MINISTRY OF EDUCATION AND SCIENCE OF UKRAINE
National Aviation University

DESIGN OF AVIATION MACHINES
AIRCRAFT AND ROCKET ENGINES

Manual

Second edition, Revised and supplemented

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Викладено теоретичні основи робочого процесу авіаційних двигунів та ракетних двигунів, короткі відомості про особливості конструкцій газотурбінних двигунів різних типів та подано опис конструкції турбогвинтового двигуна ТВ3-117-ВМА-СБМ і його основних функціональних систем.

Для студентів спеціальностей 134 «Авіаційна та ракетно-космічна техніка» та 272 «Авіаційний транспорт».

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The manual includes theoretical base of aircraft and rocket engines working processes, brief information on different types of gas turbine engines structure and turboprop ТВ3-117-ВМА-СБМ engine construction and systems description.

The book is intended for students of Specialties 134 “Aviation and Space Rocket Technology” and 272 “Aviation Transport”.

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THE FOREWORD

The book is a study guide intended for beginners-undergraduates of Aerospace Institute of National Aviation University, who will study the course "The design of aviation machines: aircraft engines". The study guide is written in accordance with the adopted program of mention course.

The book consists of two parts. In the first part (consisting of three chapters) the general information on civil aircraft engines (assigning, principle of operation, classification) is given. Brief information about the features of the working process of basic types of aero-engines is also given in this part of the book.

The second part (consisting of nine chapters) contains the description of main structural parts of the TB3-117BMA-CBM turboprop engine, which is used in the airplane ANTONOV-140. Brief information about basic functional systems of this engine is also given in this part of the book. Short description of each unit construction of the TB3-117BMA-CBM engine will be preceded by a brief explanation of theoretical fundamentals of its operation.

Brief Glossary of basic terms used in the aero-engines theory and design fields is provided in the end of the book. Reference material of physical measurement unites, which are widely used in the foreign technical literature on aircraft, and conversion factors for its recalculation to SI units are given in the Appendix.

While working on the book illustrative materials (photos, pictures, diagrams, etc.) published in the technical literature by the Antonov Aircraft Design Bureau, by the Ivchenko "Progress" Design Bureau and by "Motor Sich" JSC, were widely used. Materials about foreign aircraft engines published by the foreign authors in the books and aeronautical journals, are also used. The list of used publications is given at the end of the book.

The domestic and foreign authors of books on theory of aero-engines use different systems of symbols for designation of gas parameters in the typical cross sections of the air flow inside engine. Therefore in some chapters of the book, the pictures and diagrams are given, where both systems of adopted symbols, mentioned above, are used.

ПЕРЕДМОВА

Пропоноване увазі читачів видання призначене як навчальний посібник для студентів Аерокосмічного інституту Національного авіаційного університету. Воно написане відповідно до затвердженої програми з навчальної дисципліни «Конструкція авіаційної техніки: авіаційні двигуни».

Навчальний посібник складається з двох частин. У першій частині (що містить три розділи) викладено загальні відомості про двигуни, які застосовуються в цивільній авіації (призначення, принцип дії, класифікація двигунів повітряних суден, а також стислі відомості про особливості робочого процесу основних типів авіаційних двигунів різних типів).

У другій частині (що складається з дев'яти розділів) розглянуто конструкцію основних вузлів і загальні відомості про основні функціональні системи конкретного авіаційного турбогвинтового двигуна ТВЗ-117ВМА-СБМ1, який встановлюється на літаку Ан-140. При цьому опису конструкції кожного вузла двигуна передує короткий виклад теоретичних основ його роботи. У кінці посібника наведено глосарій основних термінів з теорії і конструкції авіаційних двигунів, а в додатку — довідковий матеріал про одиниці вимірювання фізичних величин, які використовуються у зарубіжній літературі з авіації.

При роботі над книгою були широко використані матеріали (фотографії, рисунки, схеми тощо), які наведені в технічній літературі, підготовленій Авіаційним науково-технічним комплексом імені О. К. Антонова, Запорізьким моторним конструкторським бюро «Івченко-Прогрес», Запорізьким виробничим об'єднанням (заводом) «Мотор-січ», а також матеріали про іноземні авіадвигуни, опубліковані у книгах і авіаційних журналах англійською мовою.

