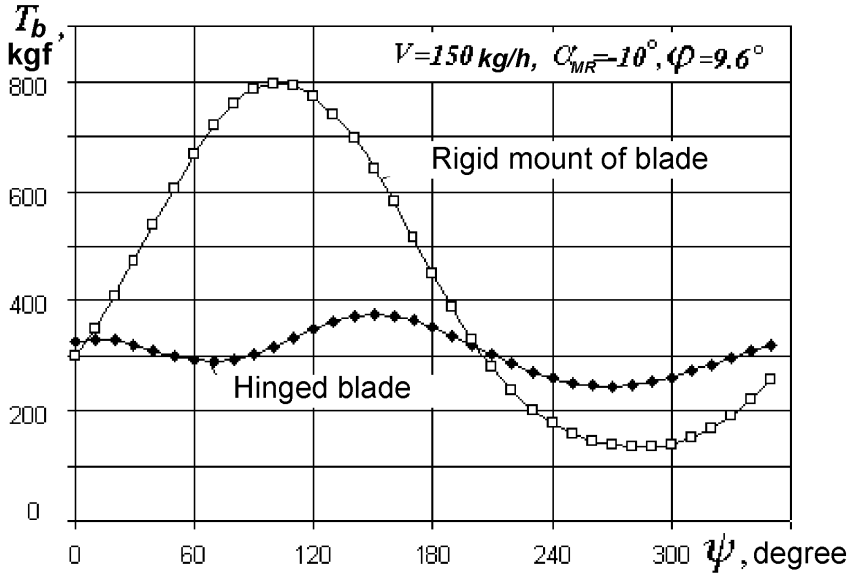


Fig. 10.38. Changing Blade Thrust Depending on Its Attachment to Hub

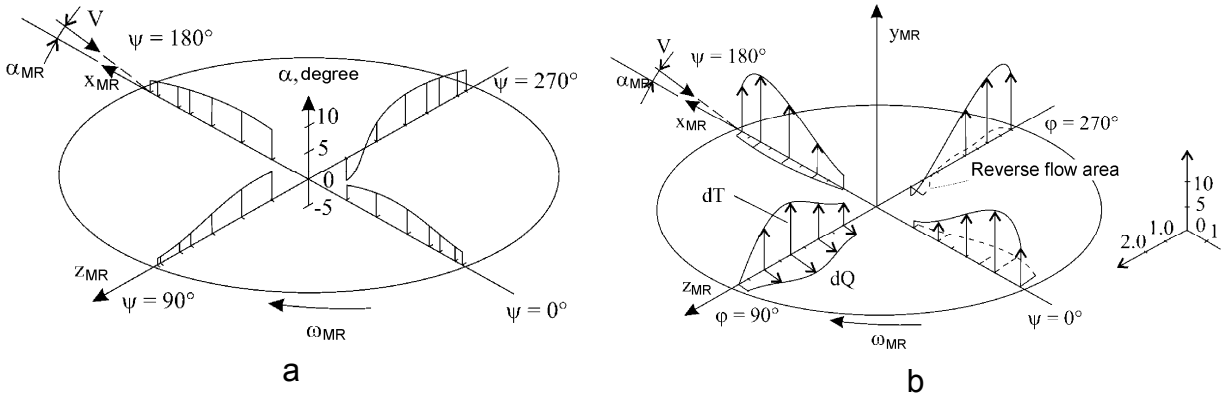


Fig. 10.39. Influence of Blade Azimuth Position on Aerodynamic Load Distribution Radiuswise: a – distribution of angle-of-attack over MR disc; b – distribution of blade aerodynamic load depending on its azimuth

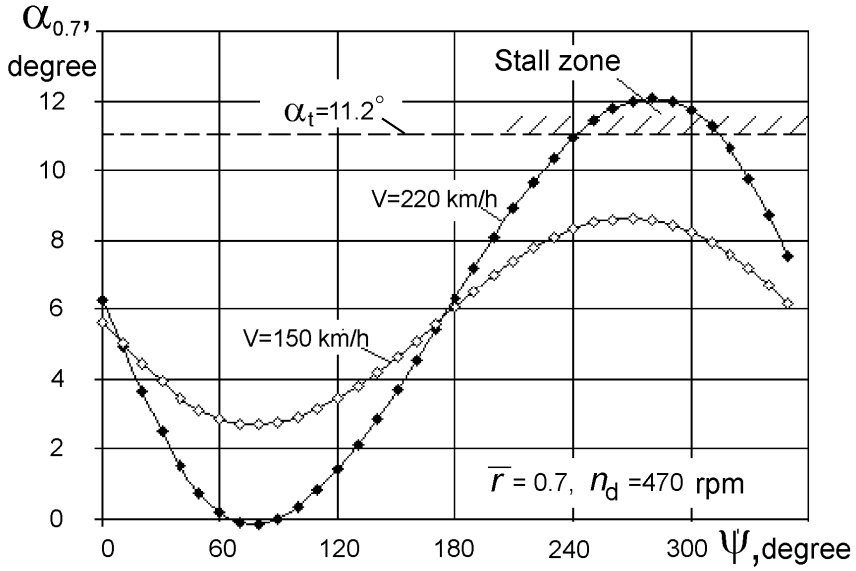


Fig. 10.40. Variation of Angle-of-attack In Blade Typical Section

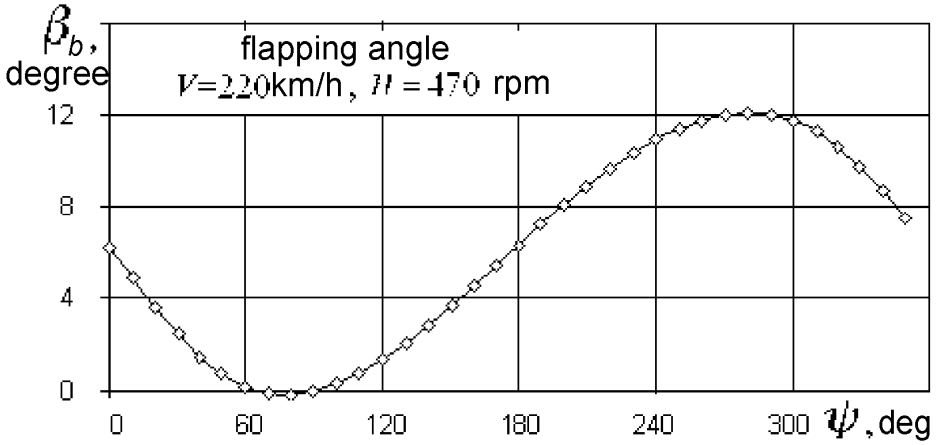


Fig. 10.41. Variation of Blade Flapping Angle During Main Rotor Single Revolution

10.1.4. Further Development and Improvement of Methods for Estimation of Rotor Aerodynamic Characteristics

Large amount of information obtained in estimation of MR aerodynamic characteristics, such as distribution of angles-of-attack of the blade sections, parameters of blade flapping motion, etc., has allowed to solve a number of practical problems on the basis of this methodology.

10.1.4.1. Determination of Total and Distributed Loads Acting on MR Blade and Hub

In general, a blade element dr, located at radius r from the axis of rotation, under oblique flow is under action of aerodynamic and weight (inertia, Coriolis, gravity) forces. The sum of the projections of all the elementary forces in the coordinate system axes Oxyz (Fig. 10.42), rigidly connected to the blade, can be written as

$$dP_x = -dQ + dml_{d.h}\omega^2 \sin\xi + 2dm(r - l_{fl.h})\omega^2 \frac{d\beta}{d\psi} \cdot \sin\beta -$$

$$-2dm(r - l_{fl.h})\cos\beta\omega\frac{d\beta}{d\psi} \quad (10.65)\psi - \omega_z \sin\psi); \tag{10.68}$$

$$dP_y - dT - dm(r - l_{fl.h})\omega^2 \frac{d^2\beta}{d\psi^2} - dmr\omega^2 \sin\beta - gdm\cos\beta +$$

$$+2dm[(r - l_{fl.h})\cos\beta + l_{fl.h}](10.65)\omega_x \cos\psi - \omega_z \sin\psi); \tag{10.69}$$

$$dP_y - dT - dm(r - l_{dr.h})\omega^2 \frac{d^2\beta}{d\psi^2} - dmr\omega^2 \sin\beta - gdm\cos\beta +$$

$$+2dm[(r - l_{fl.h})\cos\beta + l_{fl.h}]\omega\cos\beta(\omega_x \cos\psi - \omega_z \sin\psi). \tag{10.70}$$

The aerodynamic loads in formulas (10.68) - (10.70) define the result of the rotor aerodynamic calculation. As a result of the calculation the derivatives $d\beta/d\psi$ and $d^2\beta/d\psi^2$ were also obtained, which define the nature of the blade flapping motion. Having given the law of weight distribution per unit length blade radiuswise (Fig.10.43), we can calculate the components of the inertial forces, presenting in the expression (10.68) - (10.70).

Periodic blade motion in the rotor plane of rotation relative to the axis of flapping hinge, resulting from changes in forces, is characterized by an average value of the swing angle. The value of the average swing angle can be determined from the condition of zero moment created by all the forces about drag hinge:

$$M_{dr.h} = \int_{l_{dr.h}}^{R} dP_x (r - l_{dr.h}) = 0 . \qquad (10.71)$$

Having solved the system of equations (10.68) and (10.71), we determine the average value of the blade swing angle. Fig. 10.44 shows an example of variation of the Ka-26 helicopter MR blade swing angle under various flight conditions.

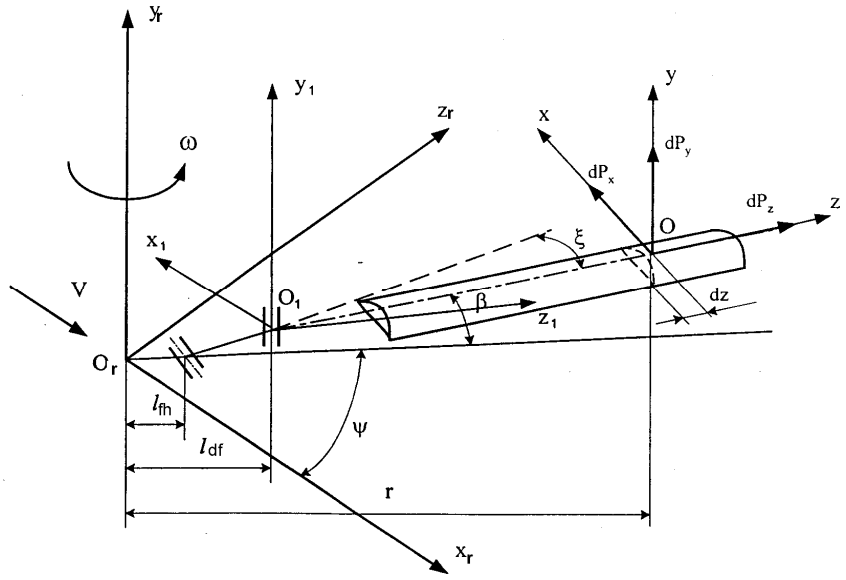


Fig. 10.42. Blade Design Scheme

Having integrated the expression for the components of the elementary forces with respect to the radius r, we can find total variable loads P_x, P_y, P_z in the blade coordinate system. As an illustration, Figs 10.45 and 10.46 show variation in load distribution radiuswise the MR blade for different azimuths as well as total force in the plane of rotation and in plane of flapping. In Fig. 10.46 the dashed line shows the calculation according to the simplified procedure developed by Volobuev.

Total loads P_{x1}, P_{y1}, P_{z1} in a coordinate system associated with the hub center, could be found from the following expressions:

$$P_{x_1} = P_x \cos\xi + P_y \sin\beta \sin\xi - P_z \cos\beta \sin\xi, \quad (10.72)$$

$$P_{y_1} = P_{y_1} \cos\beta + P_z \sin\beta, \quad (10.73)$$

$$P_{z_1} = P_x \sin\xi - P_y \sin\beta \cos\xi + P_z \cos\beta \cos\xi. \quad (10.74)$$

The total expressions for the moments at the MR rotor hub caused by one blade are determined by the formulas

$$M_{x_в} = [P_{x_1}(l_{dr.h} - l_{fl.h}) \cdot \sin\beta + P_y(l_{dr.h} - l_{fl.h}) \cdot \sin\xi] \cdot \cos\psi - \\ -P_{y_1} \cdot l_{fl.h} \cdot \sin\psi, \quad (10.75)$$

$$M_{z_в} = \left[P_{x_1}(l_{dr.h} - l_{fl.h}) \cdot \sin\beta + P_y(l_{dr.h} - l_{fl.h}) \cdot \sin\xi\right] \cdot \sin\psi - \\ -P_{y_1} \cdot l_{dr.h} \cdot \cos\psi. \quad (10.76)$$

Thus, we determined the forces and moments acting on the MR hub, which are the initial data for calculation of strength of MR shaft and other structural elements of the helicopter.

MR shaft, in addition to its main function – torque transmission – receives the other components of forces and moments acting on the hub (Fig 10.47). In the shaft section perpendicular to its axis normal stresses also act in addition to the shear stresses created by torque. The variable component of the normal stress is mainly defined by forces P_{x1}, P_{y1}, P_{z1} and moments $M_{xв}$, $M_{zв}$ in the hub plane. These loads cause the variable bending stress in rotating shaft, changing with frequency equal to the MR number of revolutions.

Distribution of load per unit length radiuswise the blade in the rotation and flapping planes makes it possible to calculate also variable stress components in the blade load-bearing members. Fig. 10.48 shows total amplitudes of the alternating stress in the blade spar, obtained by calculation, and data of the flight experiment (strain measurement).

Thus, the proposed method of MR aerodynamic estimation allows not only to determine the performance characteristics of the helicopter, but also to solve a number of practical problems related to the calculation of strength of the transmission elements, strength and durability of the torsion bar, variable loads in the blade load-bearing elements.

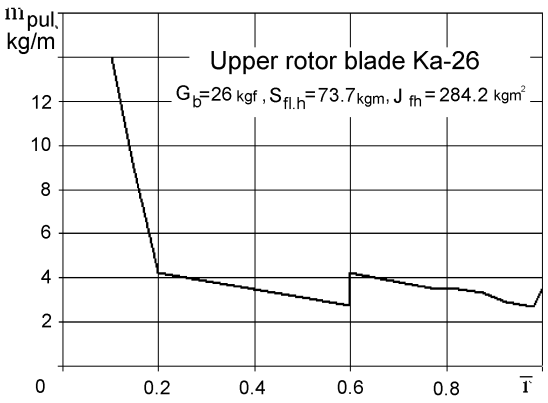


Fig. 10.43. Distribution of Load per Unit Length Spanwise Ka-26 MR Blade

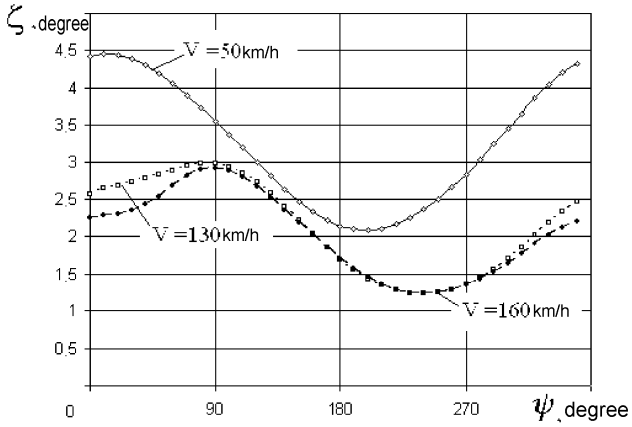
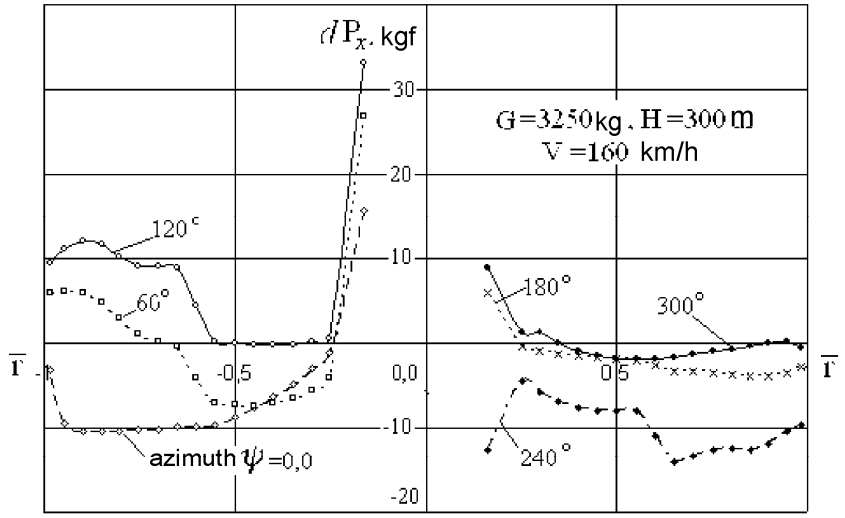
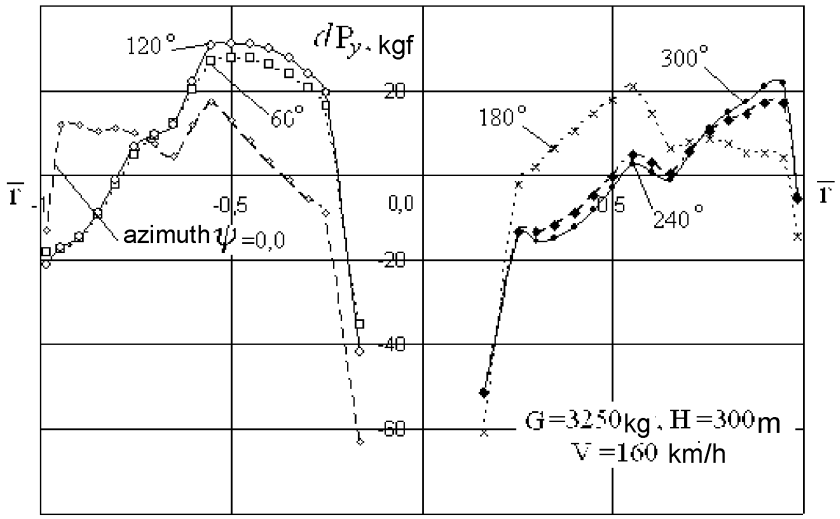


Fig. 10.44. Change of Ka-26 Blade Swing Angle Depending on Flight Regime

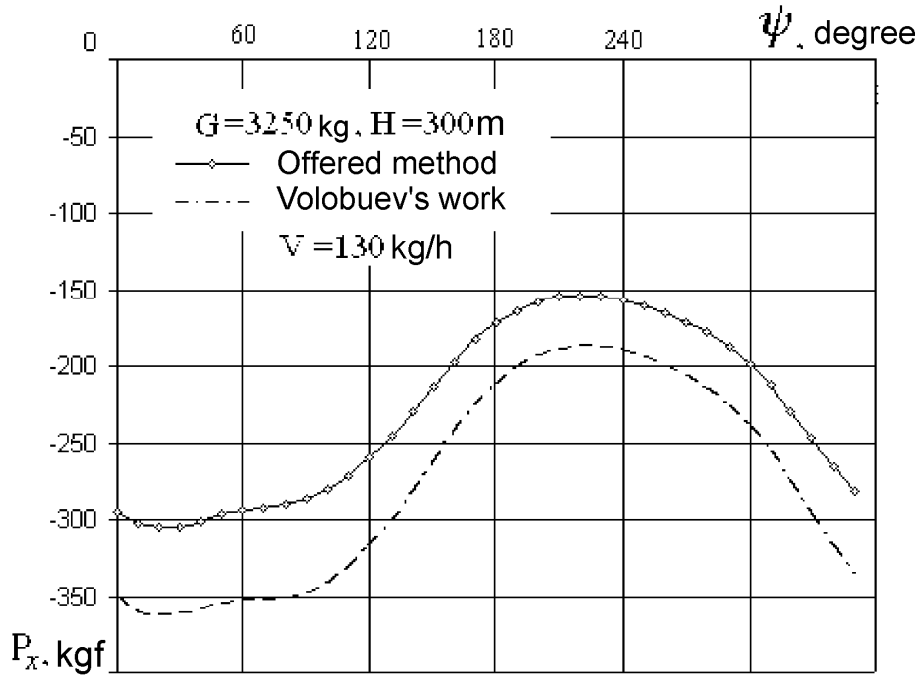


a

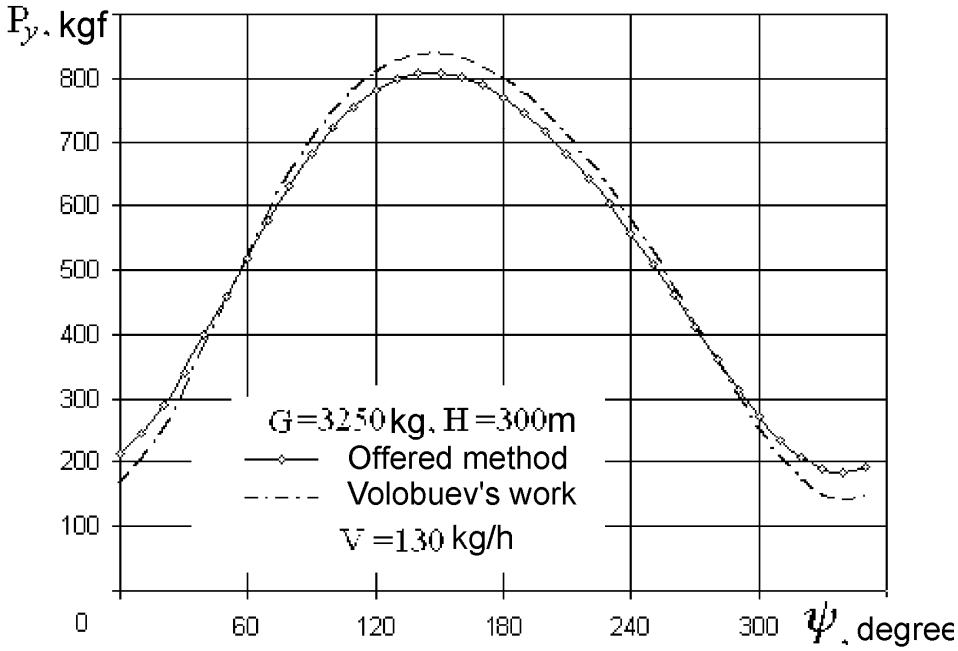


b

Fig. 10.45. Variation of Blade Load per Unit Length at Its Various Azimuth Positions: a – in plane of rotation; b – in plane of flapping



a



b

Fig. 10.46. Total Loads on MR Blade:
(a) in plane of rotation
(b) in plane of flapping