З огляду на те, що у вітчизняній і зарубіжній літературі з теорії авіаційних двигунів використовуються різні системи символів для позначення параметрів газу в характерних перерізах проточної частини двигунів, у розд. 1, 3, 5 посібника наведено рисунки і схеми, у яких використані обидві системи умовних позначень характерних перерізів газового потоку.

INTRODUCTION

The desire to fly through the air, like a bird, appeared in mankind very long time ago. However, the real way to human flights through the air appeared only at the end of the eighteenth century, when the Montgolfier brothers in France built the first balloons on which they lifted a useful cargo (first useful cargo consisted of a sheep, a duck and a rooster). In November 1783, in the same place, in France, Pilart de Rosier and the Marquis d'Arlande performed their first flight in a hot air balloon (Fig. 1, *a*). Thus began the history of aeronautics, that is, flights of aircraft which are lighter than air.

At the beginning of the nineteenth century, attempts were made in different countries to create aircraft heavier than air without engines (like gliders). However, for controlled flights of apparatus heavier than air, they need a source of energy that would move the aircraft in the air, it means that some engine should be used.

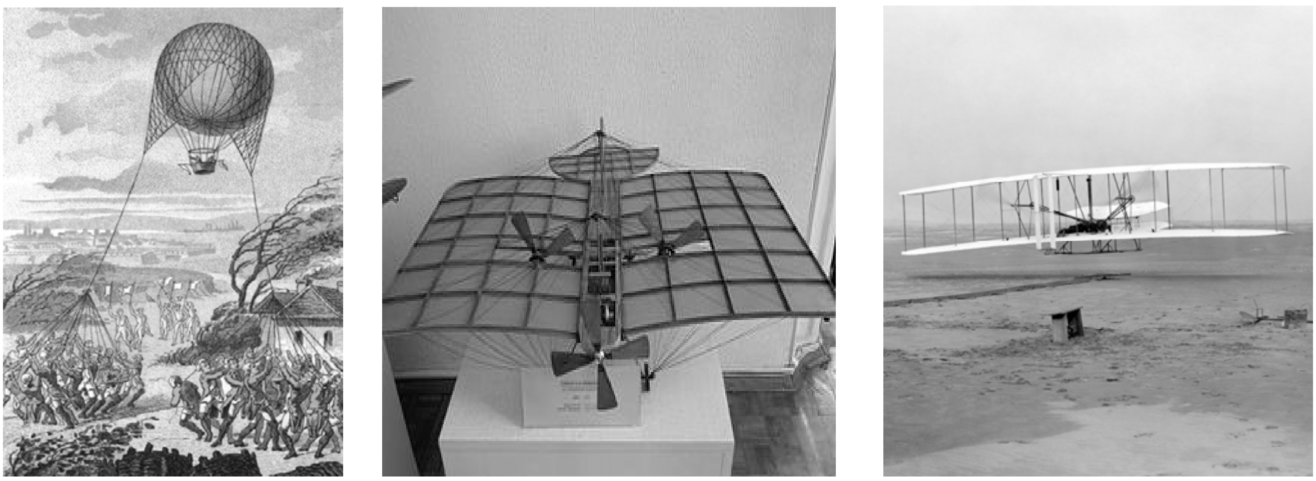
At the end of the nineteenth century (1881) A. F. Mozhaysky was granted a patent and built a plane with a steam engine (Fig. 1, *b*). The Englishman Hiram Maxim made a giant plane in 1894. The machine's two Naptha fired steam engines each produced 180 h.p., and turned two propellers each 5.33 m in diameter. The height of the device was 10.6 meters, wingspan of bearing areas — 31.5 meters, the total surface area — 522 square meters, weight — 2500 kg.

Inventor Clément Ader designed and created an aircraft in France. The main Ader's problem was wrong power-to-weight ratio of its steam engine. Being extremely heavy, the plane could barely raise itself. As steam engines were heavy and cumbersome all efforts to fly with those engines failed.

The first successful flight took place on December 17, 1903. The flight was done on the aircraft with an internal combustion engine. It was designed and built by brothers W. Wright and O. Wright in the United States (Fig. 1, *c*). At about the same time, one after another, planes appeared in Europe, mostly in France (L. Bleriot, F. A. Ferber Santos-Dumont).