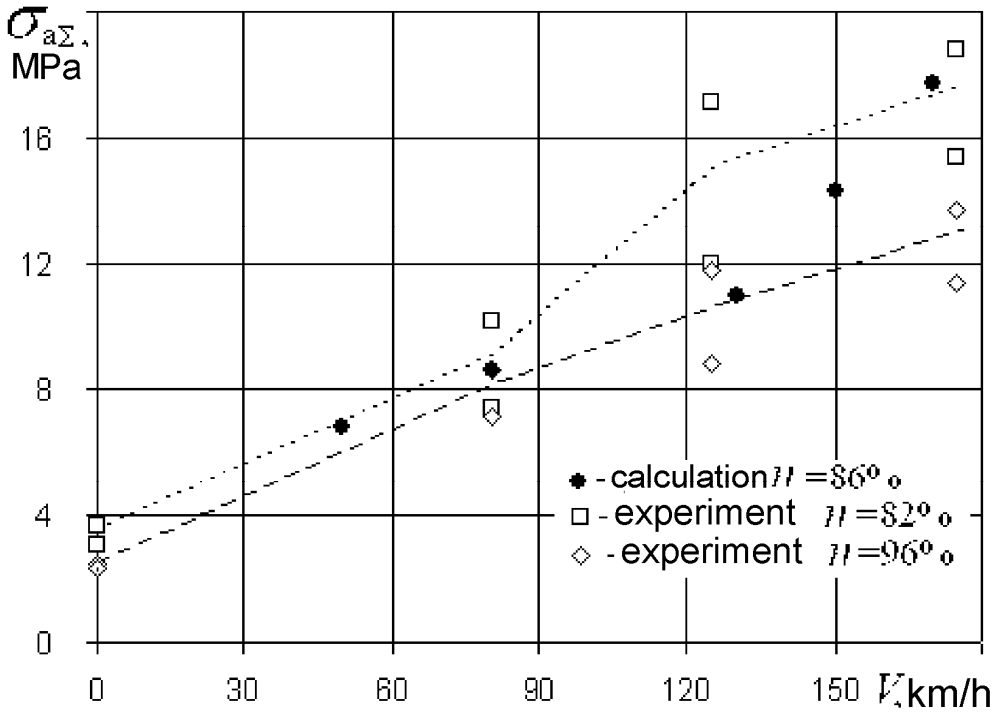


Fig. 10.47. Comparison of Calculated and Experimental Amplitudes of Variable Stresses on Ka-26 MR Blades (r=0.5)

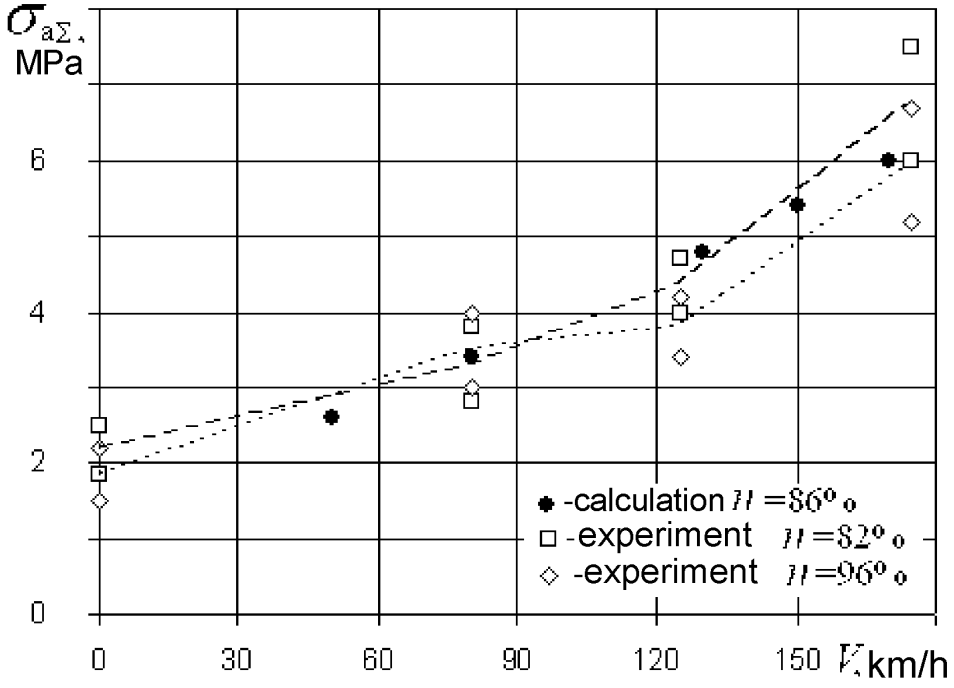


Fig. 10.48. Comparison of Calculated and Experimental Amplitudes of Variable Stresses on Ka-26 MR Blades (r=0.75)

10.1.5. Calculation of Helicopter Basic Flight and Technical Characteristics Based on Modified Mil – Yaroshenko Method

At the stage of defining tactical and technical requirements for the projected helicopter and performing preliminary draft, when laying the basic design solutions, defining geometric and weight parameters of the helicopter, its components and systems, it is advisable to find an area of geometric and kinematic parameters of the main and tail rotor, which provides desired flight characteristics of the helicopter. In carrying out these studies the statistical data on this type of helicopters and modern mathematical methods and models should be used as that make it possible to predict the flight characteristics of the designed helicopter with a minimum amount of the initial information.

In the developed system of the helicopter aerodynamic estimation, which block diagram is shown in Fig. 10.1, the special place is occupied by the "Calculation of helicopter performance using the 1st level program" subsystem (block). When calculating the helicopter basic performance characteristics in the first approximation in the subsystem the helicopter aerodynamics is estimated using the Mil–Yaroshenko method. In contrast to the Mil–Yaroshenko method the graphic-analytical method described in the "Helicopters. Design and Engineering" monograph under the editorship of M.L. Mil, the developed numerical method allows to solve the problem of aerodynamic calculation of the designed helicopter simplified arrangement based on the Glauert–Locke pulse theory equations.

10.1.5.1. Formulation of Problem

Let's consider the helicopter steady straight flight with small angles of trajectory slope. We know geometric and kinematic characteristics of main and tail rotors, their relative positions, as well as the profiling of blades (type and series of blade airfoils, its aerodynamic characteristics in the form of dependencies $C_{y_a}(\alpha), C_{x_a}(\alpha)$).

For simplicity in output of basic formulas, let's consider the helicopter steady horizontal flight. We also believe that the helicopter is balanced, that means all the moments acting on the helicopter are equal zero $M_x = M_y = M_z = M_\Sigma = 0$. Equations of helicopter motion in this case (Fig. 10.49) have the form:

$$\vec{R} \cdot \sin\delta + \vec{Q}_{pd} = 0, \tag{10.77}$$

$$\vec{R} \cdot \cos\delta - \vec{G} = 0. \tag{10.78}$$

The system of equations (10.77) – (10.78) is written in a high-speed coordinate system $Ox_a y_a$, associated with the main rotor. In formulas (10.77) – (10.78):

R – MR total aerodynamic force;

Q_{pd} – helicopter airframe parasitic drag;

G – helicopter weight;

δ – angle between the total aerodynamic force and the normal to the trajectory.

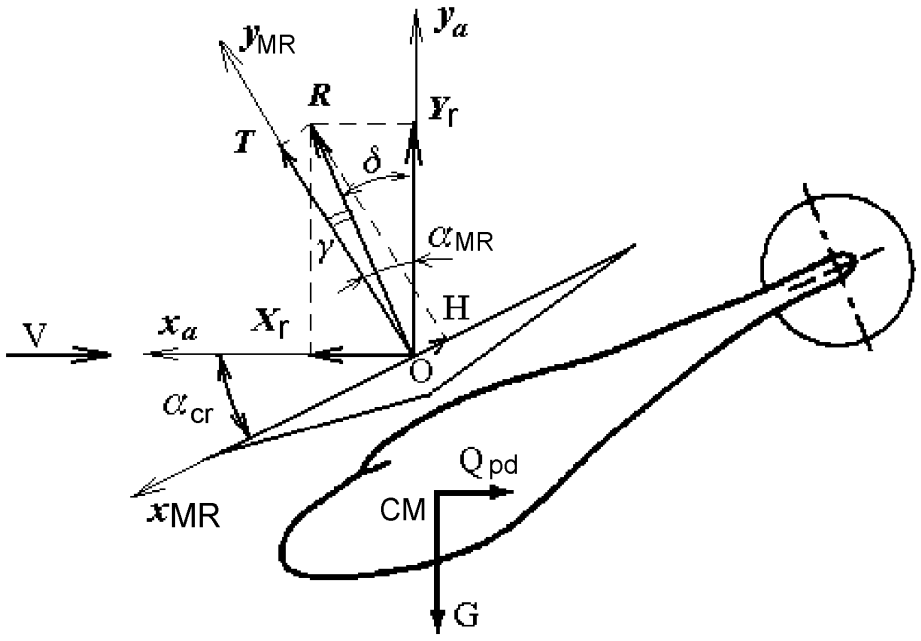


Fig. 10.49. Driving Forces Acting on Helicopter in Steady Level Flight

To the system of equations (10.77) - (10.78) add an equation that expresses the equality of power on MR shaft and available capacity of the designed helicopter power plant

$$N_{MR} = \xi \cdot N_e, \tag{10.79}$$

where ξ – power utilization rate of the power plant;

N_e – total available power of the helicopter power plant (effective power).

Power utilization rate at the stage of preliminary design is given as a function of the relative flight speed $\xi(\overline{V})$ on the basis of statistical data or simplified calculation of power losses in transmission, for driving other units, etc. The example is shown in Fig. 10.50.

The angle between the direction of resultant force R and the normal to the trajectory may be determined from the relationship

$$\delta = \alpha_r + \gamma = \alpha_r + \operatorname{arctg}\frac{H}{T}. \qquad (10.80)$$

In horizontal flight the ratio $H/T<<1$, so the angle $\gamma \approx 0$ that is it can be assumed that in level flight the condition $T=R$ is true (full aerodynamic force vector is directed along the axis of the rotor shaft). Given this assumption, the system of helicopter motion equations takes the following form:

$$\left.\begin{aligned} T \cdot \sin\delta - Q_{pd} &= 0, \\ T \cdot \cos\delta - G &= 0, \\ N_{MR} &= \xi \cdot N_e \end{aligned}\right\}. \qquad (10.81)$$

From the first equation it follows that change in flight speed can be achieved only by changing the angle-of-attack and MR thrust. At given frequency of MR rotation the thrust projection on axis Oy_a (rotor lift) balances the helicopter weight. To maintain constant altitude and flight speed you must change the value of MR collective pitch and power applied to the rotor.

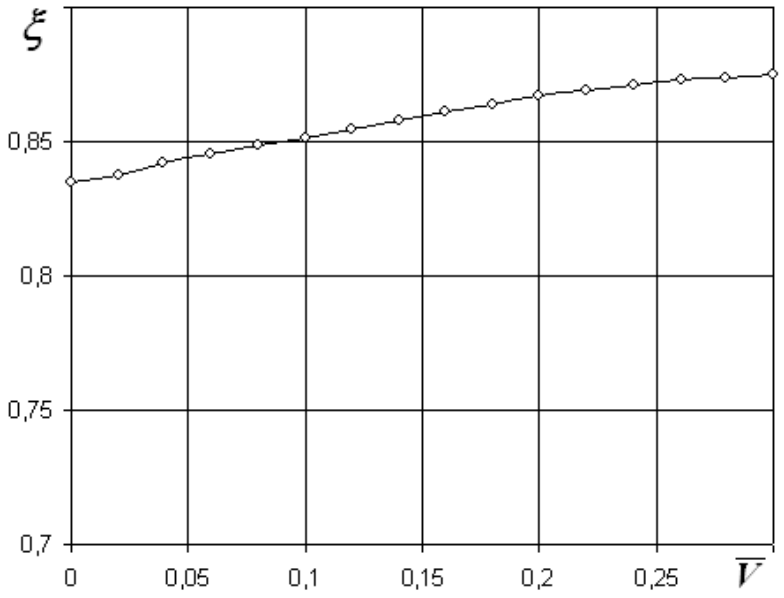


Fig. 10.50. Approximate Function of Power Plant Power Utilization Rate vs Relative Velocity

Transform the system of equations (10.81) to the form:

$$t \cdot \sin\delta + \bar{C}x_{pd}\frac{\bar{V}^2}{\sigma_r} = 0; \qquad (10.82)$$

$$t = \frac{2G}{\rho(\omega R)_r^2 \sigma_r F_r}; \tag{10.83}$$

$$m_k = \frac{2\xi \cdot N_e}{\rho(\omega R)_r^3 \sigma_r F_r}. \tag{10.84}$$

Here $\bar{C}x_{pd} = Cx_{pd} / F_r$ – helicopter airframe parasitic drag coefficient reduced to MR disc area;

$\bar{V} = V / \omega R$ – relative flight speed;

$\sigma_н = \frac{Kbl \cdot b_{0.7}}{F_r}$ – MR solidity ratio;

$t = C_t / \sigma_r$ – MR traction coefficient.

Airframe parasitic drag coefficient can be determined on the basis of statistical data for the helicopters of this class or calculated by the Serov empirical formulas:

$C_{x_{pd}} = C_x \cdot S = 0.018 \cdot m_0^{0.5646}$ – for helicopters with non-retractable LG;

$C_{x_{pd}} = C_x \cdot S = 0.0174 \cdot m_0^{0.5364}$ – for helicopters with retractable LG,

where m_0 – normal take-off weight, kg;

C_xS – area of equivalent plate, m^2.

MR solidity ratio is calculated by given geometry of MR lay-out.

Angle of the resultant force R needed for level flight is found from the equation (10.82)

$$tg\delta_{l.fl} \approx -\frac{\bar{C}x_{pd}\bar{V}^2}{\sigma_r \cdot t}. \tag{10.85}$$

The trajectory slope maximum angle in steady climbing is determined by the ratio

$$\theta_{c\,lim} = -(\delta_{res} - \delta_{fl.l}), \tag{10.86}$$

here δ_{res} – the angle of inclination of the resultant force R when using the entire available capacity of the power plant for the given flight regime. Vertical speed of steady climb is found from the known relationship

$$V_y = V \cdot \sin\theta_{c\,lim} = (\bar{V} \cdot \omega R) \cdot \sin\theta_{c\,lim}. \tag{10.87}$$

Characteristics of helicopter descending in MR autorotation are calculated based on the well-known definition of helicopter lift-to-drag ratio

$$K_r = \frac{t}{t_{x_c} - t_{x_{l.fl}}} = \frac{t}{t_{x_c} - \bar{C}x_{pd}\frac{\bar{V}^2}{\sigma_r}}, \tag{10.88}$$

where t – rotor traction coefficient on a given flight regime;

$t_{x_{l.fl}}$ – coefficient of rotor propulsive force in horizontal flight;

t_{x_c} – coefficient of rotor propulsive force in MR autorotation.

As the first approximation it can be assumed that the tangent of the helicopter descending in autorotation angle is opposite to its lift-to-drag ratio

$$tg\theta_c = -\frac{1}{K_{aut}}, \tag{10.89}$$

where K_{aut} – helicopter lift-to-drag ratio in autorotation.

Horizontal and vertical components of the helicopter flight speed for descending in autorotation are found from the relation (Fig. 10.51):

$$V_y = V \cdot \sin\theta_c = (\bar{V} \cdot \omega R)\sin\theta_c, \tag{10.90}$$

$$V_x = V \cdot \cos\theta_c = (\bar{V} \cdot \omega R)\cos\theta_c. \tag{10.91}$$

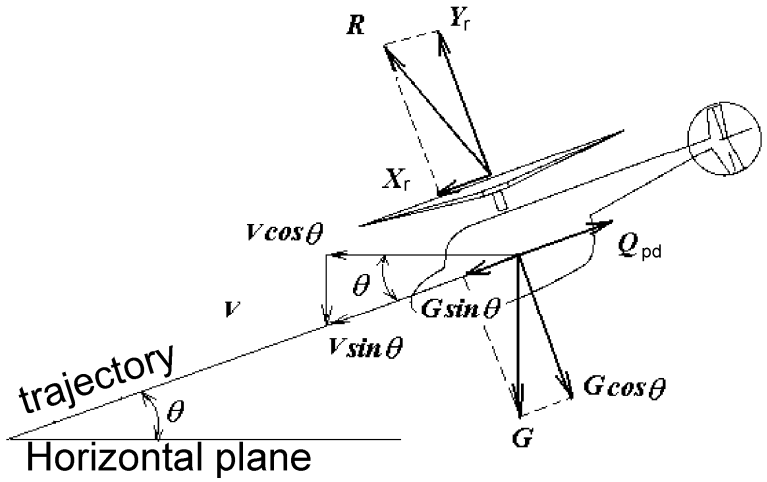


Fig. 10.51. Forces Acting on Helicopter in Descending

When calculating the tail rotor performance it is considered that in level flight the condition of the helicopter yaw trim in straight flight without slipping is true. It does not take into account the impact of main rotor and vertical tail (fin) on the nature of flow about tail rotor and its aerodynamic performance.

From the condition of helicopter yaw trim

$$M_к - T_{TR} \cdot l_{TR} = 0, \quad (10.92)$$

under given assumptions we can find required values of MR thrust and traction coefficient:

$$T_{TR} = \frac{M_к}{l_{TR}}; t_{TR} = \frac{m_k (\omega R)_r^2 F_r \cdot R}{(\omega R)_{TR}^2 \sigma_{TR} \cdot F_{TR} \cdot l_{TR}}, \quad (10.93)$$

where $(\omega R)_{TR}$, F_{TR}, σ_{TR} – TR blade tip speed, disc area, and solidity ratio accordingly.

The proposed method of calculation allows to determine the basic helicopter flight characteristics at the stage of preliminary design, where basic geometrical and mass characteristics of the helicopter and power plant are known.

10.1.5.2. Algorithm for Calculating Basic Helicopter Flight Characteristics

10.1.5.2.1. Initial Data for Calculation

You should have the following for calculation of the helicopter flight characteristics at stage of preliminary design:

1. Engine altitude performances at maximum continuous power. As an example, Fig. 10.52 shows the altitude performances of engine AI-450.

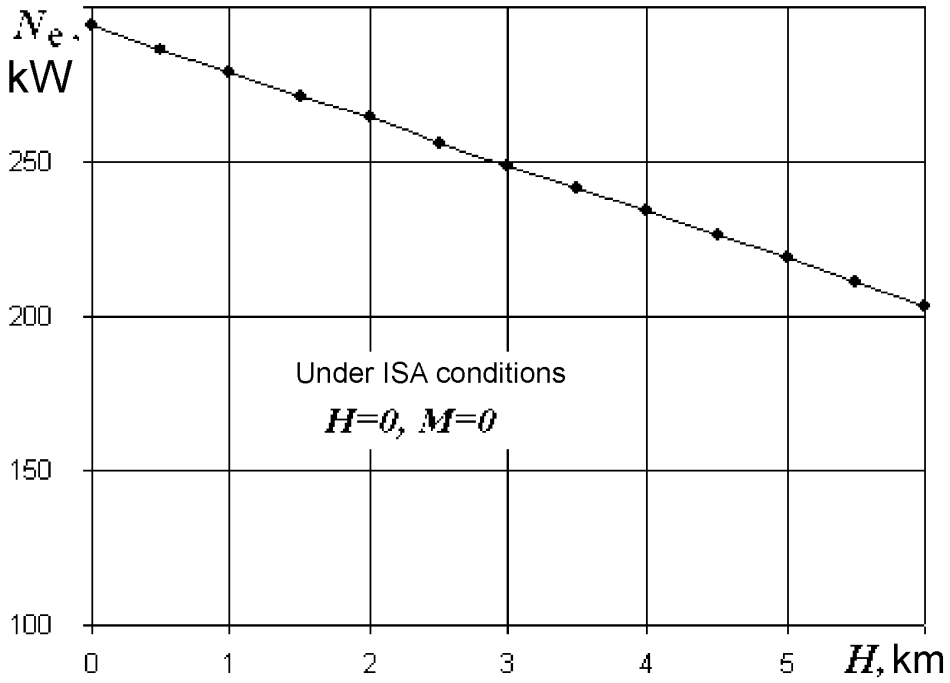


Fig. 10.52 Altitude Performances of AI-450 Engine

2. Power utilization rate of the helicopter power plant depending on the flight speed (Fig. 10.50).

3. Dependence of the helicopter airframe parasitic drag as a function of the angle-of-attack $C_{x_{pd}}(\alpha)$.

4. Geometrical and kinematic characteristics of main and tail rotors: $D_н$, $(\omega R)_н$, $b_н$, K_{bl}, range of MR collective pitch change $\varphi_{min} \le \varphi_0 \le \varphi_{max}$, tip loss factor *B*.

5. Mass and inertia characteristics of MR and TR blade: m_{bl}, $S_{fl.h.}$, $J_{fl.h.}$, γ. Blade static moment relative to flapping hinge $S_{fl.h.}$ and blade inertia moment $J_{fl.h.}$ at stage of preliminary design can be determined from the formulas:

$$S_{fl.h} = \frac{1}{2} m_{bl} \cdot g \cdot R; \qquad J_{fl.h} = \frac{1}{3} m_{bl} \cdot R^2. \tag{10.94}$$

Mass ratio of the blade density is calculated using the formula

$$\gamma = \frac{C_y^{\alpha} \cdot \rho \cdot b \cdot R^4}{2 J_{fl.h}}. \tag{10.95}$$

6. Aerodynamic characteristics of blade airfoil at specific radius $\bar{r}$=0.7 for Mach number averaged over disc. Example of such relations is given in Fig. 10.30.

7. Calculation is performed for different altitudes within the range of flight relative speed from $\overline{V}$=0.1…0.12 to $\overline{V}_{max}$ with specified step.

10.1.5.2.2. Algorithm for Calculating Helicopter Flight Characteristics

Let's consider the computing algorithm for the given speeds $V = \overline{V} \cdot (\omega R)$ and flight altitude H.

1. On the basis of formulas (10.83) – (10.84) for given flight weight G calculate MR traction coefficient t and maximum available torque coefficient m_{tq}.