The first world war gave a significant impetus in the development of aircraft and in engine design, in particular. However, neither Russia nor Ukraine had their own aircraft engine industry at that time. Only in 1921 the aircraft designer A. D. Shvetsov designed the first domestic aircraft engine. It was the M-11 air-cooled piston engine with a capacity of 100 h. p., and in 1924 this engine started to be mass-produced. The M-11 engine was installed on Po-2 (U-2) and Yak-18 and was used in aviation for over forty years. In the 1950s in the design bureau of the General Designer A. Ivchenko (now it is Zaporozhye Machine-building Design Bureau Progress) the first Ukrainian aircraft engine was created. It was the prominent aircraft air-cooled piston engine AI-14 with a power of 360 hp, which became the basis of a whole family of engines for light aircraft and helicopters.

In the 1920s and 1930s a new direction in the development of aircraft engines appeared in several countries. That direction was jet engine projects. The way from first ideas to real fabricated and tested specimen of this new type of aircraft engine took rather long time, but nowadays just that very type of engines is the most widely used for civil and military aircraft.



a *b* *c*

Fig. 1. Some pages of humane flight idea development history:
a — start of first Montgolfier's balloon; *b* — Mozhaysky's airplane; *c* — Wright Brother's airplane

PART I

BRIEF INFORMATION ON THE THEORY OF GAS TURBINE ENGINE

Chapter 1. CLASSIFICATION OF THE AIRCRAFT ENGINES

Performance characteristics and flight safety of the aircraft depend to a great extend on technical perfection of their powerplant. The powerplant is a main, and often a single, source of power in an aircraft. This power is needed first of all to provide aircraft propulsion. Besides, it is necessary to supply energy to different functional systems of an aircraft, such as hydraulic system, electric system, control system, etc. Even the life-support systems of passenger airliners use compressed and heated air, which is taken from the engines. This air provides comfortable conditions for the crew and passengers.

Modern gas turbine engine is a complicated and expansive machine, which has embodied all new achievements of science and technology. That is why each aviation specialist must be familiar with construction, physical properties and processes, which take place in engines.

In order to master any new subject, one must learn its terminology.

The name “jet engine” which is widely used in everyday speech is a slang. These engines are referred to as gas turbine engines. Nevertheless, the two names are synonymous and will be used interchangeably throughout this book. In order to understand the difference between these terms let us consider general classification of aircraft engines.

All aircraft engines may be divided into two main groups (Fig. 1.1): piston (or reciprocating) engines and jet engines. Both these types are called internal combustion engines as combustion takes place inside the engine, unlike the steam machines where combustion of fuel located outside the machine. Jet engines are divided into three main groups: rocket engines, compressorless air-breathing jet engines (Fig. 1.2) and gas turbine engines (Fig. 1.3). Each type of jet engine has its own preferred sphere of application.

1.1. Piston engines

The piston engines were the first internal combustion engines to be used in airplanes.

The development of aviation powerplants has resulted from utilization of principles employed in the designs of earlier internal combustion engines. During the late nineteenth century, a number of successful engines were designed and built and used to operate machinery and supply power to “horseless carriages”.

As the first powered flight in an airplane was made by the Wright brothers on December 17, 1903, it is safe to say that the first successful engine for an airplane was the gasoline piston engine used in the Wright airplane. This engine was designed and built by the Wright brothers and their mechanic, Charles Taylor.

The engine had the following characteristics: four cylinders, 240 in^3 (cubic inches) [39 329 cm^3], 12 horsepower (hp) [8.94 kilowatts (kW)], weight — 180 pounds (Ib) [82 kg], water cooling.

Since the first piston engine was successfully operated, many different types of engines have been designed. Many have been suitable for the operation of aircraft, and others have been failures. The failures have been the result of poor efficiency, lack of dependability (owing to poor design and to materials which could not withstand the operating conditions), high cost of operation, excessive weight for the power produced, and other deficiencies.

The main requirement which must meet aircraft powerplant engines is high power-to-weight ratio. This was attained first with lightweight piston engines and then, more effectively, with jet engines. Now the piston engines are used for small light airplanes, and jet engines are widely used in other types of flying vehicles.

1.2. Rocket engines

A rocket engine is a heat engine that can be used for spacecraft propulsion. Rocket engines take their reaction mass from one or more tanks and form it into a hypersonic jet, obtaining thrust in accordance with Newton's third law. Rocket engine differs from all the other forms of propulsion in that it does not rely on the air either to provide the oxygen for the combustion of the fuel, or to provide the mass which is thrown out backwards to produce thrust.

The rocket body moves not because the burning gases push against the air, but because the gases exert pressure against the closed end of the rocket body. Since rocket propellants contain their own oxygen supply in

Chapter 5. ENGINE ТВ3-117ВМА-СБМ MAIN DATA AND PERFORMANCE

5.1. Engine Main Data General

— Engine designation ТВ3-117ВМА-СБМ;
— Engine type .... Turboprop;
— Direction of rotor rotation .... Counterclockwise, as viewed from the exhaust unit side;
— Direction of propeller rotation .... Counterclockwise (looking forward);
— Compressor .... Axial, 12 stages;
— Combustion chamber .... Annular with 12 burners and 2 igniter plugs;
— Compressor turbine .... Axial, reaction, 2 stages;
— Power turbine .... Axial, reaction, 2 stages;
— Propeller drive transmission.. Remotely located with rear reduction gear (gear ratio — 1.09), shafting and front reduction gear (gear ratio — 12.11).

Engine overall dimensions:
— length .... 2953 mm;
— height... 1209 mm;
— width .... 988 mm;
— propeller diameter .... 3.720 mm.

Mass of engine at delivery .... 790.5 kg, maximum.

Operating cycle data

(for takeoff rating at M_{fl} = 0; H = 0; t_{amb} = +15 °C):
Pressure ratio... approx. 10;
Gas temperature at turbine entrance.....................................1235 K;
Mass airflow.. approx. 10 kg/s.

Engine data for power ratings

Maximum emergency power at M_{fl} = 0; P_{amb} = 760 mm Hg ; t_{amb} = +37 °C:
Propeller shaft power ....2058 kW;
Propeller shaft torque .... 95,3 %;
Propeller rotational speed ....1203 RPM (100,0 %).

Takeoff power: at M_{fl} = 0; P_{amb} = 760 mm Hg; t_{amb} = +30 °C:
Propeller shaft power ....1837 kW (2500 h.p.);
Propeller shaft torque ....85,0 %;
Propeller rotational speed..1203 RPM (100,0 %).

Takeoff power at M_{fl} = 0; P_{amb} = 760 mm Hg; t_{amb} = +15 °C:
Specific fuel consumption...0,206 kg / e.h.p. · h (0,28 kg / kW · h).

Takeoff power at M_{fl} = 0,5; H = 6000 m; ISA:
Propeller shaft power ....1456 kW (1980 h.p.);
Propeller shaft torque ....67,4 %;
Propeller rotational speed..1203 RPM (100,0 %).

Maximum continuous power at M_{fl} = 0; P_{amb} = 760 mm Hg; t_{amb} = +35 °C:
Propeller shaft power .... 1543,5 kW (2000 h.p.);
Propeller shaft torque .... 78,1 %.

Maximum continuous power at M_{fl} = 0.5; H = 6000 m; ISA:
Propeller shaft power .... 1359,7 kw (1850 h.p.);
Propeller shaft torque .... 68,8 %;
Propeller rotational speed.. 1100 RPM (91,4 %).

Maximum cruising power at M_{fl} = 0.5; H = 6000 m; ISA:
Propeller shaft power .... 1286 kW (1750 h.p.);
Propeller shaft torque .... 65,1 %;
Specific fuel consumption.. 0,193 kg/e.h.p. · h;
Propeller rotational speed.. 1100 RPM (91,4 %).

Flight idle power (Fl) at M_{fl} = 0.2; H = 0 m; ISA:
Propeller shaft power .... (117,6 ± 18) kW (160 ± 25) h.p.;
Propeller shaft torque .... 6 %;
Propeller rotational speed... 1100 RPM (91,4 %), maximum.

Ground idle power (G_I): at M_{fl} = 0; H = 0 m; ISA:
Propeller rotational speed... 1100 RPM (91.4 %), maximum.

Ground idle power — "low-speed taxiing" (Gl-L-ST) at M_{fl} = 0; H = 0 m; ISA:
Propeller rotational speed... 840 to 910 RPM (70,2 to 76,0 %).