2. Within the given range of the MR blade setting angle $\varphi_{min} \le \varphi_0 \le \varphi_{max}$ for every value of φ_0 determine:

– MR inflow ratio

$$\lambda = \frac{2t}{c_y^{\alpha} \cdot B^2} - \varphi_0 \left(\frac{2}{3} B + \frac{\mu^2}{B} \right); \tag{10.96}$$

– MR angle-of-attack

$$\alpha_r = \frac{\lambda}{\mu} + \frac{\sigma_r \cdot t}{4\mu\sqrt{\lambda^2 + \mu^2}}; \tag{10.97}$$

– coefficients of the blade flapping first harmonics

$$a_0 = \gamma\left[\frac{B^3}{3}\lambda + \frac{\varphi_0}{4}B^2(B^2+\mu^2)\right] - \frac{g\cdot S_{fl.h}}{J_{fl.h}\omega^2}, \quad (10.98)$$

$$a_1 = 2\mu(\lambda + \frac{4}{3}B\cdot\varphi_0)\cdot\frac{1}{B^2 - 0.5\mu^2}, \quad (10.99)$$

$$b_1 = \frac{4}{3}\mu\cdot a_0\frac{B}{B^2+0.5\mu^2}; \quad (10.100)$$

– lift coefficient averaged over MR disc

$$C_{y_0} = \frac{3t}{(1+3\mu^2/2)}; \quad (10.101)$$

– average value of blade airfoil drag coefficient C_{x_a} on the airfoil polar curve;

– required main rotor torque coefficient, which is calculated by the formula

$$m_k = C_y^\alpha\left[\begin{array}{l}\frac{C_{xp}}{4C_y^\alpha}(1+\mu^2) - \frac{B^2}{3}\varphi_0\cdot\lambda - \frac{B^2}{2}\lambda^2 - \frac{a_1}{8}(B^4+1.5B^2\cdot\mu^2) - \\ \frac{B^2}{4}a_0^2\mu^2 + \frac{B^3}{3}a_0b_1\mu - \frac{b_1^2}{8}(B^4+\frac{B^2}{2}\mu^2) - \frac{B^2}{2}a_1\lambda\mu\end{array}\right]; \quad (10.102)$$

– MR longitudinal force coefficient

$$h = \sigma_r C_y^\alpha\left[\begin{array}{l}\frac{C_{xp}}{2C_y^\alpha}\mu - \frac{B^2}{3}\varphi_0\lambda\cdot\mu + \frac{B^3}{3}\varphi_0 a_1 - \\ \frac{B^3}{6}a_0b_1 + \frac{B^2}{4}a_0^2\mu + \frac{3}{4}B^2a_1\lambda + \frac{B^2}{4}a_1\mu\end{array}\right]. \quad (10.103)$$

To fix speed and MR collective pitch let's find the tilt angle of the resultant aerodynamic force with regard to the rotor axis of rotation

$$\gamma = \operatorname{arctg}\frac{h}{t}. \quad (10.104)$$

Having accounted formulas (10.104) we determine the angle between the vector of total aerodynamic force *R* and normal to flight speed

$$\delta = \alpha_r + \gamma . \qquad (10.105)$$

Having calculated the formulas (10.96) – (10.105) for every angle of MR collective pitch ($\varphi_{min} \le \varphi_{col.p.} \le \varphi_{0max}$), we will obtain dependencies of basic aerodynamic characteristics vs MR blade setting angle $\alpha_r = f(\varphi_0) ..... m_k = f(\varphi_0)$ (Fig. 10.53 – 10.54). These dependencies correspond to given values of μ and its frequency of rotation ω_r=const.

Tilt angle of resultant force R required in horizontal flight relatively to axis Oy_a of wind axis is found from formula (10.85)

$$\delta_{l.fl} = \text{arctg} - \frac{Q_{rot}}{G} = \text{arctg} - \frac{\overline{C}_{x_{rot}} \cdot \overline{V}^2}{\sigma_r \cdot t} .$$

Having used dependencies obtained while calculations and shown in Figs 10.53 and 10.54 we can find characteristics of main rotor required in helicopter level flight $\varphi_{l.fl}$, $\alpha_{l.fl}$, a_0, a_1, b_1, m_k for given MR operation regime μ.

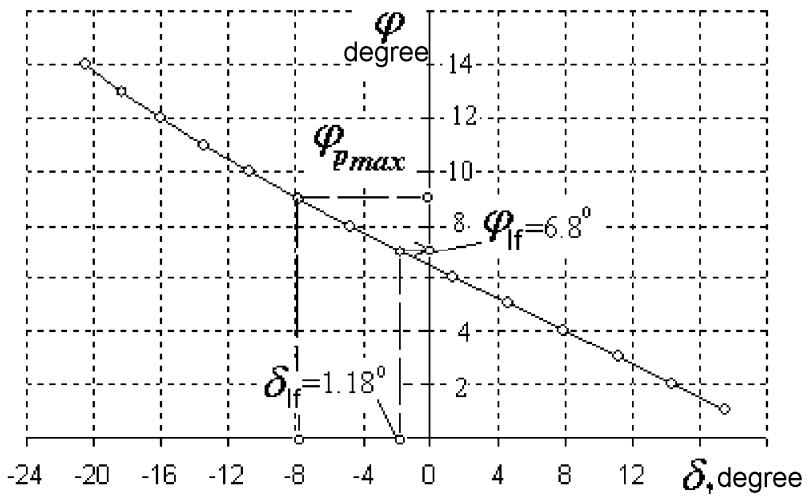


Fig. 10.53. Definition of MR Collective Pitch in Different Flight Regimes

Knowing the engine altitude performances, define the maximum available power that can be delivered to MR:

$$N_p = N_e(H) \cdot \xi, \qquad m_{rot} = \frac{2N_p}{(\omega R)^3 \cdot \rho \cdot F_r}. \tag{10.106}$$

Having applied relations (Figs 10.53 and 1.54), find characteristics of helicopter MR when using total power available: δ_{max}, φ_{pmax}, α_p, a_0, a_1, b_1.

Maximum path angle in steady climbing is determined by formula (10.86). Vertical speed of steady climb is calculated based on the formula (10.87).

It is known that during steady autorotation of main rotor the required power for its rotation is equal to zero $N_{MR} \approx 0$, respectively, the rotor torque coefficient is also zero $m_k=0$. Using the calculated dependence $m_k(\varphi)$, you can determine the angle of MR blade installation, in which the condition $m_k=0$ is true (Fig. 10.54). In general by the MR collective pitch φ_c, at which the rotor works in autorotation with given coefficient μ we can determine other rotor aerodynamic and kinematic characteristics in this regime using the graphics shown in Fig. 10.54.

Needed value of propulsive force in MR autorotation regime we find with the help of expression (Ref. Fig. 10.49)

$$t_{x_c} = t \cdot tg\delta_c. \tag{10.107}$$

With the aid of expression (10.88) determine the value of the helicopter lift-to-drag ratio $K_в$ in autorotation, and with the aid of relations (10.87) - (10.91) we calculate the characteristics of the helicopters in MR autorotation: descent angle θ_{des}, vertical rate of descent.

Knowing the value of required torque on the helicopter MR shaft in level flight or in climb $M_k = m_k \rho (\omega R)_R^2 \cdot R \cdot F_R / 2$ and the distance between the axes of the main and tail rotor in the first approximation, we can identify the needs and values of TR traction and power

$$\mu_{TR} = \frac{V}{(\omega R)_{TR}}, T_{TR} = \frac{M_к}{l_{TR}}, N_p = m_{k_{TR}} \frac{\rho(\omega R)_{TR}^3}{2} \cdot F_R. \tag{10.108}$$

Then, by analogy with the rotor, using TR geometric and kinematic characteristics, find the need to rotor traction coefficient

$$t_{TR} = \frac{2T_{TR}}{\rho(\omega R)_{TR} \cdot F_{TR}}. \tag{10.109}$$

TR aerodynamic characteristics required for helicopter balancing in yaw excluding mutual influence of rotors, vertical stabilizer, slip, etc., can be determined using the algorithm described above (10.96) - (10.105).

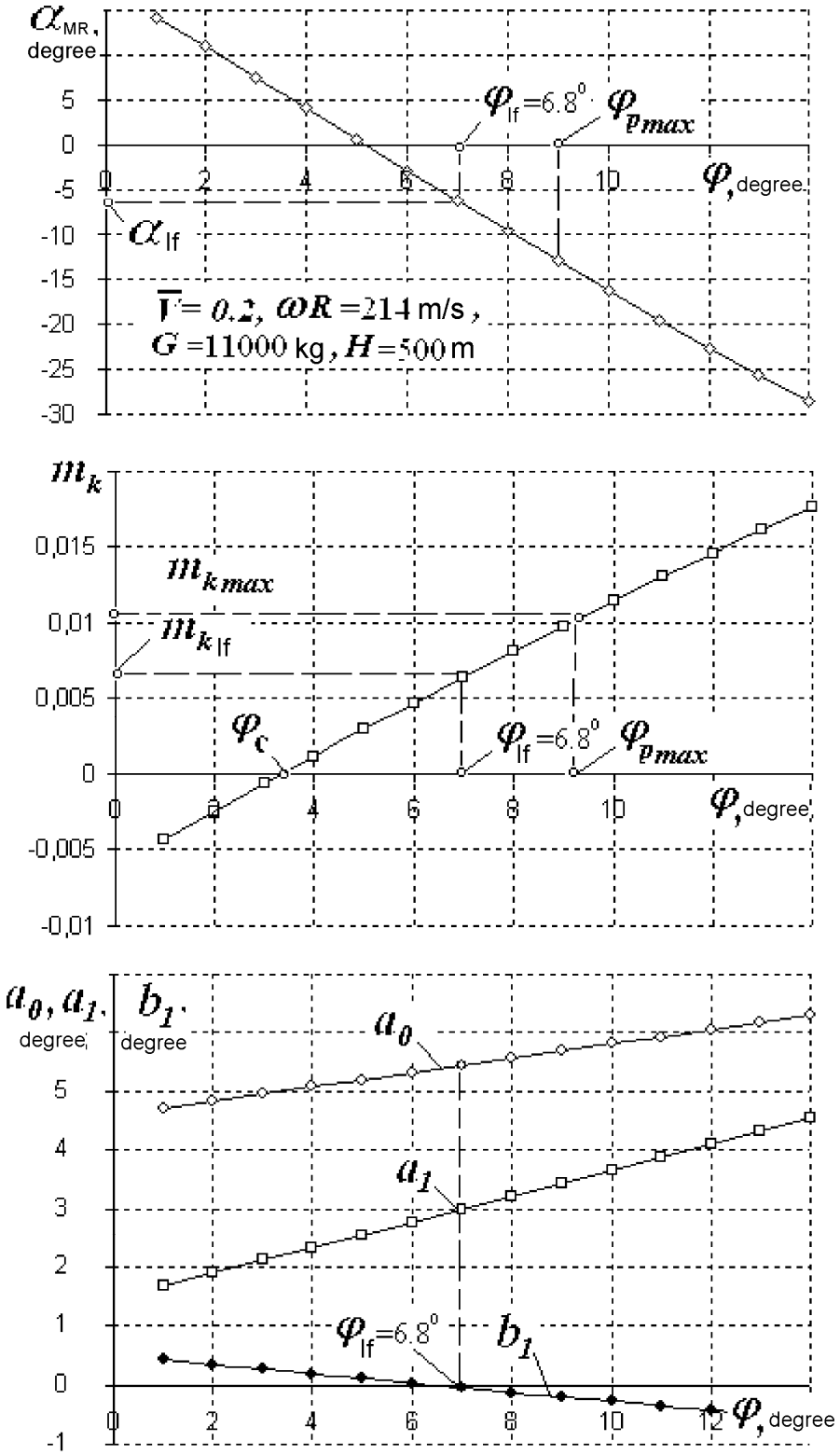


Fig. 10.54. To Calculation of MR Aerodynamic Characteristics at Constant Forward Speed

10.1.5.2.3. Structural Block Diagram of Design Software

Simplified structural design software block diagram of helicopter performance on the basis of the Mile-Yaroshenko modified method is shown in Fig 10.55.

Initial data are entered in the program in three files: two files of geometric, kinematic and inertial mass characteristics of main and tail rotor and one file of assigning characteristics of the helicopter power plants, flying weight, height and other initial data.

When the initial value of the coefficient of helicopter rotor mode $\mu=\mu_0$ (as a rule it is set $\mu_0 \approx 0.1\ldots0.12$) and given value of flight altitude we find need MR traction coefficient t and maximum torque coefficient $m_{k_{max}}$ when using total power available of the helicopter power plant (formulas (10.83)-(10.84)).

The MR aerodynamic characteristics block is intended to calculate MR coefficients and kinematic parameters within the range of the collective pitch variation $\varphi_{min}\leq\varphi\leq\varphi_{max}$ when $\mu = const$. Computing algorithm corresponds to the above given formulas (10.96) - (10.105). Calculation results may be show in a form of graphs according to the user desire (Fig. 10.53 – 10.54).

Next, calculate the characteristics of the helicopter in the horizontal flight mode at a constant value of relative flight speed $\overline{V} = const$: $\alpha_{l.fl}, \varphi_{l.fl}, m_{k_{l.fl}}$ etc. Determine the MR characteristics by using all the available power of the power plant that provides the ability to calculate the characteristics of the vertical rate of climb V_y in forward flight (see formula (10.86) -. (10.87)).

From the condition of the helicopter MR autorotation $m_k \approx 0$ find the value of the rotor collective pitch angle φ_c and calculate the descending angle θ_{des} and vertical speed V_{y_c} in autorotation.

To calculate the characteristics of the helicopter MR recount PB rotor regime coefficient μ_{TR} and determine the MR traction coefficient of condition of yaw balancing of the helicopter. Using the program calculate the aerodynamic characteristics of the rotor at different values of the collective pitch angle, executed as a single block (Fig. 10.55), determine demand thrust and power of tail rotor.

Then go to the new value of the flight speed, which is given by the MR regime coefficient $\mu = \mu + \Delta\mu$ ($\Delta\mu$ is specified discrete calculation step). Calculation is performed to a predetermined operating mode coefficient value $\mu_0 \leq \mu \leq \mu_{çàä}$.

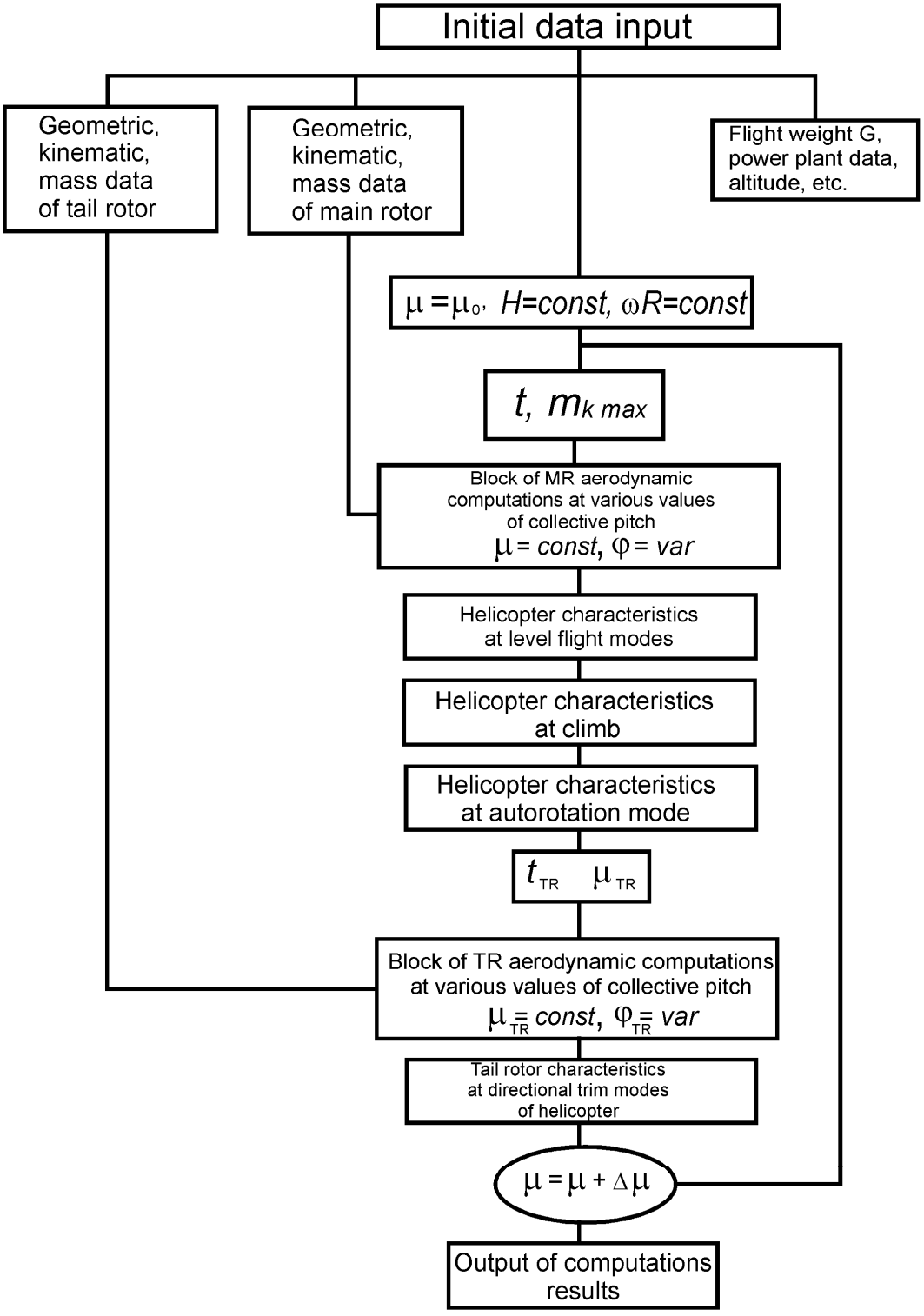


Fig. 10.55. Structural Block Diagram of Calculation Software of Helicopter Flight Characteristics

Output of calculation results of helicopter performance can be done in the form of listings or graphic material. Thus, Figs 10.56 - 10.59 show possible presentation of the information received, suitable for analysis of the projected helicopter performance on various flight modes.

Fig. 10.56 shows the dependence of demand for horizontal flight N_{req} and available N_{av} capacity on flight speed. With the help of this graph it is easy to determine the specific speeds of level flight at a given altitude: economic V_{ec}, optimal V_{opt} and maximum V_{max}. Economic speed determines the flight mode, which provides a minimum power requirement and realized excess of the maximum power ΔN. Most advantageous speed V_{MR} determines the flight mode in which the maximum range is realized.

Fig. 10.57 illustrates the dependence of the required collective pitch from MR speeds for different modes of flight: climb with the maximum available power, level flight at $H=const$, descend in MR HB autorotation mode.

Effect of MR number of revolutions on the helicopter vertical rate of descent in autorotation is shown in Fig. 10.58. This graph allows to determine the optimum speed of the helicopter with the planning engine failure, which provides a minimum rate of descent.

Characteristics of climb of the helicopter in forward flight are illustrated with the graphs shown in Fig. 10.59. With these charts you can define flight regimes of the helicopter, which provide its maximum rate of climb.

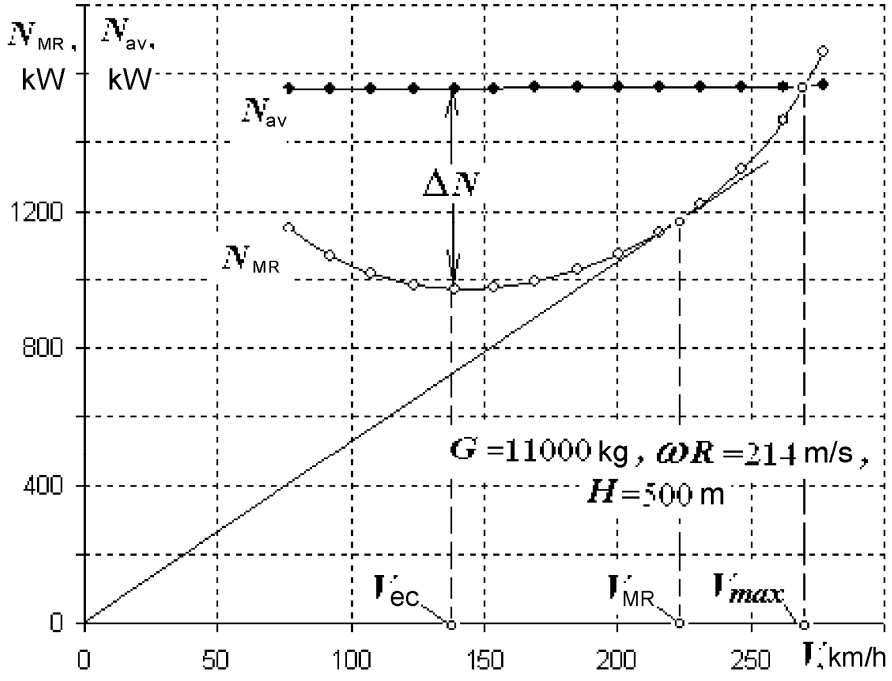


Fig. 10.56. Helicopter Demand and Available Power in Horizontal Flight

The proposed method of calculation of the helicopter performance provides a wide range of flight data with a minimum of initial information. This makes it possible to analyze the impact of a large number of geometric and kinematic parameters of the main and tail rotors, the characteristics of the power plant at the stage of preliminary project.