Thrust reversal:
Propeller shaft maximum power — not in excess of maximum continuous power;
Propeller rotational speed... 1100 RPM (91,4 %).

Propeller rotational speed for thrust reversal power ratings:
Variable within the range of... 840 to 1203 RPM (70.2 to 100.0 %).

NOTE: The engine main operation parameters are given without taking into account:

- Pressure losses in the inlet section and exhaust unit;
- Loading of the aircraft accessory drives;
- Losses due to engine air bleed for powering the aircraft systems.

Time of engine continuous operation at main power ratings:
Maximum emergency power.. 2.5 min, maximum;
In-flight emergency power (one engagement
during overhaul life)... 60 min, maximum.

Takeoff power:
At takeoff .... 5 min, maximum;
In climb to an altitude above 4000 m
in icing conditions .... 10,5 min, maximum;
Ground idle power and "Low-speed taxiing" ground
idle power, total.. 30 min, maximum;
Maximum reverse thrust .... 3 min, maximum;
Other power ratings.. Unlimited within overhaul life.

NOTES. 1. Maximum emergency power is used only in the event of engine failure at aircraft takeoff and go-around. The engine is set at this power automatically in response to the failure signal of one engine.

2. In-flight emergency power is used only for completion of the flight in the event of engine failure.

3. In abnormal flight conditions specified in the aircraft Flight Manual the in-flight emergency power may be used after 2.5-min operation of the engine at maximum emergency power, with a total time at both power ratings not exceeding 60 min.

4. After using the in-flight emergency power the engine is to be removed from the aircraft and restored by the engine manufacturer or repair organization.

5. Takeoff power may be used continuously up to 10.5 min in climb to an altitude above 4000 m in icing conditions, within the engine total time specified below.

6. Maximum continuous power may be used in cruising flight in adverse flight conditions within the engine total time specified below.

Engine total time at power ratings in percent of overhaul life:
In-flight emergency power .... 60 mm, maximum (one engagement);
Takeoff power .... 5.5 %, maximum;
Maximum continuous power.. 31 %, maximum;
Other power ratings.. Unlimited.

NOTES: 1. Upon using the maximum emergency power make an entry in the engine Log Book on the actual operating time for every engagement of this power setting individually.

Calculate engine total time considering the following relationship:

— One engagement of the maximum emergency power is equivalent to 50 flight cycles;

— 1 s of operation at the maximum emergency power is equivalent to 30 s of operation at take-off power and to 20 mm of the engine total time at all power ratings;

2. Upon using the in-flight emergency power make an entry in the engine Log Book and take into account the actual time of operation at this power rating in calculation of the engine total time.

Time of acceleration: on the ground and in flight from flight idle power to 95 % takeoff power by advancing the throttle control lever within 1 s – 5 s, maximum.

Grade of fuel (main and starting):

— Main .... ТС-1 ГОСТ 10227-86;
— Duplicate .... РТ ГОСТ 10227-86;
— Standby .... Т2 ГОСТ 10227-86;
— Foreign grades.. Type JetA-1 ace. to Specification.

DEF STAN 91-91 (DERD 2494, ASTM D 1655) and mixtures thereof in any proportion.

NOTES: 1. To prevent formation of ice crystals in fuel, the following crystal-formation preventive fluids are added to fuel: "И" ГОСТ 8313-88, "H-M" TV 6-10-1458-79, "S-748" DERD 2451 (Great Britain), MIL-I-27686F(USA), AIR 3652B (France) under the conditions and in quantities specified in the aircraft Maintenance Manual.

2. It is allowed to operate the engine on the above fuels with the use of antistatic additives Sigbol (Сигбол) ТУ 38.101741-78, ASA-3 Shell, Stadis 450 Du Pont Co.

Oil grade:

Main — ИПМ-10 ТУ38.1011299-90 with changes 1, 2; Turbonyc oil 210Ato AIR 3514/A and mixtures thereof in any proportion;

Duplicate — Mobil Turbo 319A-2 to MIL-PRF 7808L Gr 3;
Foreign — AeroShell Turbine Oil 390 DEF TA'N 91-94, Shell;
Castrol AERO 325 DEF STAN 91-94, Castrol;
Exxon Turbo Oil 2389 MIL-PRF-7808L Gr. 3, Exxon.

NOTE: Never mix oil of main and duplicate grades, of duplicate and foreign grades and foreign oil of various grades.