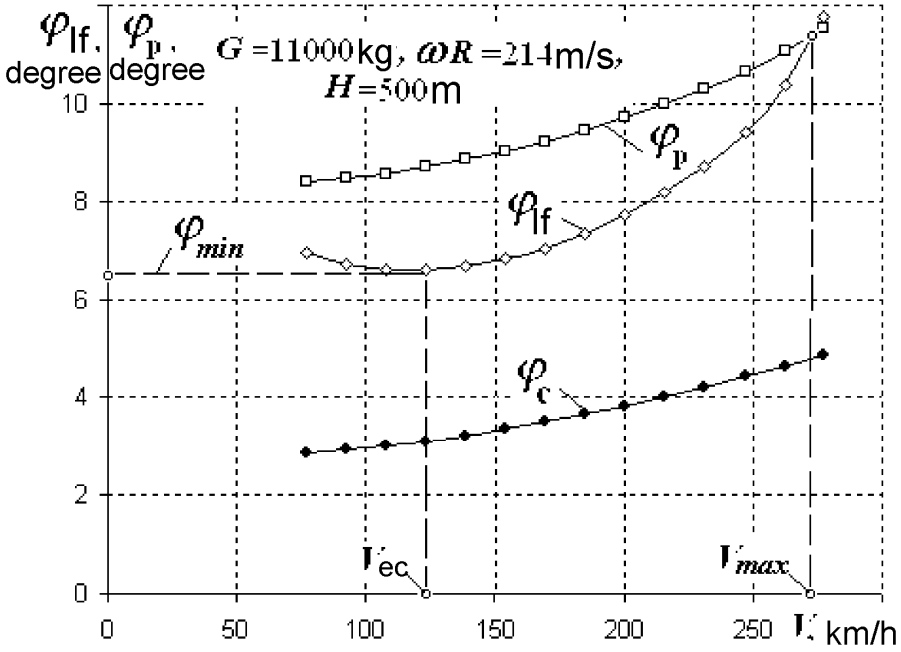


Fig. 10.57. Required and Available MR Collective Pitch Angles in different Flight Regimes

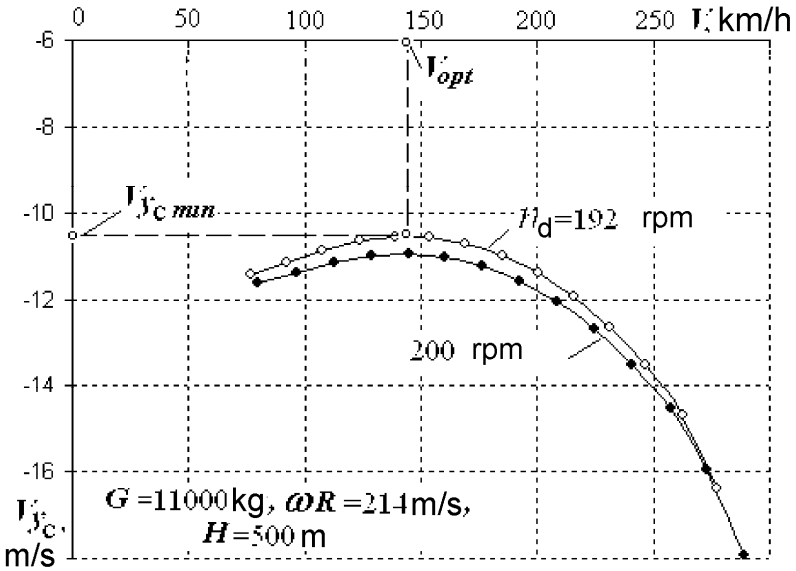


Fig. 10.58. Helicopter Vertical Descending Speed in MR Autorotation

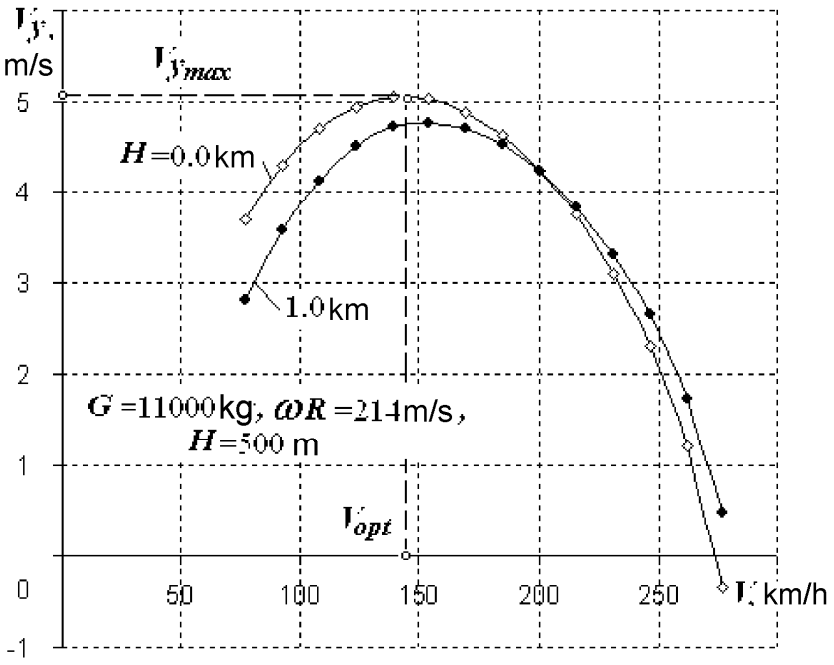


Fig. 10.59. Helicopter Vertical Rate Characteristics in Forward Flight

10.2. Calculation of Loads Acting upon Helicopter and Its Units According to Strength Standards

10.2.1. Terms and Definitions

Flight speed V – flight speed on the instrument with the correction taking into account the systematic error in the readings. It corresponds to the true flight speed in standard atmosphere at sea level.

Cruising speed V_{cr} – speed in horizontal flight at operating altitude at cruise power setting.

Maximum speed V_{max} – maximum speed of horizontal flight at maximum number of revolutions and maximum engine power.

Limit speed V_{lim} – limit allowable steep gliding speed; accepted as higher by 15% of the maximum level flight speed.

Load factor n – the ratio of the resultant of all the forces applied to the helicopter in any loading case and the weight of the rotorcraft.

Operational load factor n_{op} – load factor allowable under normal operational conditions.

Ultimate load factor n_{lim} – load factor at critical landing conditions, allowing for the permanent construction deformation, which is determined by the limit reserve energy absorption capacity of landing gear.

Design load factor n_p – load factor, equal to operational load factor, multiplied by safety factor f.

Projections of n on the axis-related coordinate system are named as:

– n_x – longitudinal load factor;

– n_y – normal load factor;

– n_z – lateral load factor.

Strength margin – ratio of destructive forces or stresses in the structural member to the efforts or stresses acting on the same element at the specified design load on the structure.

Endurance limit – the greatest stress that a material or structure can withstand without breaking repeatedly with a very large number of cycles, called the base. For steel the base is taken equal to 10^7 cycles, and for composite materials - $2 \cdot 10^7$ cycles.

10.2.2. Calculation of Aerodynamic Loads, Acting Upon Hub and Main (Tail) Rotor

The vortex method is widely used to calculate the aerodynamic characteristics of the rotor, and hence the loads acting on the main (tail) rotor, which proposes to expand a vortex sheet and its induced inductive speeds in a Fourier series in harmonics of $\cos\psi$ and $\sin\psi$ of the rotor blade azimuth position in the rotor plane. As a result it is possible to linearize the problem of the rotor rotation and thus represent the main components of the aerodynamic impact in the form of individual components. This allows for more detailed study of influence of geometric and kinematic parameters upon the rotor characteristics.

Rotor is characterized with the following basic parameters: $C_н$ – rotor longitudinal force factor; C_t – rotor thrust factor; C_s – rotor lateral force factor; φ_0 – rotor blade angle of “collective pitch”; m_k – required power factor (torque) to overcome drag caused by the rotor rotation; a_0, a_1, b_1 – rotor blade flapping factors. Specified flight weight of the helicopter allows to find required thrust factor C_t on the basis of given flight speed and blade tip speed μ ($\mu = V \cdot \cos\alpha_e / \omega R$). More over from condition of the helicopter balancing the MR effective required angle-of-attack α_e is determined. Thus rotor mode of operation is fully defined by the selection of three kinematic parameters C_t, μ and α_e. Other rotor characteristics may be obtained from calculations. In some cases when C_t is unknown, it is reasonable to use φ_0 as the reference parameter as it defines definitely C_t. Let’s analyze in detail the components of the parameters:

1. *Rotor Longitudinal Force*

Rotor longitudinal force is considered as composed of the following components:

- ΔC_{Hp} – profile drag caused by surface friction;

- $\Delta C_{H\beta}$ - drag caused by rotor flapping;

- ΔC_{Hv} – drag caused by induced drag developed by the vortex sheets;

- $\Delta C_{H\alpha}$ - drag caused by MR angle-of-attack;

Let's analyze the influence of the rotor parameters upon the components of the rotor drag.

In general the rotor profile drag is determined as follows:

$$\Delta C_{Hp}=\sigma\int_{r_o}^{1}\tilde{b}\left(\frac{1}{2\pi}\int_{0}^{2\pi}C_{x_p}\bar{W}_1^2\sin\psi d\psi\right)d\bar{r}=\frac{1}{2}c_T\mu\chi_R\frac{\sigma}{c_T}c_{xpR}, \qquad (10.110)$$

where

$$c_{xpH}=c_{xpn}^{H}+\Delta c_{xp}^{H}, \qquad (10.111)$$

$$c_{xpn}^{H}=2\int_{\bar{r}_o}^{1}\bar{r}\tilde{b}c_{xpr}^{H}d\bar{r}, \qquad (10.112)$$

$$\chi_H=1-a_{10}tg\alpha_e-0.25a_{10}^2+0.25b_{10}^2-0.25\mu\frac{a_0b_{10}}{\bar{r}}. \qquad (10.113)$$

Coefficient Δc_{xp}^{H} accounts influence of flow speed radial component $V_R=V\cos\alpha_e\cos\psi$, which increases rotor blade drag due to friction forces. Approximately it could be taken that $\Delta c_{xp}^{H}\approx 0.01$.

So it is evident that ΔC_{Hp} is under influence of both profile characteristics and flapping characteristics and flight conditions (equivalent angle of attack α_e).

Longitudinal force factor, which depends on flapping, may be represented in the form of equation:

$$\Delta C_{H\beta}=2\frac{k}{\pi}\int_{\bar{r}_o}^{1}\bar{r}\left[\frac{1}{2\pi}\int_{0}^{2\pi}\tilde{A}(\bar{r},\psi)(a_{10}-a_0\cos\psi)d\psi\right]d\bar{r}\approx$$

$$\approx\frac{c_T}{\chi}\left\{\begin{array}{l}\left[1+0.75\mu(1+1.7\mu^2)\left(-\frac{2\bar{\omega}_z}{a_0}+1.5\mu\right)\right]a_{10}\\+\left[1+0.54\mu^2(1+\mu^2)\right]\bar{\omega}_x\end{array}\right\}, \qquad (10.114)$$

where $\bar{\omega}_x$ and $\bar{\omega}_z$ - dimensionless components of MR shaft angular speed in helicopter curvilinear motion.

Hence $\Delta C_{H\beta}$ is influenced by component a_{10} caused by flapping and helicopter curvilinear motion.

Influence of inductive speeds of vortex sheet upon the rotor longitudinal force (ΔC_{Hv}) is defined by the formula

$$\Delta C_{Hv}=\frac{1}{2\chi^2}\left\{\left[1+1.5\mu^2(1+1.8\mu^2)\right]\int\limits_{\bar r_0}^{1}\frac{\tilde\Gamma^2}{\bar r}d\bar r-(1+1.8\mu^2)\int\limits_{\bar r_0}^{1}\tilde\Gamma^2 d\bar r\right\}c_T^2. \quad (10.115)$$

It follows from (10.115) that ΔC_{Hv} directly depends on law describing circulation distribution over blade span.

Drag dictated by the angle-of-attack is determined as follows:

$$\Delta C_{H\alpha}=-2\mu\alpha_e\frac{c_T}{\chi}\int\limits_{\bar r_0}^{1}\left[\frac{1}{2\pi}\int\limits_{0}^{2\pi}\tilde\Gamma(\bar r,\psi)\sin\psi d\psi\right]d\bar r. \quad (10.116)$$

Having summed (10.110) - (10.116), in general we get the rotor longitudinal force. Note that rotor C_H is under great influence of the blade flapping in addition to speed circulation blade spanwise.

2. *Blade Flapping Coefficients*

Let's consider basic relations defining blade flapping.

As analytical (integral) relations defining a_0, a_{10}, b_{10}, are rather cumbersome and their realization is appropriate in computing programs then let's consider computations of these coefficients as applicable to a rectangular blade with linear twist. Then it could be written:

$$a_0=\frac{\gamma}{C_y^\alpha}\bar r_*\frac{c_T}{\sigma}-\Delta a_0\approx 0.125\gamma\frac{c_T}{\sigma}, \quad (10.117)$$

$$a_{10}=\frac{4}{(1-0.5\mu^2)}\left(\frac{2}{3}\mu\varphi_0+\frac{1}{2}\mu^2\alpha_e-\frac{3}{8}\frac{c_T}{\chi}\right), \quad (10.118)$$

$$b_{10}=\frac{\frac{4}{3}\mu a_0-4\bar V_b}{1+0.5\mu^2}. \quad (10.119)$$

Thus blade flapping coefficients are under great influence of the effective angle-of-attack (α_e), flight mode μ and rotor loading C_t. Note that expansion of rotor inductive speed $\bar V_\alpha$ in Fourier series and thus accounting its irregularity over rotor disc $\bar V_\alpha$ as opposed to classical Glauert theory ($\bar V_\alpha$=const) allows to determine value more precisely (10.119), as otherwise it was considered that $\bar V_\alpha=-c_T/8\chi$. Therefore account of inductive speed irregularity over rotor disc principally influences the value of a_{10} and in general characteristics of flapping.

Note that approach described above allows to compute rotor blade flapping coefficients for its isolated (free) motion. In fact the rotor blade motion is under action of flapping compensator k and angles of tilt of swash plate assembly θ_1 and θ_2. In this case blade flapping coefficients are determined as follows:

$$a_1 = a_{10} + \Theta_{2e}, \quad b_1 = b_{10} - \Theta_{1e}, \tag{10.120}$$

$$\Theta_{2e} = \Theta_2 + kb_1, \quad \Theta_{1e} = \Theta_1 + ka_1. \tag{10.121}$$

Then finally equation (10.120) accounting expression (10.121) will be as follows:

$$a_1 = \frac{a_{10} + kb_{10}}{1+k^2} + \frac{\Theta_2 - k\Theta_1}{1+k^2}, \quad b_1 = \frac{b_{10} - ka_{10}}{1+k^2} - \frac{\Theta_1 + k\Theta_2}{1+k^2}, \tag{10.122}$$

where θ_1 and θ_2 – angles of blade setting by means of swash plate assembly at azimuths $\psi=0$ and $\psi=\pi/2$ correspondingly.

In case of helicopter curvilinear motion its main rotor moves along circle. This in turn causes additional rotor blade flapping. Values of increment of flapping coefficients are defined by expressions:

$$\begin{aligned}
\Delta a_0 &= -\gamma \frac{\omega_x}{6\omega}\mu; \\
\Delta a_1 &= -\left(\frac{\omega_z}{\omega} \frac{\frac{8}{\gamma}+k}{1+k^2} - \frac{\omega_x}{\omega} \frac{1-\frac{8}{\gamma}k}{1+k^2} \right); \\
\Delta b_1 &= -\left(\frac{\omega_x}{\omega} \frac{\frac{8}{\gamma}+k}{1+k^2} + \frac{\omega_z}{\omega} \frac{1-\frac{8}{\gamma}k}{1+k^2} \right).
\end{aligned} \tag{10.123}$$

As in practice only the first harmonic of flapping is considered then relation of the blade flapping angle with its azimuth position is described by the expression

$$\beta_{bl} \approx a_0 - a_1 \cos\psi - b_1 \sin\psi. \tag{10.124}$$

For proper coordination of swash plate and rotor cone driving dynamics a rotor cyclic pitch angle of the blade setting depending on its azimuth position is entered.

$$\varphi_{bl} = \varphi_0 - k\beta_{bl} + \varphi_{ctl}, \tag{10.125}$$

Where under accepted rule of determining θ_1 and θ_2 we have

$$\varphi_{bl} = \varphi_0 - k\beta_{bl} + \Theta_1 \cos\psi + \Theta_2 \sin\psi. \tag{10.126}$$

Analysis of equations (10.125) and (10.126) has shown that cyclic pitch depends on flapping compensator value ($k \neq 0$, motion path (linear or curvilinear) and tilt angles of swashplate assembly χ and η, which relate with θ_1 and θ_2 by means of expressions

$$\Theta_1 = -(D_2\chi + D_1\eta), \quad \Theta_2 = D_1\chi - D_2\eta, \tag{10.127}$$

where D_1 and D_2 – transmission ratios of control system in longitudinal and transverse planes.

Thus blade flapping is defined both by own rotor characteristics and design solutions k and given control θ_1 and θ_2;

3. *Rotor Thrust Coefficient*

Rotor thrust coefficient may be represented by the equation

$$c_T = \chi \frac{k}{\pi} \bar{\Gamma}_{av}, \tag{10.128}$$

where $\bar{\Gamma}_{av} = 2\int_{\bar{r}_0}^{2} \bar{r}\bar{\Gamma}_r d\bar{r}$, and distribution of $\bar{A}_r$ has the following form

$$\bar{\Gamma}_r = B\left[\varphi_e \bar{r}\left(1 + \frac{1}{2}\frac{\mu^2}{\bar{r}^2}\right) + \mu\, tg\alpha_e\right], \tag{10.129}$$

$$B = \frac{4\pi C_y^{\alpha}\mu\bar{b}}{8\pi\mu + C_y^{\alpha}k\bar{b}\left(1 + \frac{\mu^2}{\bar{r}^2}\right)}, \quad \varphi_e = \varphi_{0e} + \Delta\varphi, \quad \alpha_e = \alpha_r + \Theta_{2e}. \tag{10.130}$$

It follows from (10.129) and (10.130) that to determine rotor thrust coefficient you should specify a law of changing blade twist spanwise, blade chord, aerodynamic characteristics of blade airfoils, flight mode and MR angle-of-attack.

4. *Rotor Torque*

Power delivered to rotor for its rotation is intended to overcome torque developed by the blade profile and inductive drag. Then required torque is determined as follows:

$$m_k = m_p + m_i + m_b + m_{in}, \tag{10.131}$$

where m_p - rotor drag longitudinal moment coefficient caused by flow friction over blade surface.

m_i - rotor inductive drag coefficient originating due to rotor normal aerodynamic force (rotor thrust);

m_b - power factor, dictated by curvilinear motion;

m_{in} - power factor depending on inertia Coriolis forces lying in the rotor rotation plane and caused by helicopter curvilinear motion and blade flapping.

When considering components of rotor torque as while analyzing the drag let's write basic formulas in the form useful for analysis.

$$m_p = m_{pn} + \Delta m_p, \tag{10.132}$$

where m_{pn} - longitudinal moment coefficient accounting flow axial motion through the rotor plane (main component);

Δm_p - this component account influence of flow speed radial component $V_R = V\cos\alpha_e \cdot \cos\psi$. Approximately increment of moment coefficient caused by friction forces due to radial speed component may be determined by the formula $\Delta m_p \cong 5\cdot 10^{-3}\bar{V}^2\sigma$.

Design relation to determine profile moment coefficient has the following form

$$m_{pn} = \frac{1}{4}(1+\frac{3}{4}(1+\frac{1}{2}\mu^2)\,a_{10} + 3\bar{V}^2)\sigma\, k_p c_{xpo}, \tag{10.133}$$

where k_p accounts influence of blade shape in planform and average value of drag over disc on torque coefficient

$$k_p = \frac{4}{1+3\bar{V}^2}\int_{\bar{r}_0}^{1} \bar{r}\tilde{b}(\bar{r}^2 + \frac{3}{2}\bar{V}^2)d\bar{r}. \tag{10.134}$$

Torque coefficient dictated by the rotor induced speed is defined by the expression

$$m_i = \frac{c_T^2}{4\chi^2\bar{V}} I, \tag{10.135}$$

where I - rotor induction coefficient, which is found from relation

$$I = 2\int_{\bar{r}_0}^{1} \bar{r}\bar{\Gamma}^2 \left\{1+1.5\mu^2\left[1+1.83\mu^2 - \frac{1.7}{\bar{r}}(1+1.6\mu^2) + \frac{0.7}{\bar{r}^2}(1+1.5\mu^2)\right]\right\} d\bar{r}. \tag{10.136}$$

Rotor induction coefficient I accounts distribution of circulation and vertical component of inductive speed over disc.

Thus when analyzing relations (10.133) and (10.135) it is evident that rotor torque coefficient similar to longitudinal force coefficient depends on blade flapping, speed circulation distribution over disc, and rotor profile characteristics.

Torque coefficients dictated by the curvilinear motion are defined by the equation

$$m_b = \frac{C_y^{\alpha}\sigma}{2\chi\gamma}(\dot{\bar{\omega}}_z\bar{\omega}_z \pm \dot{\bar{\omega}}_x\bar{\omega}_x), \qquad (10.137)$$

And Coriolis inertia forces

$$m_{in} = \frac{C_y^{\alpha}\sigma}{\gamma}(\pm\bar{\omega}_x b_{10} + \bar{\omega}_z a_{10}); \qquad (10.138)$$

5. *Lateral Force Coefficient*

Lateral force may be represented as follows

$$\Delta S = \Delta S_p + \Delta S_\alpha + \Delta S_\beta + \Delta S_v + \Delta S_b$$

In linear steady motion of main rotor the coefficient $c_{xp\psi}$ presents an average over the blade length of the coefficient in symmetrical points in relation to axis OZ (approximately equal $\Delta S_p \approx 0$). This covers also curvilinear motion.

Lateral forece coefficient component, which depends on the angle-of-attack, is determined by the formula

$$\Delta c_{S\alpha} = 2\bar{V}\sin\alpha_e \frac{c_T}{\chi}\int_{\bar{r}_o}^{1}[\frac{1}{2\pi}\int_0^{2\pi}\tilde{\Gamma}(\bar{r},\psi)\cos\psi d\psi]d\bar{r}.$$

After transformation and suppression of small quantities we obtain

$$\Delta c_{S\alpha} \approx -0.8 c_T \bar{V}\sin\alpha_e(1+0.58\mu^2)A_c,$$

where $A_c = \frac{1}{a_o}(\frac{\dot{\omega}_z}{\omega^2} \pm 2\bar{\omega}_x)$ and $A_s = \frac{1}{a_o}(\pm\frac{\dot{\omega}_x}{\omega^2} - 2\bar{\omega}_z)$.

Component, which depends on blade flapping is determined by the formula

$$\Delta c_{S\beta} = -\frac{2c_T}{\chi}\int_{\bar{r}_o}^{1}\{\frac{1}{2\pi}\int_0^{2\pi}\frac{1-A_s\sin\psi - A_c\cos\psi}{1+1.5\mu\sin\psi}\times$$

$$\times[\mu a_0 - \bar{r}b_{10} + (\bar{r}a_0 - \mu b_{10})\sin\psi - \mu a_{10}\cos\psi]d\psi\}d\bar{r}.$$

Having suppressed small quantities the following expression may be obtained

$$\frac{\Delta c_{S\beta}}{c_T} \approx 1.06 b_{10} -$$

$$-0.8\{\mu[1+0.6\mu^2(1+6.3\mu^2)] - 0.7[1-0.6\mu^2(1+1.1\mu^2)]A_s\}a_0 -$$

$$-1.2\mu(1+0.6\mu^2)A_c a_{10}.$$

The component depending on the rotor induced speed is determined by the equation

$$\Delta c_{Sv} = \frac{2c_T}{\chi}\int_{\bar{r}_o}^{1}[\frac{1}{2\pi}\int_0^{2\pi}\bar{v}_y\tilde{\Gamma}(\bar{r},\psi)\cos\psi d\psi]d\bar{r}\,.$$

After proper transformations we obtain

$$\frac{\Delta c_{Sv}}{c_T} \approx -0.08\frac{c_T}{\bar{V}_1}\{[1-\mu\ \ (1-0.68\mu\ \ )]+0.9\mu^2+$$
$$+0.42A_s[\mu-1.3(1+1.2\mu^2)+1.26\mu]-$$
$$-2.5A_c[1+0.6\mu^2-0.33-1.663\mu\bar{V}_1]\}.$$

Let's determine component of the lateral force coefficient depending on the motion curvilinearity

$$\Delta c_{Sb} = \frac{2c_T}{\chi}\int_{\bar{r}_o}^{1}\bar{r}\tilde{\Gamma}[\frac{1}{2\pi}\int_0^{2\pi}\frac{1-A_s\sin\psi-A_c\cos\psi}{1+1.5\mu\sin\psi}\times$$
$$\times\cos\psi(\bar{\omega}_z\cos\psi\pm\bar{\omega}_x\sin\psi)d\psi]d\bar{r}.$$

Having performed integration and proper transformations we obtain

$$\frac{\Delta c_{Sb}}{c_T} = \frac{0.2}{\chi}[\mu(1+1.7\mu^2)(A_s\bar{\omega}_z\pm A_c\bar{\omega}_x)+$$
$$+2.5(1+0.58\mu^2)\bar{\omega}_z];$$

6. MR Hub Moment Coefficients

Moments on the MR hub depend on aerodynamic and mass forces. Thus we obtain coefficients of pitch moment and roll moment accordingly

$$m_{zz} = m_{z\alpha} + m_{zm} + m_{zm}^{(1)} + m_{zfl.h},\ m_{xx} = m_{x\alpha} + m_{xm} + m_{xm}^{(1)} + m_{xfl.h}.$$

Coefficients $m_{zm}^{(1)}$ and $m_{xm}^{(1)}$ account influence of inertia forces dictated by Coriolis and angular accelerations.

Proper components of the hub moment are defined by the formulas

$$m_{z\alpha} = -2\bar{l}_r c_T\int_{\bar{r}_0}^{1}\tilde{\Gamma}[\frac{1}{2\pi}\int_0^{2\pi}\frac{\bar{W}\cos\psi}{1+1.5\mu\sin\psi}\times(1-A_s\sin\psi-A_c\cos\psi)d\psi]d\bar{r}\,.$$

Finally we obtain

$$m_{z\alpha} = 0.5c_T\bar{l}_{fl.}A_c[1+0.58\mu^2-0.8\mu^2(1+1.7\mu^2)\bar{\gamma}_1 \approx 0.5\bar{l}_{fl.}A_c\,.$$

Pitch moment coefficient caused by inertia forces in blade flapping at C_y^{α}=5.6 and even distribution of blade mass, when $\bar{r}_{fl.} = 0.667$, equals:

$$m_{zm} = 4.2\frac{\bar{I}_{fl.}}{\gamma_{bl}}\sigma a_1,\ \ a_1 = a_{10} + \theta_{2e}.$$

Moment of inertia force vertical components relatively to axis $Z_{от}$ caused by Coriolis and angular accelerations is defined by the formula

$$M_{zm}^{(1)} = -\frac{k_{bl} l_{fl.}}{2\pi}\int_0^{2\pi}\cos\psi d\psi(\int_{l_{fl.}}^{R} dk_{\omega_z} + \int_{l_{fl.}}^{R} dk_{\omega_x} + \int_{l_{fl.}}^{R} dI_{\dot{\omega}_z} + \int_{l_{fl.}}^{R} dI_{\dot{\omega}_x}).$$

Here components $dk_{\omega_x} = \pm 2\omega_x \omega r\cos\psi dm$ and $dk_{\omega_z} = -2\omega_z \omega r\sin\psi dm$ are Coriolis inertia forces related with angular speeds ω, ω_x and ω_z, originating on the blade element. Values $dI_{\dot{\omega}_x} = \pm\dot{\omega}_x r\sin\psi dm$ and $dI_{\dot{\omega}_z} = \pm\dot{\omega}_z r\cos\psi dm$ are inertia forces of the blade element dictated by angular accelerations $\dot{\omega}_x$ and $\dot{\omega}_z$. Having performed integration

$$M_{zm}^{(1)} = -0.5 k_{bl} l_{fl.} S_{fl.} \omega^2 a_0 A_c.$$

After transition to coefficients we obtain

$$m_{zm}^{(1)} = 0.5\bar{l}_{\Gamma}\frac{\sigma}{\bar{r}_{\Gamma}}\frac{c_y^{\alpha}}{\gamma_{\Gamma}} a_0 A_c,$$

Hence

$$m_{zm}^{(1)} = 4.2\frac{\bar{I}_{fl.}}{\gamma_{bl}}\sigma a_0 A_c.$$

For roll moment coefficient let's write

$$m_{x\alpha} = -2\bar{l}_{\Gamma} c_T \int_{\bar{r}_o}^{1}\tilde{\Gamma}[\frac{1}{2\pi}\int_0^{2\pi}\frac{\bar{W}\sin\psi}{1+1.5\mu\sin\psi}\times$$

$$\times(1 - A_s\sin\psi - A_c\cos\psi)d\psi]d\bar{r}.$$

After transformation and computation of integrals we obtain

$$m_{x\alpha} = -c_T\bar{l}_{\Gamma}\{\mu(1+1.7\mu^2)(\tilde{\gamma}_1 - 0.75) -$$

$$-0.5A_s[1+1.7\mu^2(1+1.9\mu^2) - 2.24\mu^2(1+1.9\mu^2)\tilde{\gamma}_1]\} \approx$$

$$\approx -0.5\bar{l}_{\Gamma}A_s.$$

Similar to the formulas given above we can write

$$m_{xm} = 4.2\frac{\bar{l}_{fl.}}{\gamma_{bl}}\sigma b_1, \quad b_1 = b_{10} - \theta_{1e},$$

$$m_{xm}^{(1)} = -4.2\frac{\bar{l}_r}{\gamma_{bl}}\sigma a_0 A_s.$$

Moment coefficients $m_{z\,fl.h}$ and $m_{x\,fl.h}$, accounting distance between flapping and drag hinges may be represented as follows

$$m_{z\,fl.h} = -\frac{1}{2}\frac{\bar{l}_{dr.h} - \bar{l}_{fl.h}}{\bar{l}_{dr.h}} m_k b_1,$$

$$m_{x\,fl.h} = \frac{1}{2}\frac{\bar{l}_{dr.h} - \bar{l}_{fl.h}}{\bar{l}_{dr.h}} m_k a_1.$$

10.2.3. Equations of Helicopter Spatial Motion Dynamics

In general case of helicopter spatial motion to obtain equations of motion in body axes you should consider not only motion of the helicopter center of mass, but also dynamics of motion of main (and tail) rotor. Dynamics of rotor motion is defined by the blade flapping. Flapping coefficients (a_0, a_1, b_1) are used to describe flapping, methodic of determination of these coefficients is described in paragraph 10.2.2.

Motion equation are obtained from condition that blades are hinge attached to the hub and considered as absolutely rigid. In arbitrary moment of time vector of linear speed of the body axes origin ($OXYZ$) and vector of fuselage rotation angular speed are arbitrary oriented in space and form angle-of-attack and angle of slip with wind axes ($OX_aY_aZ_a$) (Fig. 10.60).

To determine helicopter spatial position the Euler angles are used, which link normal earth axes ($OX_gY_gZ_g$) and body axes (Fig. 10.61).

Note that system of differential equations obtained for general case of helicopter motion is complicated and contains many non-linear aerodynamic and inertia components. Its application is reasonable for development of flight simulators for fine-tuning automatic control system and definition (selection) of the mathematical model coefficients.

To determine loads acting upon the helicopter units and assemblies it is enough to consider in the spatial motion the equations of motion not accounting dynamics of blade flapping.

Then in general case spatial motion in body axes is described by the system of differential equations.

$$m\left(\frac{d\vec{V}}{dt}+\vec{\Omega}\times\vec{V}\right)=\vec{R}_{aer}+\vec{G}(\psi,\vartheta,\gamma),$$
$$\frac{d(J\cdot\vec{\Omega})}{dt}+\vec{\Omega}\times J\cdot\vec{\Omega}=\vec{M}_{aer}+\vec{M}_{reac}+\vec{M}_{gyr}. \qquad (10.139)$$

First equation of this system describes motion of the center of mass (CM), the second equation describes its rotation relative to the center of mass.

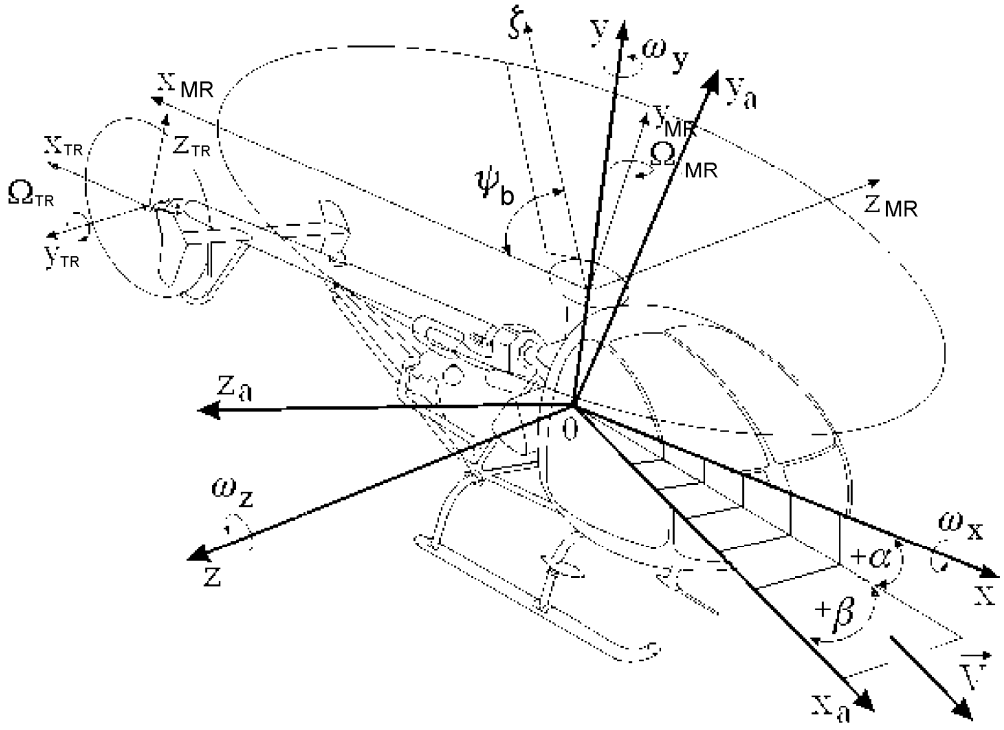


Fig. 10.60. Applicable Axes. Law of Signs

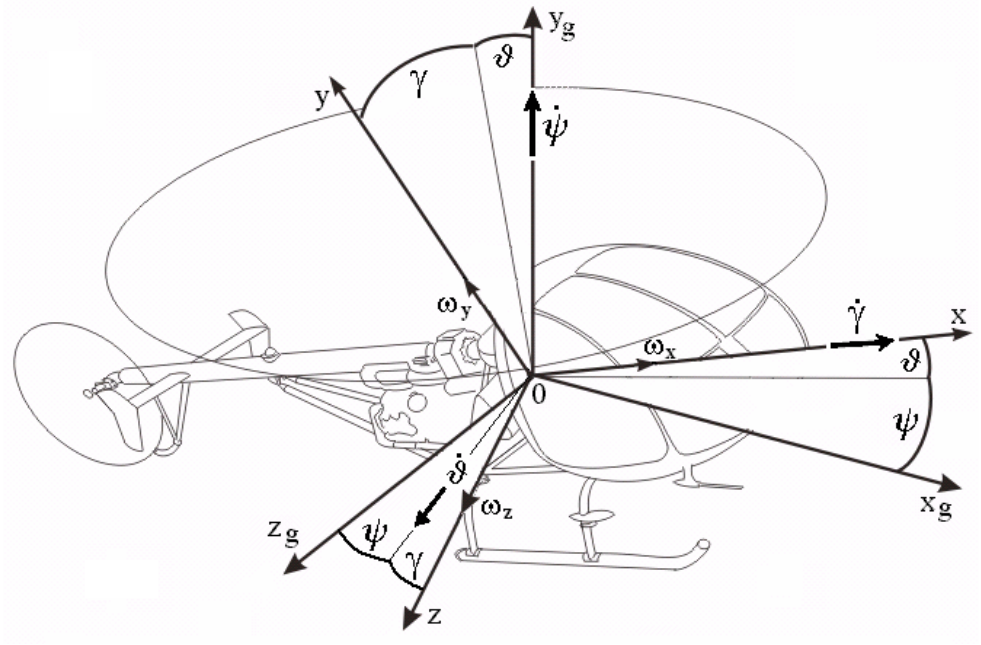


Fig. 10.61. Applicable Axes. Euler Angles

In projections on body axes the first equation of the system (10.139) has the following appearance

$$\begin{aligned} m(\frac{dV_x}{dt} + \omega_y V_z - \omega_z V_y) &= \sum R_x - G\sin\upsilon, \\ m(\frac{dV_y}{dt} + \omega_z V_x - \omega_z V_z) &= \sum R_y - G\cos\upsilon\cos\gamma, \\ m(\frac{dV_z}{dt} + \omega_x V_y - \omega_y V_x) &= \sum R_z + G\cos\upsilon\sin\gamma, \end{aligned} \qquad (10.140)$$

where $V_x=V_{x0}+W_x$; $V_y=V_{y0}+W_y$; $V_z=V_{z0}+W_z$; (W_x, W_y, W_z) – components of wind speed vector in the body axes.

Projections ω_x, ω_y, ω_z of helicopter angular speed vector $\vec{\Omega}$ are related with Euler angles by the following expressions:

$$\begin{aligned} \omega_x &= \dot{\gamma} + \dot{\psi}\sin\upsilon, \\ \omega_y &= \dot{\psi}\cos\upsilon\cos\gamma + \dot{\upsilon}\sin\gamma, \\ \omega_z &= \dot{\upsilon}\cos\gamma - \dot{\psi}\cos\upsilon\sin\gamma. \end{aligned} \qquad (10.141)$$

Feedback is as follows:

$$\begin{aligned} \dot{\upsilon} &= \omega_y\sin\gamma + \omega_z\cos\gamma, \\ \dot{\psi} &= \frac{1}{\cos\upsilon}(\omega_y\cos\gamma - \omega_z\sin\gamma), \\ \dot{\gamma} &= \omega_x - \mathrm{tg}\upsilon(\omega_y\cos\gamma - \omega_z\sin\gamma). \end{aligned} \qquad (10.142)$$

Knowing projections of the speed vector it is possible to determine angles-of-attack and angles of slip by the following formulas:

$$\begin{aligned} \alpha &= \arcsin(-\frac{V_y}{\sqrt{V_x^2 + V_y^2}}), \\ \beta &= \arcsin(\frac{V_z}{\sqrt{V_x^2 + V_y^2 + V_z^2}}). \end{aligned} \qquad (10.143)$$

Left part of the system second equation (10.139) represents itself full derivative of the moment of momentum.

Full derivative dK/dt in the body axes has the following form

$$\frac{d\vec{K}}{dt}=\frac{\partial\vec{K}}{\partial t}+\vec{\Omega}\times\vec{K}=\frac{\partial\vec{K}}{\partial t}+\begin{vmatrix} i & j & k \\ \omega_x & \omega_y & \omega_z \\ K_x & K_y & K_z \end{vmatrix}, \tag{10.144}$$

$$\begin{aligned} K_x &= I_x\omega_x - I_{xy}\omega_y - I_{xz}\omega_z, \\ K_x &= -I_{xy}\omega_x + I_y\omega_y - I_{yz}\omega_z, \\ K_x &= -I_{xz}\omega_x - I_{yz}\omega_y + I_z\omega_z, \end{aligned} \tag{10.145}$$

$$\begin{aligned} &\frac{dK_x}{dt}=I_x\frac{d\omega_x}{dt}-I_{xy}\frac{d\omega_y}{dt}-I_{xz}\frac{d\omega_z}{dt}+\omega_y(-I_{xz}\omega_x-I_{yz}\omega_y+I_z\omega_z)- \\ &-\omega_z(-I_{xy}\omega_x+I_y\omega_y-I_{yz}\omega_z)=I_x\frac{d\omega_x}{dt}-I_{xy}\frac{d\omega_y}{dt}-I_{xz}\frac{d\omega_z}{dt}-I_{yz}\omega_y\omega_y+ \\ &+I_z\omega_y\omega_z+I_{xy}\omega_x\omega_z-I_y\omega_y\omega_z+I_{yz}\omega_z\omega_z=A'-B'-C'+(I_z-I_y)\omega_y\omega_z+ \\ &+I_{yz}(\omega_z^2-\omega_y^2)+I_{xy}\omega_x\omega_z-I_{xz}\omega_x\omega_y, \end{aligned} \tag{10.146}$$

$$\begin{aligned} &\frac{dK_y}{dt}=-I_{xy}\frac{d\omega_x}{dt}+I_y\frac{d\omega_y}{dt}-I_{yz}\frac{d\omega_z}{dt}+\omega_z(I_x\omega_x-I_{xy}\omega_y-I_{xz}\omega_z)- \\ &-\omega_x(-I_{xz}\omega_x-I_{yz}\omega_y+I_z\omega_z)=-A''+B''-C''+I_x\omega_x\omega_z-I_{xy}\omega_y\omega_z-I_{xz}\omega_z^2+ \\ &+I_{xz}\omega_x^2+I_{yz}\omega_x\omega_y-I_z\omega_x\omega_z=-A''+B''-C''+I_{xz}(\omega_x^2-\omega_z^2)+(I_x-I_z)\omega_x\omega_z+ \\ &+I_{yz}\omega_x\omega_y-I_{xy}\omega_y\omega_z, \end{aligned} \tag{10.147}$$

$$\begin{aligned} &\frac{dK_z}{dt}=-I_{xz}\frac{d\omega_x}{dt}-I_{yz}\frac{d\omega_y}{dt}+I_z\frac{d\omega_z}{dt}+\omega_x(-I_{xy}\omega_x+I_y\omega_y-I_{yz}\omega_z)- \\ &-\omega_y(I_x\omega_x-I_{xy}\omega_y-I_{xz}\omega_z)=-A'''-B'''+C'''-I_{xy}\omega_x^2+I_y\omega_x\omega_y-I_{yz}\omega_x\omega_z- \\ &-I_x\omega_x\omega_y+I_{xy}\omega_y^2+I_{xz}\omega_y\omega_z=-A'''-B'''+C'''+I_{xy}(\omega_y^2-\omega_x^2)+(I_y-I_x)\omega_x\omega_y+ \\ &+I_{xz}\omega_y\omega_z-I_{yz}\omega_x\omega_z. \end{aligned} \tag{10.148}$$

System of equations (10.139) should be added with equation, which describes dynamics of MR shaft rotation

$$I_r \frac{d\omega_{MR}}{dt} = M_{eng} - M_{trq}. \qquad (10.149)$$

Motion relatively to normal earth axes is described by the formula

$$\begin{aligned}
&\dot{x}_g = V_{xg} = V_x \cos\psi\cos\upsilon + V_y(-\cos\psi\sin\upsilon\cos\gamma + \sin\psi\sin\gamma) + \\
&+V_z(\cos\psi\sin\upsilon\sin\gamma + \sin\psi\cos\gamma), \\
&\dot{y}_g = V_{yg} = V_x \sin\upsilon + V_y\cos\gamma\cos\upsilon + V_z(-\cos\upsilon\sin\gamma), \qquad (10.150) \\
&\dot{z}_g = V_{zg} = -V_x \sin\psi\cos\upsilon + V_y(\cos\psi\sin\gamma + \sin\psi\sin\upsilon\cos\gamma) + \\
&+V_z(\cos\psi\cos\gamma - \sin\psi\sin\upsilon\sin\gamma).
\end{aligned}$$

To solve the system of equations (10.139) it is necessary to know values of forces and moments included in the system right part in every moment of time t: total aerodynamic force $\vec{R}_{aer}(t)$, weight $\vec{G}$, moments of aerodynamic forces $\vec{M}_{aer}(t)$, reaction moment $\vec{M}_{reac}(t)$ and gyroscopic moment $\vec{M}_{gyr}(t)$ (Fig. 10.62).

The highest difficulty in solving the system of equations (10.139) is introduced by the integration of the second equation, which describes helicopter motion relatively to the center of mass.

Solution of the differential equation system (10.142) features with the following: as the pitch angle υ approaches to the value of $\pm\pi/2$ the value of $\cos\upsilon$ tends to zero, that results in uncertainty (loss of order) in the computational algorithm.

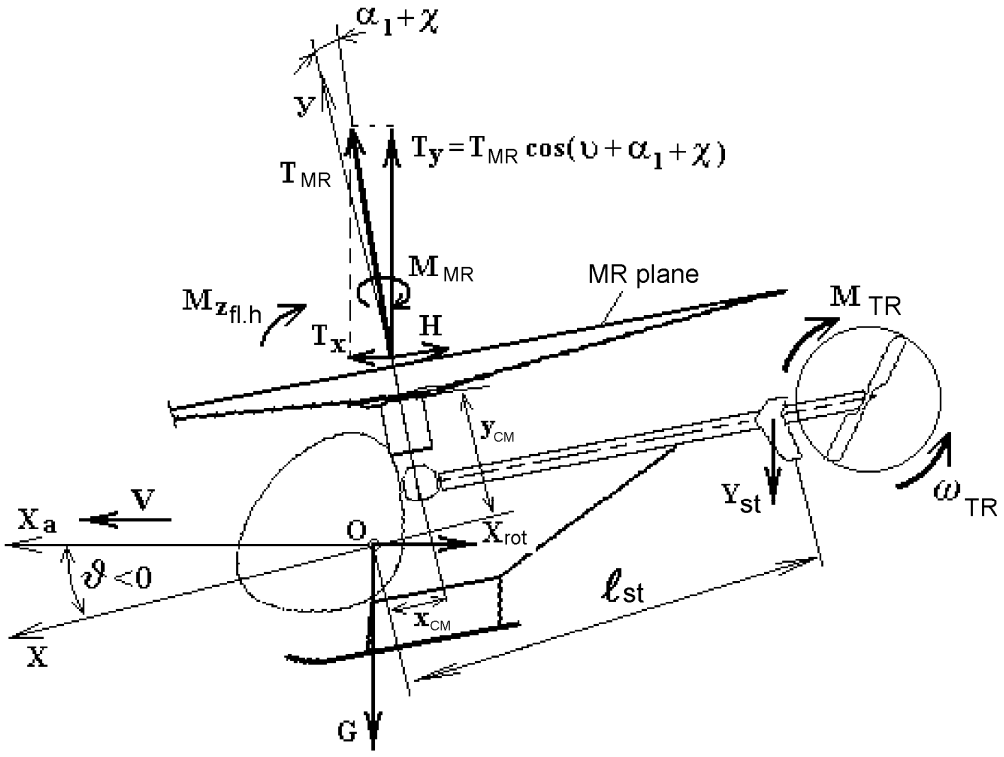


Fig. 10.62. Forces and Moments Acting upon Helicopter

In numerical integration of the system (10.142) the values of angles ψ, υ, γ may vary within an arbitrary range (exceeding $\pm 2\pi$), and accepted variation of these angles lies within the range $\psi \in [0, 2\pi]$, $\upsilon \in [-\pi/2, \pi/2]$, $\gamma \in [-\pi, \pi]$. For their calculation use Rodriguez – Hamilton operators $\Lambda = \Lambda(\lambda_0, \lambda_1, \lambda_2, \lambda_3)$, where

$$\begin{aligned}
\lambda_0 &= \cos\frac{\psi}{2}\cos\frac{\upsilon}{2}\cos\frac{\gamma}{2} - \sin\frac{\psi}{2}\sin\frac{\upsilon}{2}\sin\frac{\gamma}{2},\\
\lambda_1 &= \cos\frac{\psi}{2}\cos\frac{\upsilon}{2}\sin\frac{\gamma}{2} + \sin\frac{\psi}{2}\sin\frac{\upsilon}{2}\cos\frac{\gamma}{2},\\
\lambda_2 &= \sin\frac{\psi}{2}\cos\frac{\upsilon}{2}\cos\frac{\gamma}{2} + \cos\frac{\psi}{2}\sin\frac{\upsilon}{2}\sin\frac{\gamma}{2},\\
\lambda_3 &= \cos\frac{\psi}{2}\sin\frac{\upsilon}{2}\cos\frac{\gamma}{2} - \sin\frac{\psi}{2}\cos\frac{\upsilon}{2}\sin\frac{\gamma}{2}.
\end{aligned} \qquad (10.151)$$

Transition to Euler angles is performed through the formulas

$$\begin{aligned}
&\cos\psi = \left[2(\lambda_0^2 + \lambda_1^2) - 1\right]/\cos\upsilon, \quad \cos\gamma = \left[2(\lambda_0^2 + \lambda_1^2) - 1\right]/\cos\upsilon,\\
&\upsilon = \arcsin 2(\lambda_0\lambda_3 + \lambda_1\lambda_3), \qquad \cos\upsilon = \sqrt{1 - 4(\lambda_0\lambda_3 + \lambda_1\lambda_2)^2},
\end{aligned} \qquad (10.152)$$

which define change of angles ψ, υ, γ within the considered range.