Oil consumption.. 0,4 l/h, maximum.
Engine oil system .... self-contained, circulation type.

Engine starting system .... Air, automatic.

Engine attachment .... Engine is attached in the aircraft in three points:

- Two trunnions on the shafting front casing;
- Trunnion on the rear reduction gear casing.

Quantity of bled air and air bleed conditions:

— for air condition system.. 0,25 kg/s;
— for APU heating .... 0,06 kg/s;
— for heating of underfloor space .... 0,0275 kg/s;
— for aircraft anti icing system .... 0,25 kg/s;
— for anti icing system of air-oil cooller air intake .... 0,025 kg/s;
— for aircraft anti icing system .... 0,25 kg/s.

5.2. Engine Performance

General. Variation of the engine parameters versus the flight altitude (H), ambient air temperature (t_{amb}) and Mach number (M_{fl}) characterizing the indicated airspeed (IAS), is stipulated by the fuel supply laws.

The fuel supply laws have been selected proceeding from the conditions of obtaining the engine parameters, which ensure the required aircraft performance and the reliable operation of the engine. The fuel supply laws are ensured by the engine electronic automatic control system. At the steady engine power ratings, the electronic system forms the following control laws:

- maintaining the propeller power (N_{prop}) proportional the throttle control lever (TCL) angular position;
- limitation of the compressor drive turbine exhaust gas temperature EGT ($t_{comp.\ turb}$) versus TCL angular position. At the engine power ratings from flight idle to maximum cruising power, the limit temperature ($t_{comp.\ turb}$) is of the maximum cruising power rating;
- limitation of physical (measured) value of the gas generator rotor rotation speed (RPM)(n_{GG}) versus the throttle control lever angular position. At the engine power ratings from flight idle to maximum cruising power, the limited value of gas generator rotor rotation speed n_{GG} is equal to of maximum cruising power rating;
- limitation of the maximum corrected gas generator rotor speed $n_{GG\,corr} = n_{GG}\sqrt{\frac{288}{T_{in}^*}}$, where T_{in}^* is the absolute air temperature (K) at the engine inlet; 288K is the absolute air temperature for standard atmosphere under ground conditions;

- limitation of the maximum propeller shaft torque (M_{trq});
- maintaining the propeller rotational speed (n_{prop}) appropriate for each engine power rating.

At the engine power ratings from the takeoff power to the maximum emergency (in-flight emergency) power, inclusive, the propeller rotational speed n_{prop} is 1203 RPM. At the power ratings from the ground idle to the maximum continuous, inclusive, the propeller rotational speed $n_{prop}$is 1100 RPM. At the ground idle at "Low-speed taxiing" the propeller rotational speed n_{prop} is 840 to 910 RPM.

Description of engine performances. There are three different dependencies which are used to characterize an engine performance . These three are named as: throttle performance, altitude-speed performance and climatic performance. Graphs, corresponding these three performances are shown in Fig. 5.1.

Throttle Performance. In general case **t**he throttle performance means variation of main engine parameters (engine power, gas generator rotor speed, compressor outlet air pressure gas temperature at the combustion chamber outlet section or gas temperature at the compressor drive turbine unit exhaust,) versus variation of the fuel flow rate (G_f) through the combustion chamber. When engine operates at the ground and when airplane is unmovable (flight speed V_{fl} = 0) the fuel flow rate (G_f) is determined fully by position of power control lever in cockpit. The throttle performance in ground standard atmospheric conditions (H = 0, M_{fi} = 0,P_{amb} = 101,3 kPa (760 mm Hg), t_{amb}= +15 °C) is presented in Fig. 5.1, *a*. As it is visible in graphs propeller horse power N_{prop} increasing from idle lever up to take-off value is achieved due to very considerable increasing G_f(more than four times).The compressor outlet air pressure (P_c) is increased too approximately two times. This process is accompanied by very considerable increasing the compressor drive turbine exhaust gas temperature ($t_{comp.turb}$) from 425 °C up to 680 °C and gas generator rotor speed (n_{GG}) rising from approximately 80 % at idle operation mode up to 100 % at take-off operation mode.

In its turn, the propeller rotational speed varies versus the engine), within the engine power ratings variation from the ground idle operating mode to the maximum emergency power, as dependence of the fuel flow rate (G_f), power rating, as specified subpart 5.1.