Derived system of differential equations is solved with known numerical procedures. Researches have shown that it is reasonable to solve it by the Runge-Kutt method of the fourth order of accuracy.

Explanation of Origination of Coriolis Acceleration and Rotor Gyroscopic Moment

Coriolis acceleration (complementary acceleration)

$$\vec{a}_k = 2(\vec{\omega}_e \times \vec{V}_r),$$

$\vec{\omega}_e$ – angular speed of movable axes relatively to fixed axes. As applicable to rotor these are angular speeds of the rotor axes $(\vec{\omega}_x, \vec{\omega}_y, \vec{\omega}_z)$;

$\vec{V}_r$ - relative speed of point M in movable coordinate axes, that is speed of point M relatively to movable coordinate axes (Fig. 10.63, 10.64). For example, rotation of rotor blade point $\vec{V}_r = \vec{\omega}_{\hat{a}} \times \vec{R}_M$. It is considered that point M is weightless, that is has no mass.

$$\vec{a}_k = 2\vec{\omega}_e \times (\vec{\omega}_в \times \vec{R}_M) = 2\vec{\omega}_e \times \vec{\omega}_в \times \vec{R}_M = 2\vec{\Omega} \times \vec{R}_M, \qquad \vec{\Omega} = \vec{\omega}_e \times \vec{\omega}_в.$$

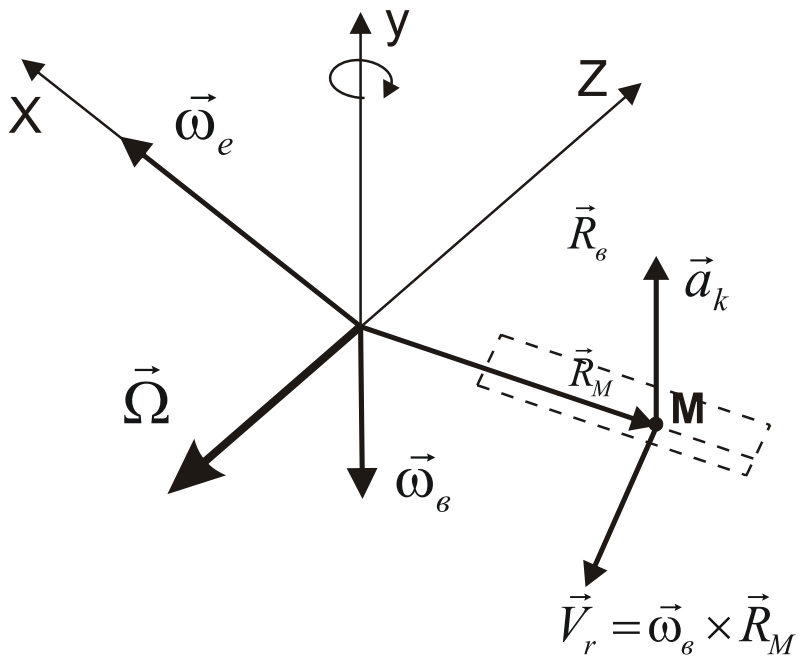
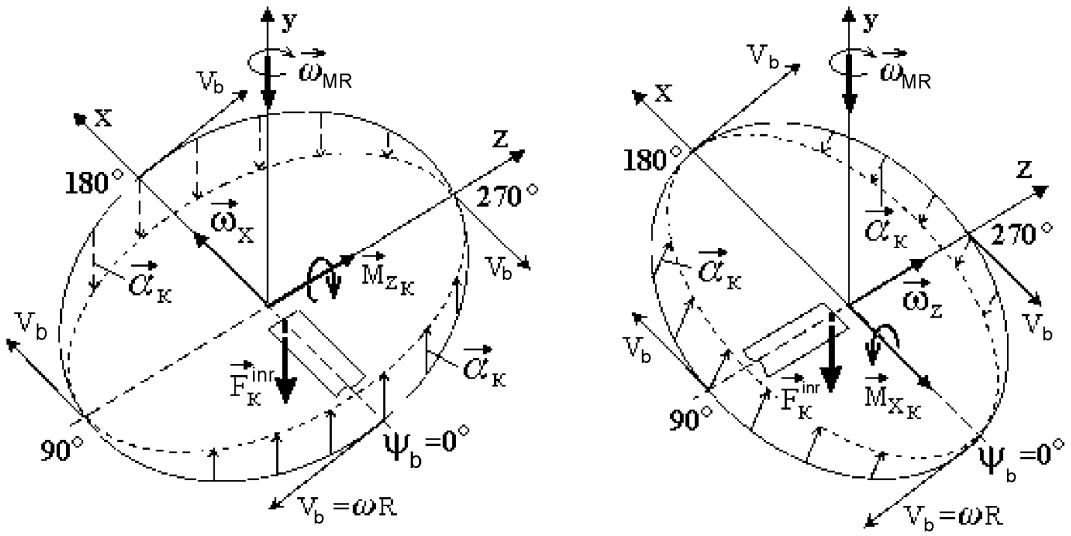


Fig. 10.63. Coriolis Acceleration



$$\vec{\alpha}_K = 2\vec{\omega}_{tran} \times \vec{V}_{rel}$$

$$\vec{\omega}_{tran} = \{\vec{\omega}_X, \vec{\omega}_Z\};\ V_{rel} = V_b = \omega R$$

$$\Rightarrow$$

$$M_{X_K} = -k_b S_r l_r \omega\omega_Z$$

$$M_{Z_K} = k_b S_r l_r \omega\omega_X$$

Inertia moment direction from Coriolis forces

Left hand rotor

perturbance		moment action	
RH roll	$(+\omega_X)$	Nose up	$(+M_{Z_K})$
LH roll	$(-\omega_X)$	Nose down	$(-M_{Z_K})$
Nose up	$(+\omega_Z)$	LH roll	$(-M_{X_K})$
Nose down	$(-\omega_Z)$	RH roll	$(+M_{X_K})$

Fig. 10.64. Rotor Gyroscopic Features

10.2.4. Strength of Helicopter Transmission System

Helicopter transmission system strength is determined by permissible stresses in compliance with requirements to aviation engine transmission.

As the basic design value the maximum torque is taken transmitted from the engine to main rotor, and for transmission of tail rotor – from the engine to tail rotor. With that account variable forces and moments acting upon transmission from the rotor blades, which are determined by results of mathematical modeling or by empirical formulas got on the basis of theoretical points, and also friction force and temperature conditions in bearings of clutches, gears and so on.

Loading design cases: *I-l, II-l, V-l, I-z, III-z*.

In case *II-l* static component of maximum torque on the MR axis is tentatively accepted as equal to moment M_{trq}, determined by case *I-l*, but acting opposite that is from the rotor to engine.

Dynamic strength of transmission shafts

All transmission shafts have to be calculated for bending vibrations and torsional oscillations for all regimes of operation of rotor and engines.

Dynamic tests of transmission or its units are performed with maximum amplitudes of variable stresses got in transmission or in its units during fatigue tests or during any of rotorcraft flight regimes, such as supplying power to rotor (start), take-off at maximum power, acceleration near ground with climbing, level flight with V_{max} and V_{min}, coming out of gliding in flight with engines operating $n_y^{op}=0.6\ n_{max}^{op}$, coming out of gliding with turn and load factor $n_y^{op}=1.5$, deceleration, flight in autorotation, hovering at static ceiling. Number of loading cycles is determined depending on loading case.

Structural tests for case *V-l* are performed under loads corresponding to maximum stress amplitudes measured in flight at V_{cr}. Number of loading cycles is determined by the formula

$$N_1 = 3.6 \cdot 10^3 \eta \nu_1 \tau_1, \qquad (10.153)$$

where τ_1 – total life of the structure under test, hours;

ν_1 – frequency of loading, in seconds;

N_1 – is accepted equal of at least 10^7 cycles for steel and of at least $2 \cdot 10^7$ cycles for duralumin and composite materials.

Coefficient η is calculated depending on number of tested specimens.

Number of tested specimens, n	1	3	6
Reserve coefficient per cycle, η	6	3	2

Dynamic tests for loads of design case *I-l* проводят на силы, действующие на конструкцию в этом случае.

Number of loading cycles is determined by the formula:

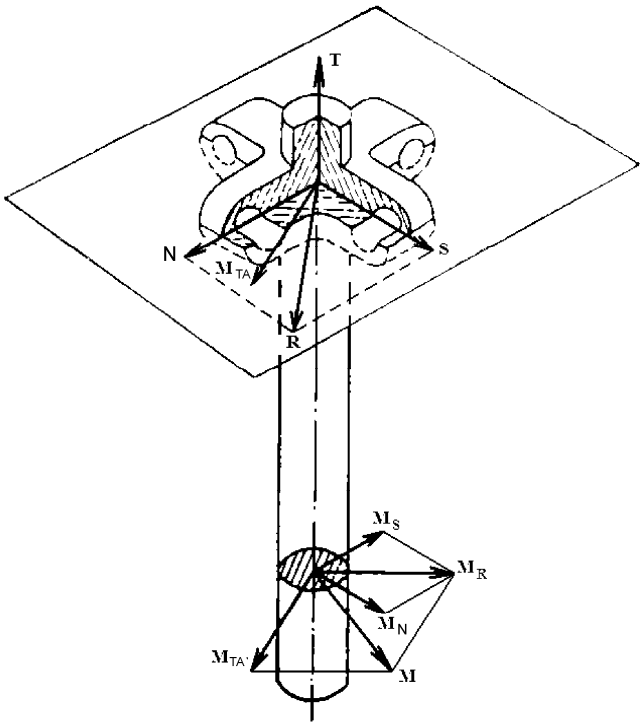
$$N_2 = 12\tau_{max}\nu, \qquad (10.154)$$

where τ_{max} - helicopter total life, hours;

ν – frequency of variable stress maximum amplitudes, s.

Loading Transmission with Loads Developed by Rotor

Main rotor shaft takes in addition to rotor torque other components of force and moment acting upon the hub from blades. In the shaft section perpendicular to its axis in addition to tangent (dictated by torque) normal stresses act. Variable component of normal stress is basically defined by the rotor-created constant force R and moment M in the rotor plane of rotation (Fig. 10.65). The force and moment cause in the rotating shaft variable bending stresses changing with frequency equal to number of rotor revolutions. Force and moment values are found from the helicopter aerodynamic calculation.



10.65. Design Cases of Loading Helicopter in Flight on Condition of Transmission Strength

Loading Cases in Flight

Case I-l – maximum positive load factor

Let's analyze an exit of steep gliding or "steep climb" when supplying power to main rotor, sharp pull of longitudinal control lever and simultaneous increase of MR collective pitch.

Flight regime is defined by the equation $V_{FR} = 1.15V_{max}$ and nominal engine operation.

Maximum vertical operational load factor in the helicopter CG n^{op}_{max} and corresponding number of MR revolutions n_{MR} are accepted to be equal:

$n^{op}_{max}=3$ (according to Airworthiness Requirements) or $n^{op}_{max}=3.5$ (according to AP-29);

$n_{MR}=1.2n_{MR\ nom}$, but not less than $n_{MR}=1.1n_{MR\ max}$,

where $n_{MR\ nom}$ – MR number of revolutions at nominal engine operation;

$n_{MR\ max}$ – maximum permissible MR number of revolutions given in its Maintenance Manual.

MR thrust T^{op}, and also longitudinal H^{op} and lateral S^{op} components of MR aerodynamic force are determined from aerodynamic calculations (subsection 10.2.2) and equilibrium of all forces and moments acting upon the helicopter in maneuvering with specified load factor (subsection 10.2.3).

MR thrust $T^{op}=n^{op}_{max}G$. In absence of data the thrust longitudinal components is assumed to be $H^{op}=0.15T^{op}$, and lateral component – $S^{op}=0.075T^{op}$.

Operational torque on MR shaft is found from the equation:

$$M^{e}_{trq} = kM^{av}_{trq}, \qquad (10.155)$$

where M^{av}_{trq} - mean torque of MR shaft at given power.

k – correction factor depending on power plant type, for gas turbine it equals 1.25.

Case II-l – maximum centrifugal force

Let's analyze exit of gliding in autorotation. Flight regime is characterized with coefficient $\mu=0.2$ at flight altitude $H=0.5H_{DYN}$.

Maximum load factor in aircraft CG is accepted to be equal:

– for maneuvering rotorcraft $n_p{}^{op} = 3.0$;

– for non-maneuvering rotorcraft $n_p{}^{op} = 2.5$.

Maximum number of main rotor (rotors) revolutions corresponding to $n_p{}^{op}$, is defined by vale of $n=1.3n_0$, where n_0 – MR number of revolutions in autorotation for specified flight conditions prior to exit of gliding.

Main rotor thrust $T_{MR}=n_p{}^{op}G$.

Tail rotor thrust $T_{TR}=0$.

Components of main rotor thrust are determined on the basis of aerodynamic calculations and balancing all forces and moments acting upon rotorcraft.

In absence of calculation results the thrust longitudinal component is taken equal $H=0.15T$, and lateral component is taken equal $S=0.075T$.

Case III-l – minimum load factor in flight with engine operating

Let's analyze entry of rotorcraft into steep gliding in motoring flight with $\mu=0.25$.

Flight altitude – at sea level.

Minimum load factor in center of gravity:

– for maneuvering rotorcraft $n_p^{op}=0$;

– for non-maneuvering rotorcraft $n_{min}= -0.5$.

Case IV-l – exit of gliding with steep turn

Let's analyze exit of gliding with steep turn to the left or to the right. Direction of turn is selected opposite to direction of tail rotor thrust.

Flight regime is characterized with coefficient $\mu=0.25$ with number of revolutions and MR collective pitch corresponding to nominal engine power.

Flight altitude – at sea level.

During turn the rotorcraft CG is under action of load factors n_y^{op} and n_z^{op}, which values are determined depending on rotorcraft maneuverability.

For maneuvering rotorcraft it is accepted

$$n_y^{op}=3.0; \quad n_z^{op}=1.25.$$

For non-maneuvering rotorcraft it is accepted

$$n_y^{op}=2.2; \; n_z^{op}=1.0.$$

Maximum roll angle in turn $\gamma<35°$.

Maximum slip angle $\beta=\pm 20°$.

Thrust components are on the basis of aerodynamic calculations. In absence of calculation results the thrust longitudinal component is taken equal $H=0.15T$, and lateral component is taken equal $S=0.075T$.

Case V-l – translation flight with power supplied to main rotor

Let's analyze loading of rotorcraft with forces originating in level flight within the whole range of speeds from the regime characterized with $\mu=0$ (hovering) to regime with maximum flight speed;

For this situation the following regimes are basic:

– accelerated climbing (from $\mu=0$ to $\mu=0.15$);

– level flight at working altitude with cruising speed V_{cr};

– level flight at working altitude with maximum speed V_{max};

– deceleration from $\mu=0.15$ to $\mu=0$.

If during any other regime of translation flight large variable load factors could originate at MR blades (for example, side flight or flight with minimal speed) then consider this regime additionally to specified flight regimes.

For all regimes maximum permitted flight weight is accepted.

Case VI-l – flight with maximum velocity head

Let's analyze motoring flight in gliding with speed $V_{gl}=1.15V_{max}$.

Flight altitude – at sea level.

Cases of Loading Rotorcraft When Flying in Disturbed Air

There is considered effect of disturbed air upon rotorcraft in different regimes of translation flight within the range from $\mu=0$ to $\mu=\mu_{max}$.

When flying with μ_{max} the effective speed of the sharp limited air flow is accepted equal $W=12$ m/s; when flying with $\mu<\mu_{max}$ value of W is determined by the formula

$$W=12\frac{\mu_{max}}{\mu},$$ but not greater 20 m/s.

Case IX-l – Horizontal Wind Gust

Let's analyze action of single wind gust upon rotorcraft, which acts sideward at different speeds in horizontal flight.

Wind gust speed $W=12\frac{\mu_{max}}{\mu}$, but not greater 20 m/s.

More over let's analyze continuous flight with cruising speed at side wind speed $W=15$ m/s.

Rotorcraft loading in both cases is determined according to angle of slip $tg\beta=\frac{W}{V}$.

For rotorcraft with tail rotor the increment of TR thrust is found from the formula

$$\Delta T_{PB}=\pm\frac{\pi}{4}C_{yPB}^{\alpha}\rho W\omega_z R_{PB}^{3}\sigma_{PB}.$$

Case X-l – Vertical Wind Gust

In hovering or during vertical flight under action of upward flow with effective speed of W=12 m/s the MR hub is under action of load factor

$$n^e = 1 + \frac{k\pi}{4} C_y^\alpha \rho \frac{\omega R^3 \sigma}{G} (W - \Delta V_i),$$

where $k=0.7$ – coefficient, which accounts the flow gradient;

ΔV_i – average increment of induced speed over disc area caused by increment of thrust coefficient.

In the presence of forward speed consider a helicopter loading under action of vertical gust when flying in disturbed atmosphere within the range of regime coefficients from $0.5\mu_{max}$ to μ_{max}. Assuming that helicopter incomes into sharply limited air flow. Value of operational load factor is determined by flight speed and effective speed of upward flow

$$n_y^e = 1 + \rho V W \frac{C_{yMR}^\alpha S}{2G}.$$

Case Xl-l – Turn in Hovering

In hovering with nominal numbers of revolutions and nominal engine power let's consider a rotorcraft sharp turn relative to its vertical axis for 360°, done using pedals and with average angular speed of $\omega_y=\pm 1$ rad/s.

Average angular speed may be normalized by the value determined by the empirical formula

$$\omega_y^{norm} = 0.05\omega_{MR\,nom},$$

where $\omega_{MR\,nom}$ – MR angular speed at nominal engine power.

For single-rotor vehicles the turn direction is oppose to TR thrust.

Cases of Loading During Landing

Rotorcraft strength and reserve energy absorption capacity are determined for two basic kinds of landing:

a) vertical landing with power applied to main rotor;

b) landing with gliding approach to aerodrome without power applied to main rotor (landing with autorotating main rotors).

It is assumed that thrust direction coincides with MR axis of rotation excluding for special cases.

For rotorcraft equipped with two or more main rotors account joint thrust of all rotors simultaneously.

Below cases of loading are considered providing for operation of rotorcraft under bad visibility.

Descending speed in vertical landing with simultaneous hit by forward and rear LG wheels

In vertical landing with power applied to main rotor (rotors) the descending speed

$$V_y^e = 0.1\frac{G}{N_{max}}\sqrt{\frac{F}{\sum F}} \text{ or } V_y^e = 0.12\frac{G}{N}\sqrt{\frac{F}{\sum F}}.$$

Maximum allowable descending speed

$$V_{y1}^{norm} = 0.15\frac{G}{N}\sqrt{\frac{F}{\sum F}}, \text{ but not greater than 2.5 m/s.}$$

Here N (N_{max}) - nominal (maximum) rotorcraft power, in h.p.;

G – rotorcraft flight weight, kg;

$\sum F$ – total area of MR disc areas.

Descending speed in gliding landing without applying power to main rotor (rotors)

In gliding landing without power application to main rotor (rotors) the limit descending speed equals:

For rear (main) landing gear:

$V_{y2}^{norm} = 0.8\sqrt{\frac{G-Y}{\sum F}}$, but not less than $V_{ymin}^{norm} = 1.25V_y^e$ and not more than V_{ymax}^{norm} =3.6 m/s;

For nose LG

$V_{y2}^{norm} = 0.6\sqrt{\frac{G-Y}{\sum F}}$, but not more than 85 % of accepted for rear landing gear;

where Y – wing lift, determined for given landing speed at maximum trim angle.

Reserve energy absorption capacity. Earth reaction vertical components

Impact energy, absorbed during landing by the rotorcraft landing gear that is landing gear absorption capacity, is determined by formulas

$$A = \Delta E = \frac{m_{red}V_y^{e\,2}}{2},$$

$$A_1^{norm} = \Delta E = \frac{m_{red}V_{y1}^{e\,2}}{2},\ A_2^{norm} = \Delta E = \frac{m_{red}V_{y2}^{e\,2}}{2},$$

where m_{red} — reduced rotorcraft mass.

For nose LG (Fig. 10.66, a)

$$m_{red} = \frac{G}{g(1 + \frac{a^2}{i^2})}.$$

For rear LG during vertical landing (Fig. 10.66 b)

$$m_{red} = \frac{G}{g(1 + \frac{b^2}{i^2})},$$

where $i = \sqrt{\frac{I_z}{M}}$ – inertia radius.