Increase in fuel supply to the combustion chamber results in increase in the power fed to the operating medium, which causes increase in the gas temperature, gas generator rotor speed and compressor outlet air pressure. In this case, the engine power increases. Observed within 0,15 to 0,17 maximum continuous power is a slight variation in the throttle performance as within this range of the power ratings the compressor seventh-stage air bleed valves close (open). The above indicated variation is shown in Fig. 5.1, *a*, for clarity as break in the appropriate lines.

Climatic performance(Engine parameters variation versus ambient air temperature) is dependence of the basic engine parameters at the maximum emergency and takeoff power ratings in the static ground conditions (H = 0, P_{amb} = 101,3 kPa(760 mm Hg), M_{fl} = 0) versus the ambient air temperature is presented in Fig. 5.1, *b*. At t_{amb} > 30 °C for takeoff power and t_{amb} > 37 °C for maximum emergency power the engine operates with the constant turbine exhaust gas temperature ($t^*_{comp.turb}$) corresponding to ($t^*_{comp.turblim}$) for each of the above power ratings. In this case, with increase in the engine inlet air temperature the propeller power decreases. At t_{amb}< 30 °C (t_{amb}< 37 °C) maintained in the engine is the propeller power appropriate for each engine power rating, according to the law, described above. In this case, the fuel flow rate, gas generator rotor speed and compressor turbine exhaust gas temperature decrease, while the compressor outlet air pressure increases with decrease in the engine inlet air temperature.

Variation of the propeller power and the specific equivalent fuel flow rate, with the engine operating at the maximum cruising power at altitude (H = 6000 m; M_{fi} = 0,3, 0,4, 0,5; P_{amb}= P_{ambISA}) versus the ambient air temperature, is similar to shown in Fig. 5.1, *b*.

At flight speeds, corresponding to M_{fi} = 0,3 and M_{fi} = 0,4, at an ambient air temperature below minus 30 °C, the engine automatic control system limits the corrected gas generator rotor speed *(n_{GGcorr})* at a flight speed corresponding to M_{fl} = 0,5, at t_{amb} < –25 °C, maintained is the propeller power (N_{prop}) corresponding to the preset engine power rating. At higher ambient air temperatures and the above indicated flight speeds the engine operates with the compressor turbine exhaust gas temperature ($t^*_{compJuib}$) limited.

Altitude-airspeed performance: The altitude-airspeed performance data of the engine operating at maximum continuous power in standard atmospheric conditions are presented in Fig. 5.1, *c* as the basic engine parameters versus the flight altitude and speed. Up to an altitude of approximately 3500 m, at all presented airspeeds, the engine power is maintained by controlling the propeller power (N_{prop}); at altitudes above 4800 m the engine power is limited by the compressor turbine exhaust gas temperature $t^*_{compturb}$).

At the engine power below or above the maximum continuous, variation of the engine parameters, with change in the altitude-airspeed conditions, is identical to that of the engine operating at the maximum continuous power, except for the engine power ratings and flight conditions at which the maximum physical (measured) or corrected gas generator rotor speed is reached. Reaching the maximum physical or corrected gas generator rotor speed results in the fuel supply procedure which prevents exceeding of this rotor speed.