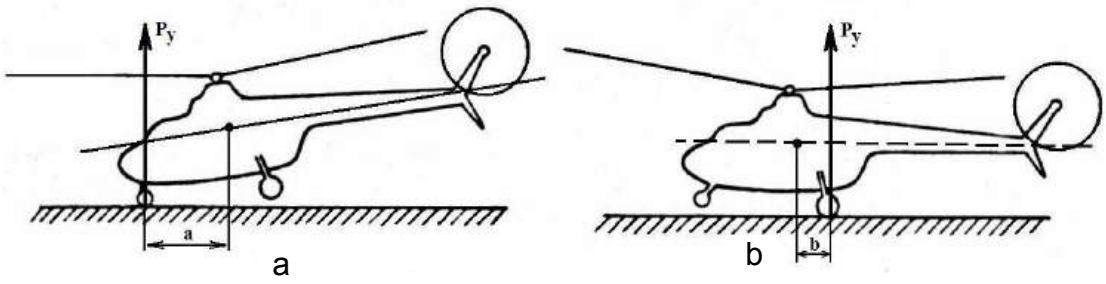


Fig. 10.66. Landing with Hit by Nose (a) or Rear (b) Landing Gear

Calculation of rotorcraft shock absorption is performed using maximum among two values of limit absorption capacity A_1^{lim} and A_2^{lim} (Fig. 10.67).

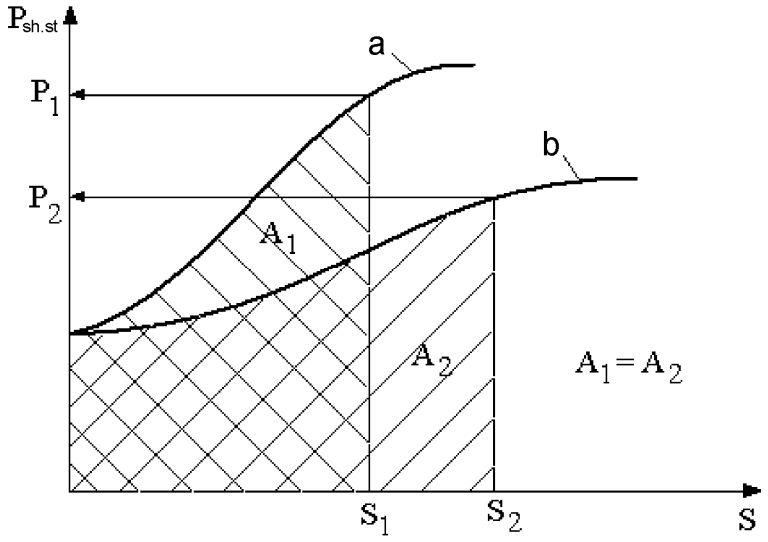


Fig. 10.67. Shock Strut Compression Diagram:
a – “rigid” shock strut, b – ”soft” shock strut

Consider the characteristics of the LG compression (Fig. 10.66). Knowing the energy that must be absorbed by the shock struts, using these characteristics is easy to determine the maximum value of the vertical force. Obviously, the area under the curve $P_{sh.s}=f(S)$, where $P_{sh.s}$ is compression forces, S – shock strut stroke, i.e. compression force work A should be equal to the kinetic energy ΔE. Thus, the maximum amount of compression force is determined from the condition $A=\Delta E$ and depends, therefore, not only on the ΔE, but also on the characteristics of the LG shock struts. If the shock strut is designed so that compression characteristics is gently sloping (Fig. 10.67, curve b), the maximum value of the compression force will be less than the steep characteristic (Fig. 10.65, curve a). However, the need stroke in the first case is greater than in the second case, so that the same amount of energy has been absorbed.

In accurate calculations you must take into account that part of the energy is absorbed during compression of the wheel. Since the dependence of the wheels compression applied to them is close to linear, the wheel tire work A_{TIRE} can be expressed by the approximate formula

$$A_{tire} = \frac{c_w P_w^2}{2},$$

where c_w – wheel tire stiffness;

P_w – vertical force acting on the wheel.

The force acting along the shock strut axis, i.e. compression force can be expressed in terms of the reaction force. Force $P_{wH} = P_V$.

It follows from the above the method for determining the force P_y in checking calculations. It is calculated from the known impact energy ΔE and built characteristic of compression $P_{sh.s}=f(S)$ taking into account the work of wheel tire A_{tire}. Determine also load factor

$$n_y = \frac{P_y}{P_{park}},$$

where P_y – force value calculated as stated above;

P_{park} – parking load on the same LG leg.

Engineering calculations come in reverse order. Set the value of n_y^{op}. Determine the maximum amount of force

$$P_y^{op} = P_{park} \cdot n_y^{op}.$$

The value P_y^{op} is used to design a shock strut. In order to prevent essential weighting of structure the value of n_y^{op} should be assigned not more than 2 – 2.5. This value of n_y^{op} as usual is permissible also by considerations of wheel strength. As a rule catalogues contain maximum allowable load for the wheel of certain type. It should be not less than force P_y^{op}.

It follows from the foregoing that the landing load value is not uniquely defined. The designer can design a rigid shock strut for shock energy absorption during landing and therefore landing loads will be larger. If there were no limitations on the size of shock struts, it would be possible to provide a very low loads during landing by designing the strut with a soft characteristic.

Operational n^{op} and limit n^{lim} overloads during rotorcraft landing

$$n^e = \frac{P^e_{w\,max}}{P_w}.$$

Value of operational overload n^{op} is determined as a ratio of wheel maximum operating load $P^e_{w\,max}$ at absorption capacity A^{op} to parking wheel load:

$$n^e = \frac{P^e_{w\,max}}{P_w}$$

Limit overload n^{lim} is calculated as a ratio of wheel maximum load $P^{norm}_{w\,max}$, acting on the wheel at limit absorption capacity $A_1{}^{lim}$ or $A_2{}^{lim}$, to the wheel parking load P_w:

$$n^{norm} = \frac{P^{norm}_{w\,max}}{P_w}.$$

Case *I-p – symmetrical landing on nose and rear landing gear*

Consider vertical landing with simultaneous hit by nose and rear LG legs (Fig.10.64).

Rotorcraft is in parking position; LG compression corresponds to absorption capacity $A_1{}^{lim}$. Operating and limit overloads for nose and rear LG are determined as it was specified above, overload n^{op} is not less than 2.

Earth reaction is directed vertically upward and is applied to the center of the tire area contacting with ground surface.

Main rotor (rotors) thrust is accepted equal to $T=0.75T_0$ or $T=0.25T_0$ depending of design case.

Case II-p – one-side landing on the front and rear landing gear

Consider landing of a rotorcraft of four-wheel configuration on a slope site or landing with roll. The site lateral slope or rotorcraft roll equals 10°.

Forces acting on wheels are directed perpendicular to ground surface and their value the same as in case I-p.

Main rotor thrust equals $T=0.25T_0$.

$$\frac{B_{nose}}{B_{rear}} \leq 0.5.$$

Case II-p is not considered for rotorcraft having the ratio of forward track B_{nose} to rear track B_{rear}.

Case III-p – asymmetrical landing on one leg of nose LG and one leg of rear LG

Consider vertical landing (V=0) of rotorcraft with four-wheel LG configuration on uneven site while simultaneous hitting by one forward leg and one rear leg located diagonally .

Forces acting on wheels are directed vertically upward and accepted similar to case I-p.

Two other LG legs do not touch ground.

Case III-p is not considered for rotorcraft having the ratio of forward track B_{fwd} to rear track B_{rear}:

$$\frac{B_{nose}}{B_{rear}} \leq 0.5.$$

Case IV-p – landing on rear LG (special case I-p)

Consider landing on rear LG (V=0). Rotorcraft stands on two legs of rear LG (Fig. 10.64, b) and on tail bump with shock strut compression corresponding to absorption capacity A_2^{lim}.

In the absence of tail bump the rotorcraft trim (clearance) backward is assumed to be equal to maximum permissible angle of turnover $\theta_{w.p.}$.

Forces acting on rear LG wheels are directed vertically upward. They are determined according to absorption capacity A_2^{lim} and helicopter position at the moment of landing.

Force acting upon the tail bump (tail skid), is determined by the absorption capacity diagram for the tail skid absorption units at A_w^{lim} and directed upward vertically.

Main rotor thrust $T=0.75T_0$.

Case V-p – hit against ground by nose LG wheels (special case I-p)

Consider hit against ground by nose LG wheels after rotorcraft landing on rear LG with maximum angle of turnover backward (case IV-p).

Helicopter is considered as standing on nose LG wheels with shock struts compressed (Fig. 10.64, a) according to absorption capacity A_1^{lim} and touching the ground by rear LG wheels with not compressed shock struts.

Forces acting on wheels are directed vertically upward and accepted similar to case I-p. Rear LG wheels are not loaded. Main rotor thrust T=0.

Case VI-p – asymmetrical landing on rear LG (special case of I-p)

Consider asymmetrical vertical landing on rear LG (V=0). Helicopter position is determined by compression of loaded leg according to A_1^{lim} and not compressed shock struts of other legs, touching the ground surface.

Force acting upon the LG loaded leg and directed upward vertically is accepted similar to case I-p.

Main rotor thrust $T=0.25T_0$.

Case VII-p – landing with not spun-up wheels with translational speed on rear LG (only nose LG after the helicopter was rolled over).

Let's analyze two variants of loading during landing in autorotation on rear LG with not spun-up wheels.'

1-st variant. The rotorcraft is considered in the landing moment on rear LG at maximum allowable angle of overturn and compression of shock struts corresponding to forces acting upon the wheel.

Every leg of rear LG is under action of:

– vertical force $P_y=0.7P_y^{lim}$ (value of P_y^{lim} is determined by A_2^{lim});

– horizontal force directed backward $P_x^{lim}=n_x^{lim}\sum P_{wh.rear}$

$$n_x^{lim}=\frac{3.2}{R_1}\sqrt{\frac{I_k V_{land} n_y^{lim\,3/2}}{S^{1/2}P_{rear w.}}},$$

where $\sum P_{wh.rear}$ – total parking load on wheels of one leg of rear LG, kg.

Here n_y^{lim} is determined by limit absorption capacity A_2^{lim};

I_{wh} – wheel moment of inertia, kgms^2;

R_1 – rolling radius of compressed wheel, m;

S – vertical projection of LG shock strut stroke, m;

V_{land} – horizontal speed of rotorcraft landing in autorotation without usage of MR collective pitch, m/s.

Value of n_x^{lim} is accepted as minimum $0.25\ n_y^{lim}$.

2-nd variant. Rotorcraft is in parking position. Shock struts are compressed according to loads on LG. Nose LG wheels touch the ground but not loaded.

Every leg of rear LG is under action of forces P_y^{lim} and P_x^{lim}, which are determined similar to the first variant of loading. Force P_x^{lim} acts with alternating sign in direction of rotorcraft motion.

In both variants of LG loading thrust $T=0.75T_0$.

<u>To explanation or ground reaction influence in landing site plane</u>

When landing with translational velocity the force P_x originates due to friction against ground surface in addition to force P_y, due to the fact that wheels are not spun-up (Fig. 10.66,а). At initial step of wheel spin-up (Fig. 10.66,б) it is directed opposite to helicopter speed and first rises in absolute value, and then reduces to zero. Feature of changing force P_x with time is explained by the laws of motion and spin of the wheel and its hub.

Initially, the angular velocity of the wheel ω_{wh} is small. Therefore, the circumferential speed of the wheel at the point of touching the ground $\omega_{wh}R_{wh}$ is less than wheel hub speed V_{hub} that at the initial time equals to the translational speed of the helicopter. Total speed of the wheel surface point relative to the ground $V_{wh}=V_{hub}-\omega_к R_к$ is directed forward. Originating force of friction, that is forces P_x, is directed backward respectively. Influence of the force P_x causes not only the wheel spin-up, but also the LG deformation, which is the reason of wheel hub shift in the direction opposite to the direction of helicopter motion. Wheel angular velocity ω_{wh} increases due to its spin-up. At some point in time the peripheral speed $\omega_{wh}R_{wh}$ becomes equal to V_{hub} that means the wheel has spun-up and P_x becomes zero. Subsequently (Fig. 10.68, c) the landing gear oscillates, and in LG attachment nodes reactions appear transmitted to the fuselage and directed alternately forward and backward. The action of these forces is accounted so that the force P_x in calculations is

considered not only directed backward, but also forward. The greatest value of the force P_x, when it is directed forward or backward is determined by special calculations. Approximately one can assume that they are approximately equal in magnitude and expressed as $\pm P_{x\,max}=0.5\text{-}0.7P_y$.

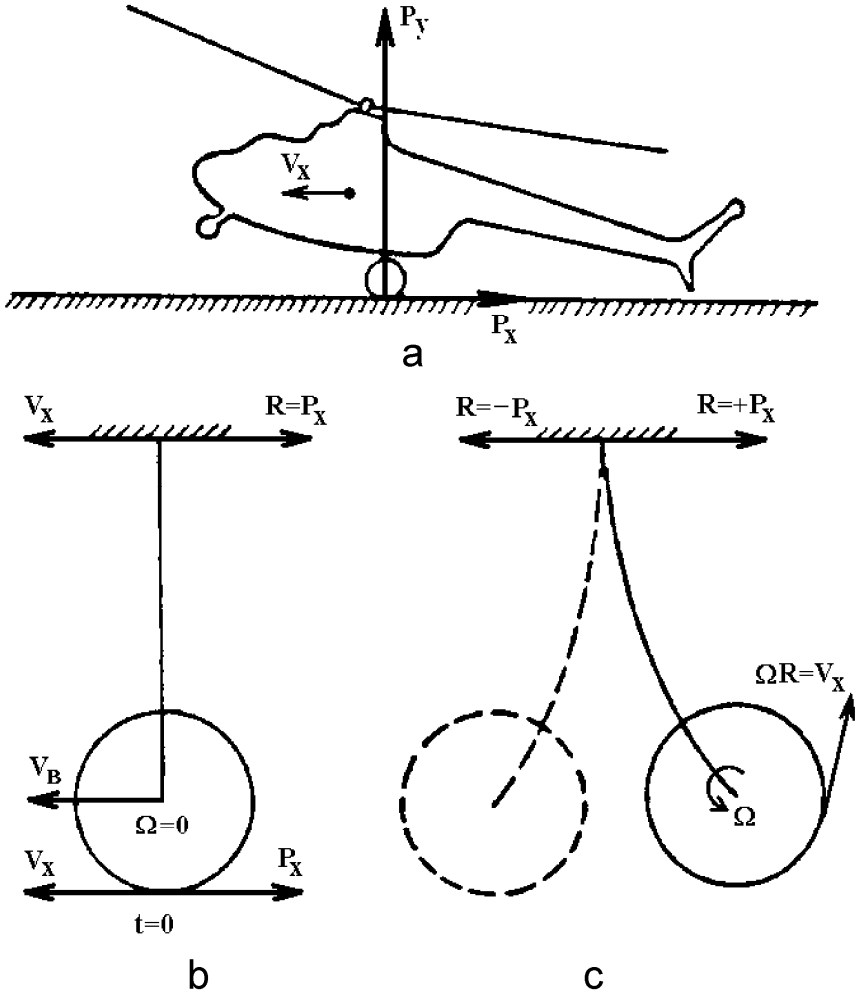


Fig. 10.68. LG Loading When Landing with Transnational Speed

a – helicopter landing;

b – starting moment of wheel spin-up;

c – LG oscillations

Case VIII-p – Landing with Side Hit

Let's analyze landing of rotorcraft with drift. Rotorcraft is in parking position with proper compression of loaded wheels (Fig. 10.69).

If right (left) leg of rear LG is loaded with force

$$P^{op}_{hit\,rear} = 0.7n^{op}_{rear}\sum P_{wh\,rear}, \quad P^{op}_{z\,rear} = 0.35n^{op}_{rear}\sum P_{wh\,rear},$$

then left (right) leg is loaded with force

$$P^{op}_{hit\,rear} = 0.3n^{э}_{rear}\sum P_{wh\,rear}, \quad P^{op}_{z\,rear} = 0.15n^{op}_{rear}\sum P_{wh\,rear},$$

Force acting upon the nose leg is determined by the formula

$$P^{op}_{hit\,front} = 0.5n^{op}_{front}\sum P_{wh\,front}, \quad P^{op}_{z\,front} = 0.25n^{op}_{front}\sum P_{wh\,front},$$

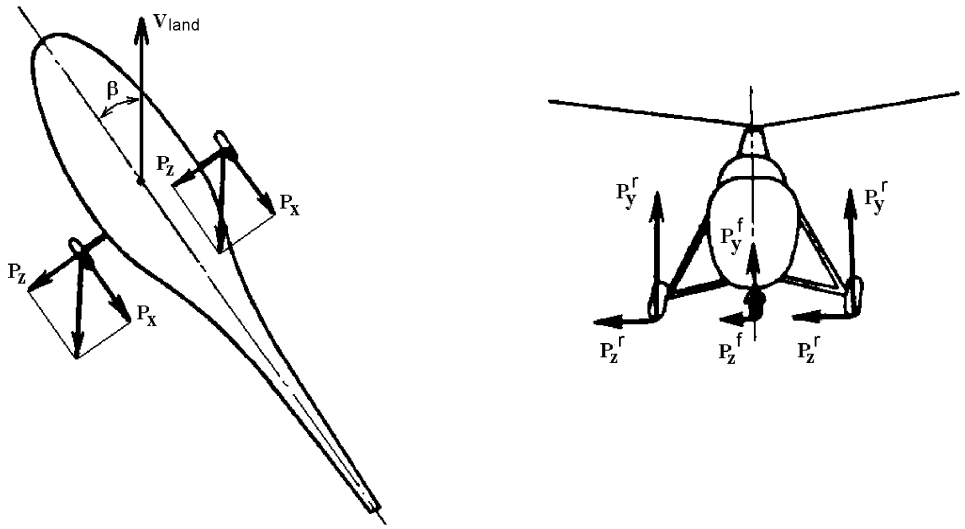


Fig. 10.69. Earth Reactions When Landing with Slip

Here $\sum P_{w.nose}$, $\sum P_{w.rear}$ – total parking load on nose wheels and one rear wheel.

Loads $n_{nose}{}^{op}$ and $n_{rear}{}^{op}$ are determined as it is specified above, for every LG leg in compliance with its reserve energy absorption capacity.

MR thrust $T=0.75T_0$.

Case X-p – Landing with brake application to wheel

In presence of brake wheel let's consider landing with braking.

Rotorcraft is in the parking position with shock struts compressed in accordance with the load acting on the landing gear.

Total thrust developed by rotors $T=0$.

Each leg of the landing gear with brake wheels is under action of forces

$$P^{op}_{y} = 1.5\sum P_{wh}, \quad P^{op}_{x} = \frac{M^{op}_{max}}{R_1},$$

where M^{op}_{max} – maximum braking moment of one leg wheels;

R_1 – wheel radius compressed by force P^{op}_{y};

P_{wh} – wheel parking load.

For castoring wheels the landing gear strength should be sufficient for any direction of the force.

The values of the safety factor in all design case, except *VII-p*, are accepted as follows: for fuselage calculation f = 1.65, for LG calculation f = 1.5. In case *VII-p* you can accept for fuselage calculation f = 1.3, and for LG calculation f = 1.2.

Case of loading in ground conditions

Durability of rotorcraft, its units and mooring equipment is checked for loads acting on them in a variety of conditions of ground operation.

Case 1-z – power application to rotor

Consider the case of a sharp engagement of clutches of main and tail rotors with the engine running. Rotorcraft is in the parking position. The main rotor blades under its own weight hang down freely, based on the lower stop.

Main rotor shaft is under action of the moment

$$M_{MR}^{op} = k_1 \frac{1.5 \cdot 716.2 N_n}{n_{MR}},$$

where $k_1 = \frac{t_1}{t}$ - coefficient of main rotor smooth engagement by coupling clutch: here t_1 – main rotor spin-up time without coupling clutch, t – main rotor spin-up time with coupling clutch, n_{MR} – main rotor number of revolutions in a minute; N_n — engine nominal power, h.p.

The blade is under action of moment relatively to rotor shaft

$$M_{bl}^{op} = \frac{M_{MR}^{op}}{0.9k},$$

where k – main rotor number of blades.

In presence of tail rotor the moment on its shaft

$$M_{TR}^{op} = k_2 \frac{1.5 \cdot 716.2 N_{TR}}{n_{TR}}.$$

Each blade of the tail rotor is under action of the moment

$$M_{blTR}^{op} = \frac{M_{TR}^{op}}{0.9k_{TR}},$$

where N_{TR} – nominal power applied to tail rotor;

n_{TR} – tail rotor number of revolutions in a minute;

k_{TR} – tail rotor number of blades;

$k_2 = \frac{t_2}{t_3}$ – coefficient of tail rotor smooth engagement by coupling clutch (t_2 – tail rotor spin-up time without coupling clutch, t_1 – tail rotor spin-up time with coupling clutch).

Case II-3 – Blade Fall on Vertical Stop

Let's analyze a blade blowing up due to wind gust when MR rotates slowly with subsequent impact of blade against low vertical stop under own weight. Rotorcraft is in parked position.

Thrust and centrifugal forces on MR blades are taken equal zero.

Blade tip height H_{max} above ground is determined by turn of its shank nearby flapping hinge up to upper stop. When blade hits the stop it is under action of bending moment $M_{dyn}{}^{op}=n^{op}M_{st}$ (here M_{st} is a bending moment caused by static action of the blade own weight. In the presence of flapping control then height of blade flapping H is determined by the angle of blade shank turn β_0. Height of blade flapping H above ground is accepted minimum $0.75H_{max}=0.25H_0$ (here H_0 – blade tip flapping height above ground when it is located on the lower stop, m).

Maximum load factor due to blade hit against vertical stop is determined by the formula

$$n^{op} = k_{dyn}\left(1 + \sqrt{1 + \frac{2S_{fl.h}\xi \sin\beta_0}{\int_0^R qy_{st}dr}}\right),$$

where ξ - aerodynamic damping coefficient,

$$\xi = \frac{1 - e^{-0.09\gamma\beta_0}}{0.09\gamma\beta_0} \approx 1 - 0.045\gamma\beta_0;$$

k_{dyn} – dynamic coefficient determined by graph given in Fig. 10.70;

S_{flap} – blade static moment relatively flapping hinge, kgm;

β_0 – maximum angle of blade shank turn from lower vertical stop, rad;

q – blade mass per unit length, kg/m;

$y_{ст}$ – blade deflection at concerned radius, m;

$\gamma = \frac{c_y^{\alpha} b_{07}\rho R^4}{2I_{fl.h}}$ – blade mass characteristic.

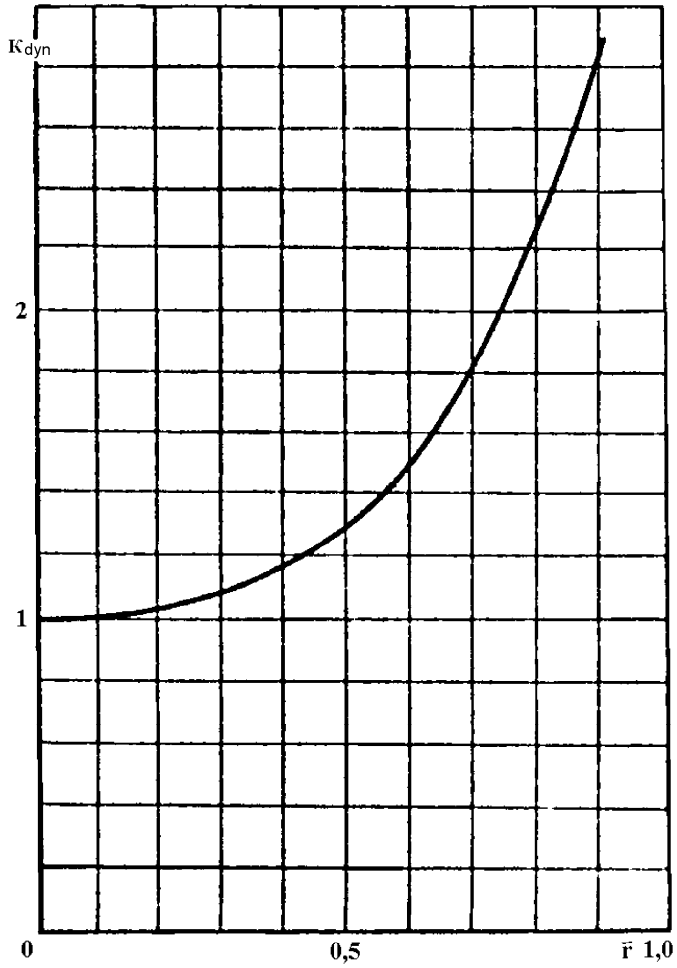


Fig. 10.70. Dynamic Factor

Case III-z – Towing and Taxiing of Rotorcraft

Let's analyze towing and taxiing of rotorcraft over aerodrome (with speed V_{tow} = 10 km/h) or over rough surface (V_{tow} =3 km/h). Rotorcraft is in parked position with tight towing cables. MR blades hang free. There is no MR and TR thrust. It is assumed that towing device is under action of force parallel to the ground surface. This case is intended to provide such selection of shock-absorbing characteristics of the towing device, due to which the loads arising in the rotorcraft units are less then loads in flight and during landing.

1-st variant. Impact with rough by nose LG legs

Every leg of the nose LG is under action of forces

$$P^{op}_{y\,nose} = n^{op}_{nose} \sum P^{op}_{wh.nose},$$

$$P^{op}_{x\,nose} = 0.5 n^{op}_{nose} \sum P^{op}_{wh.nose},$$

$$P^{э}_{z\,nose} = \pm 0.25 \sum P^{op}_{y\,nose}.$$

Nose LG strength for any scheme of towing is checked by the specified loads.

Every leg of the rear LG is under action of forces

$$P^{op}_{y\,rear} = \sum P_{к\,rear},\ P^{op}_{x\,rear} = 0.1\sum P_{wh\,rear}\ .$$

Force in the towing rod for three-wheel LG:

$$T^{op}_{tow} = 2(0.5n^{op}_{nose}\sum P_{wh\,nose} + 0.2\sum P_{wh\,rear})\,.$$

where $P_{wh\,nose}$ and $P_{wh\,rear}$ – total loads on wheels of one leg of nose and rear LG correspondingly.

2-nd variant. Every leg of the nose LG is under action of forces:

$$P^{op}_{y\,nose} = \sum P_{wh\,nose},\ P^{op}_{x\,nose} = 0.1\sum P_{wh\,nose}\ .$$

Every leg of the rear LG is under action of forces:

$$P^{op}_{y\,rear} = n^{op}_{rear}\sum P^{op}_{wh\,rear},\ P^{op}_{x\,rear} = 0.5n^{op}_{rear}\sum P^{op}_{wh\,rear},\ P^{э}_{z\,rear} = \pm 0.25\sum P^{op}_{y\,rear}\,.$$

And moment

$$M^{op}_{y\,rear} = 0.125n^{op}_{rear}\sum P_{wh\,rear}D_{rear},$$

where $D_э$ – wheel diameter, m.

Towing force

$$T^{op}_{tow} = 2(0.5n^{op}_{rear}\sum P_{wh\,rear} + 0.2\sum P_{wh\,nose})$$

Shock strut compression is selected in compliance with loads acting upon legs.

Operational load factors n^{op}_{rear} and n^{op}_{nose} are determined by operational reserve energy absorption capacity A^{op}_{rear} and A^{op}_{nose}.

Case IV-z – squall wind

Let's consider loads of rotorcraft and its mooring equipment created by squall wind.

Rotorcraft is parked with braked wheels, tied with cables or chains to anchors. MR blades are secured with slings. Wind speed W=40 m/s.

Let's consider the wind directions, which develop the highest loads on the rotorcraft, its units or mooring equipment.

10.2.6. Fuselage Strength

For fuselage, cabins, tail boom, and also for windows and doors in flight and ground conditions the loading safety factor is taken to be equal $f = 1.5$.

Under all landing conditions for forces acting upon landing gear the safety factor with regard to operational load factor n^{op} is taken equal to 1.65 under absorption by the

landing gear of operational impact energy A^{op}. Additionally the fuselage static strength should be checked for loads originating during engine-driven landings with forward speed (of airplane type), with safety factor $f = 1.65$.

When determining fuselage strength you should analyze the following basic cases of loading: *I-l, IV-l, VIII-l, I-p - VIII-p, V-z.*

In presence of tail boom the fuselage strength should be additionally checked for cases *IX-l, IV-p, V-p*.

Loads on Pilot Windows and Fuselage Nose Part

Loads on pilot windows and fuselage nose part (navigator cabins) are determined by the following cases of loading: *I-l, II -l, Vl-l* and *lX-l.*

Distribution of aerodynamic load is determined by the wind tunnel tests of the fuselage model in the presence of operating main rotor, and also in cases of flight with and without slip. Velocity head is taken to be equal:

$q=q_{lim}$ – when flying without sleep;

$q=q_{max}$ – when flying with maximum allowable angle of β_{max}. Here $q_{lim}=1.33q_{max}$.

In the absence of data allowing to set lower value the maximum permissible in flight angle of slip at maximum speed is taken equal $\beta_{max}=\pm 15^0$.

Special Requirements to Strength of Cabin and Fuselage in Emergency Landing

Strength of flight compartment, passenger cabin and fuselage, and also holding power of all units and cargo, which displacement in emergency creates hazards for passengers and crew members, should be enough to provide safety in emergency landing of rotorcraft. If landing gear is of retractable type then analyze landing with landing gear being retracted.

In emergency landing the following acceleration components determining operational loads are taken for all body forces:

forward, along vehicle axis – 4 g,

backward, along vehicle axis - 2g,

downward the axis y - 4g,

upward the axis y - 1.5g

sideward along axis z - $\pm$2g,

For structural elements safety factor is taken equal $f= 1.2$.

Under specified conditions of emergency landing it is allowed that some structural elements could be broken if their damage is not dangerous for crew or passengers.

Every individual item that could hurt an onboard parson, and every person onboard remains fixed under action of loads corresponding to the following design inertia loads relative to surrounding structure:

upward – 4;

forward – 16, backward – 4;

sideward – 8;

downward – 20, if the seat movement is provided.

Loads on Seats and Safety Belts

Strength of seats and safety belts is checked as it is described in the above paragraph. With that mass of one passenger is taken equal to 80 kg. For aerial-delivery craft mass of one passenger is taken equal 100 kg.

Loads on Windows and Blisters

Aerodynamic loads on passenger cabin windows and blisters are determined by results of wind tunnel tests of the fuselage model for the above mentioned flight regimes. External positive pressure on the windows got for any case should be increased by $0.3q$, where q – velocity head of this case of loading.

Loads on Doors and Ramps

Doors and ramps are calculated for action of aerodynamic forces determined in the same way as the loads acting upon windows.

If in flight the door or ramp is used as a floor for passengers or crew members then you should consider joint action of aerodynamic load and concentrated force.

$$P^{op}=n^{op}G_{hum}.$$

Value of n^{op} is found by the case *VIII-l*.

With that to determine strength of attachment and load-bearing members of door and ramp you should consider the most adverse variants of directing the force P^{op}.

Doors and corresponding locks of open doors in ground conditions are checked for even distributed load from wind ($W=15$ m/s), kg:

$$P=\pm 15F,$$

where F- area of leaf, m^2.

Safety factor is taken as equal $f=2$.

Strength of ramp used on ground as a bridge for transporting cargo into fuselage cabin is calculated for maximum movable load specified for this rotorcraft. Load factor is taken equal $n^{op}=2$.

Safety factor is taken equal $f=1.5$.

Loading of Cargo Floor

Strength of cargo floor of transport and aerial-delivery rotorcraft is determined for every basic variant of loading provided for the project. For that consider the following cases of loading: *I-l, lV-l, VIII-l, I-p, IV-p* and *VII-p*.

More over the cargo floor structure is checked for loads acting during rolling cargo into the fuselage when loading. With that $n^{op}=2$.

Safety factor is taken equal $f=1.5$.

10.2.7. Strength of Control System and Swashplate Assembly

In absence of boosters the control system part strength is calculated for loads from the pilot side given below. In case if hinge (axial) moments create higher loads on the control system then in calculations use these loads.

Safety factor is taken equal $f = 2$.

If reversible or irreversible booster control is used, then the control linkage section from the control lever (pedals) to the booster is calculated in the same way as in absence of boosters. For remaining section of the control linkage the strength design conditions and safety factors are assumed on the bases of the design and operation experience available.

In presence in the control system of different auxiliary and automatic devices (autopilot, auto stabilizer and auto handling device and others) efforts in the controls should be determined accounting action of these components. With that the most severe conditions of these component actions with regard to strength of individual control elements should be considered.

Pre-estimation of fatigue strength of the control linkage coming from the corresponding control to the booster or in the absence of booster to the link taking variable loads (inertia damper, irreversible control link and others) is executed for loads of case *V-l* for flight regimes at V_{cr} and V_{max}. At that loads are determined either by results of dynamically similar models or the helicopter prototype or in the way of calculations.

During the subsequent estimation of fatigue strength of the same control linkage section besides the most often occurring in operation cases *V-l*, you should account also short-term increase of loads in maneuvering cases of loading in flight and when flying in disturbed air. With that loads are determined by means of measurements in the process of factory flight and ground life tests. Obtained results of measuring loads are reduced to the conditions given by the proper design cases.

Measuring loads acting in the main and tail rotor control systems is performed at maximum disarrangement permissible in operation.

Operational Loads on Controls from Pilot

Below you can find operational loads on control levers, pedals and steering wheel for helicopter having flight weight of 5000 kgf and higher. For helicopters having flight weight up to 1500 kgf the loads indicated are reduced by 30 %. For helicopters having intermediate values of flight weight the loads are determined in the way of linear interpolation between values of operational loads accepted for helicopters having flight weight of 1500 and 5000 kgf.

Loads on Control Levers, Pedals and Steering Wheel

Loads on Main Rotor Control

Force of the control lever in kilos is taken to be equal:

For longitudinal control channel:

P_x^{op} =± 65 (Designer's Manual), ±90 (Airworthiness Requirements), ±45 (AP-29),

For lateral control channel:

P_z^{op} =± 40 (Designer's Manual), ±50 (Airworthiness Requirements), ±30 (AP-29).

In presence of steering wheel the moment on the wheel axis to determine strength of the lateral control channel is taken to be equal in kilos:

$M_{st.w.}^{op}$ = ±20D (Designer's Manual), ±30D (Airworthiness Requirements),

where D — steering wheel diameter, m.

Simultaneous Loading of Longitudinal and Lateral Control Channels

Under simultaneous loading of longitudinal and lateral control channels it is accepted that 75 % of forces acting the control lever correspond to isolate loading of every control channel.

Effort on the collective pitch lever, kg:

$P_{col.p.}^{op}$ =±40 (Designer's Manual), ±60 (Airworthiness Requirements), ±45 (AP-29).

Lodads on Tail Rotor Control (Directional Control)

Effort on one pedal, kg:

P_{ped}^{op}=±60 (Designer's Manual), ±90 (Airworthiness Requirements), ±60 (AP-29).

Force direction is determined by the line connecting the pilot seat center with the pedal pad.

Pedal attachment strength is checked also for simultaneous loading of every pedal with force P_{ped}^{op}.

Loads on Elevator and Movable Stabilizer Control

In presence of elevator and movable stabilizer on the rotorcraft minimal operational load on the control lever in kilos is taken to be equal:

P^{op}=±40 (Designer's Manual), ±75% (Airworthiness Requirements), ±45 (AP-29).

Loads on Rudder Control

In presence of rudder on the rotorcraft minimal operational load on the control lever in kilos is taken to be equal:

P^{op} =±60 (Designer's Manual), ±75% (Airworthiness Requirements), ±60 (AP-29).

Loads on small levers

Minimal operational load on lever is accepted equal, in kilos

P^{op} =±15 (Designer's Manual), ±20 (Airworthiness Requirements), ±20 (AP-29).

Loads on Double Control

Double control system is checked for loads created by both pilots individually and for simultaneous joint action of two pilots as for the same as in opposite sides. Under simultaneous action the operational load from every pilot is accepted to be equal to 75 % of design load.

Hinge Moments of Control Surfaces

Hinge moment coefficients from aerodynamic forces acting upon control surfaces are determined by data of wind tunnel tests of full-scale rotorcraft or its model with main rotor operating.

Hinge moment values are determined by calculations for every classic cases of loading rotorcraft in flight.

Axial (hinge) moments of main and tail rotor blades

Constant and variable parts of axial (hinge) blade moment of main and tail rotors are determined by stresses in the control system, measured in flight during testing the rotorcraft or its prototype and increased by 20%. Measurements are executed at flight regimes corresponding to the cases:

For main rotor - *I-l, II-l, IV-l, V-l*;

For tail rotor - *V-l, XI-l*.

When these data are unavailable it is allowed to use data of wind tunnel tests of dynamically similar rotor models, and also statements given below.

Axial moments created by forces acting upon MR blades in case V-l

According to pre-estimation of the MR control system strength by design data it is allowed to use the following formula to determine amplitudes of axial (hinge) moment acting in level flight with $V_{кр}$ and V_{max}

$$\Delta M_{LG} = \pm 1.3 \cdot 10^{-4} D^4 (1 + 145\mu^3),$$

where D is the MR diameter in meters, and μ is the MR regime of operation.

Frequency of hinge moment variable part ν=n, where n is the MR number of revolutions.

Static part of hinge moment at any flight speed is accepted as not less than the value corresponding to the force on the control lever P = 10 kg.

Axial Moments Acting upon MR Blade in Maneuvering

When calculating axial moments caused by forces acting upon MR blades in cases I-I and IV-I, it is allowed to use the formula

$$\Delta M_{LG} = \pm \frac{0.17 T D^4}{k\omega^2 (GI)_{av}} (1 + 145\mu^3)$$

Here T, μ and ω should correspond to the analyzed case of loading and regime of MR operation. $(GI)_{av}$ — averaged blade torsional stiffness lengthwise, kgm^2.

Static component of the axial moment should correspond to the effort on the control lever, determined according to the paragraph "Control System Loading".

Loads Acting upon Booster Control System

In the present of booster the whole control system is calculated considering efforts from booster by maximum hinge (axial) moments and loads on control lever, pedals or steering wheel in compliance with specified requirements.

Control Linkage Stiffness

Minimal stiffness of the control linkage is determined on safety condition from MR and TR blade flutter and movable control surfaces, and also on condition of absence of dangerous vibrations in the control system.

Loads acting upon swashplate assembly

Strength of swashplate assembly is determined by loads acting upon it from the side of MR blades. Axial moments from blades are calculated according to the described rules.

Dynamic strength of Swashplate Assembly

Dynamic strength of swashplate assembly is subject to testing for loads for cases *I-l, IV-l* and *V-l*.

Dynamic tests of the swashplate assembly are executed for loads corresponding to maximum amplitudes of variable stresses in structure determined in level flight at $V_{кр}$, and corresponding additional static loading. Number of loading cycles is determined by the formula

$$N_1 = 3.6 \cdot 10^3 \eta \nu_1 \tau_1,$$

But not less than 10^7 cycles for steel structure, and not less than $2 \cdot 10^7$ cycles for duralumin and composite structures.

In case when any other flight regime gives stress amplitudes exceeding those at $V_{кр}$, then total structure or its parts should be additionally tested for these variable stresses.

In this case number of loading cycles N_2 is determined from calculation of probable duration τ_2 of rotorcraft operation at this regime and frequency of loading ν_2.

$$N_2 = 10 \cdot 3.6 \cdot 10^3 \eta_N \nu_2 \tau_2 .$$

In case of dynamic tests of swashplate assembly by calculated data the forces in the swashplate rods are calculated by axial (hinge) moments on MR blades.

Dynamic Strength of Main and Tail Rotor Control System

Dynamic strength of all elements of main and tail rotor control system has to be subjected to dynamic testing the weakest and most complicated for strength design elements for loads indicated in this section.

Static Tests and Functioning of Control System

Strength of control system should be proved by the static tests for loads stipulated by this section.

Functioning of the control system is checked during testing the fuselage for cases *I-l* and *IV-l* under operational load.

Structural deformations should not cause additional forces on control levers or pedals.

10.2.8. Tail Unit Strength

For rotorcraft safety factor is taken to be equal $f=1.5$.

Loads Acting Upon Stabilizer

When determining stabilizer strength analyze the following cases of loading: *I-l, III-l, VI-l, VIII-l, IX-l* and *X-l*.

Main Rotor Influence upon Stabilizer Loading

If stabilizer in any regime occurs in the air flow pushed by the main rotor then the stabilizer angle-of-attack changes during every revolution of rotor so many times as the number of blades of main rotor.

Amplitude of the angle-of-attack variable component is detrmined by the formula

$$\Delta\alpha_{st} \pm \frac{1+1.2\mu^2}{24\mu^2}\chi c_{yo}\sigma.$$

Full loading acting upon stabilizer

$$P_{st}^{op} = (\alpha_{st} \pm \Delta\alpha_{st})c_{yst}^{\alpha} qS_{st}.$$

where $\chi=f(\mu, \alpha_{eq})$ – flow angularity factor dictated by main rotor and defined by wind tunnel tests of the rotorcraft model with operating and non-operating rotors.

For approximate calculations it is taken:

$\chi=2.4$;

C_y – generallized coefficient of MR lifting force;

σ – MR solidity ratio;

μ – MR operation characteristic.

Loads acting upon stabilizer due to disturbed air

When flying in disturbed air (case VIII-l) the stabilizer angle-of-attack and loads acting upon it are determined for three regimes: $\mu=0$, μ_{cr} and μ_{max}.

Regime $\mu=0$

Stabilizer angle-of-attack $\alpha_{st}=90°$.

Distribution of aerodynamic load over chord is even, over span is proportional to chords.

Maximum load acting upon stabilizer

$$P_{st}^{op} = 1.2\frac{\rho(V_i + W)^2}{2}S_{st}.$$

Regime $\mu_{кр}$

$$\alpha_{st1} = \alpha_{st} \pm \Delta\alpha_{st} \pm \frac{W}{V_{cr}},$$

where α_{st} – stabilizer angle-of-attack in horizontal flight with $\mu=0.2$;

$\Delta\alpha_{st}$ – value determined according to this paragraph;

V_i – flow speed under main rotor at distance h_{st} (where h_{st} – distance from MR plane of rotation to stabilizer);

W – wind gust effective speed, determined similar to cases *VIII-l, IX-l* and *X-l*.

Variable load acting upon stabilizer is determined by formula

$$P_V^{op} = \pm c_{set}^{\alpha} \Delta\alpha_{st} q_{cr} S_{st}.$$

Static component of load acting upon stabilizer

$$P_{st}^{op} = \pm c_{set}^{\alpha} (\alpha_{st} \pm \frac{W}{V_{cr}}) q_{cr} S_{st},$$

where q_{cr} – velocity head in flight with speed V_{cr}.

Distribution of aerodynamic load over the stabilizer is defined by the stabilizer wind tunnel tests for maximum and minimum value of α_{st}.

Regime μ_{max}

Load is found similar to regime under μ_{cr}, but q_{max} and V_{max} and W=12 m/s are accepted instead of q_{cr} and V_{cr}.

If $\alpha_{st} > \alpha_{cr}$, then load distribution chordwise is taken over trapezoid with the center of pressure located at distance $0.45 b_{stab}$ from the stabilizer leading edge (Fig. 10.71).

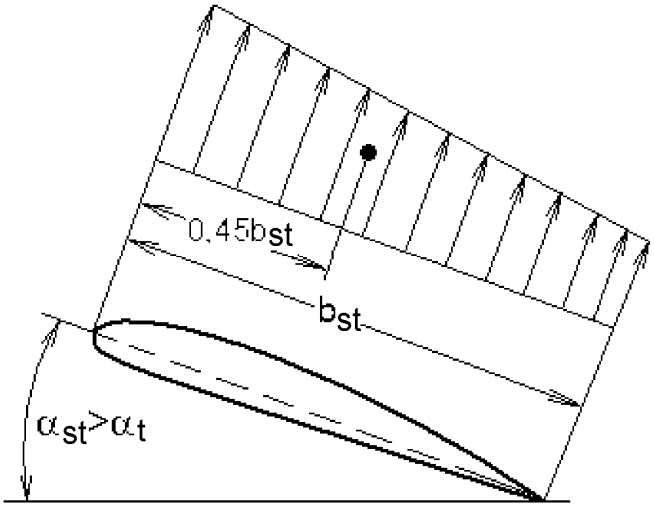


Fig. 10.71. Load Distribution Chordwise

Stabilizer Dynamic Strength

Dynamic strength of stabilizer and its attachment system has to be proved by tests under loads acting in cases *I-l, V-l, IX-l* and *X-l*.

If it could be proved by means of measuring strength in flight or by computational methods that strength variable component in the stabilizer load-bearing members and its fasteners is twice or more lower the material endurance limit of this member then dynamic tests of structure are not obligatory.

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О. Г. Гребеніков, А. М. Гуменний, О. І. Долматов та ін.

ПРОЕКТУВАННЯ ВАЖКИХ ОДНОГВИНТОВИХ ВЕРТОЛЬОТІВ ТА ЇХНІХ ТРАНСМІСІЙ

Частина 1

(Англійською мовою)

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Технічний редактор: Л. О. Кузьменко
Комп'ютерне верстання: О. М. Чмовж, М. В. Кириленко

Зв. план, 2017
Підписано до видання 29.08.2017
Ум. друк. арк. 18.3. Обл.-вид. арк. 20.62. Електронний ресурс

Видавець і виготовлювач
Національний аерокосмічний університет ім. М. Є. Жуковського
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Свідоцтво про внесення суб'єкта видавничої справи
до Державного реєстру видавців, виготовлювачів і розповсюджувачів
видавничої продукції сер. ДК № 391 від 30.03.2001