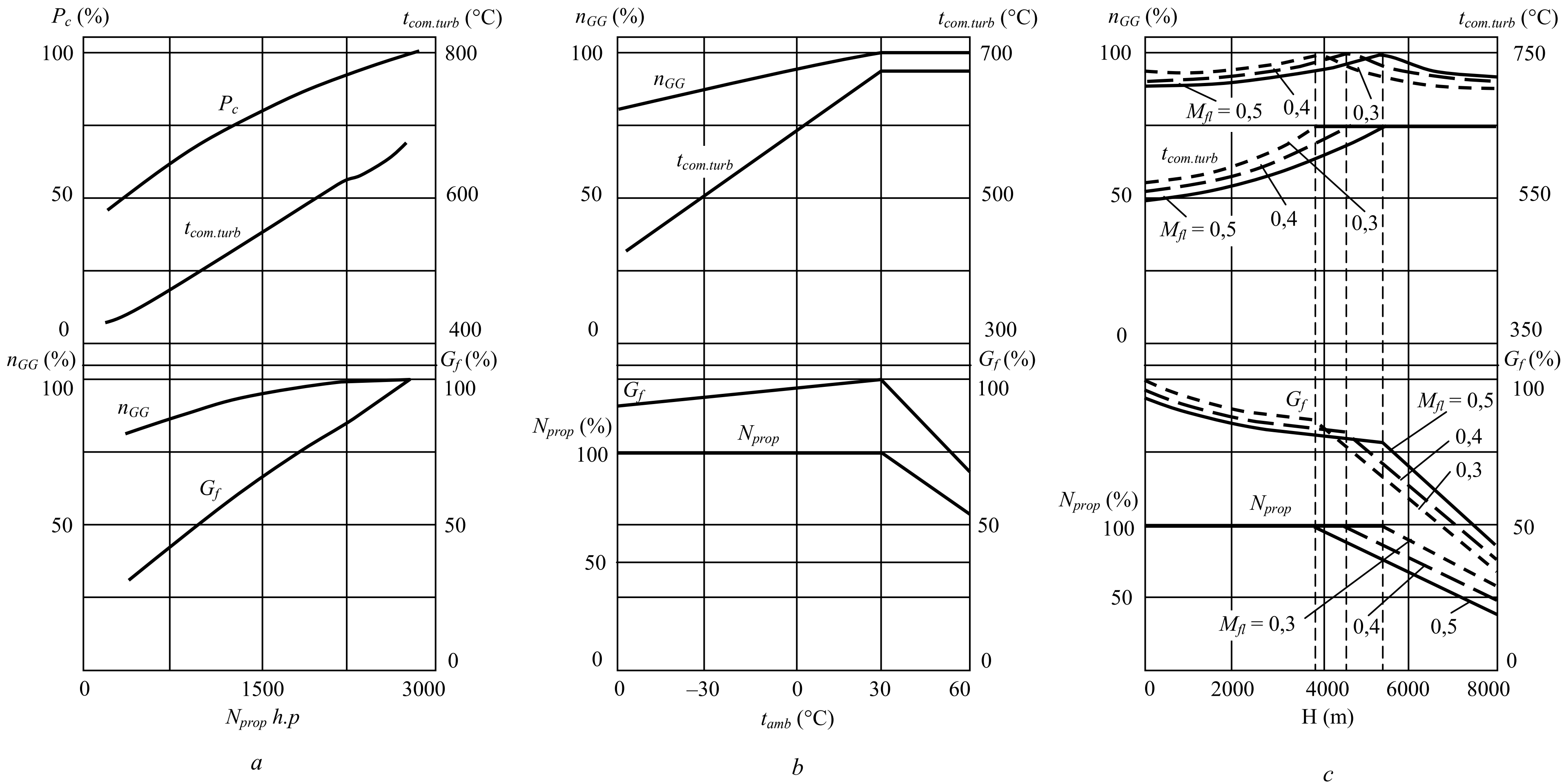


Fig. 5.1. Engine performance:
a — throttle performance at $H = 0$, $M_{flL} = 0$, $t_{amb\ ISA}$, $P_{anb}^{*}{}_{ISA}$; *b* — variation of basic engine parameters at maximum emergency and takeoff power ratings versus variation of ambient air temperature (t_{amb}) at $H = 0$, $M_{fl} = 0$, $t_{amb\ ISA}$, $P_{amb\ ISA}$; *c* — engine altitude-airspeed performance at maximum continuous power versus variation of flight altitude (H) and airspeed (M_{fl}) under standard atmospheric conditions

REVIEW QUESTIONS

1. Name main general data of ТВ3-117ВМА-СБМ turboprop engine.
2. What are the main overall dimensions and mass of ТВ3-117ВМА-СБМ turboprop engine?
3. Call the main operating cycle parameters of a gas turbine engine and note the values of these parameters for ТВ3-117ВМА-СБМ turboprop engine when it operates at $M_{fl} = 0$; $H = 0$; $t_{amb} = +15$ °C condition.
4. What main power ratings are usually given for turboprop engines?
5. Note the value of output power of the ТВ3-117ВМА-СБМ turboprop engine at maximum emergency and takeoff operating modes.
6. What grades of fuel and oil are recommended to be used for the ТВ3-117ВМА-СБМ turboprop engine?
7. What working body is used by the ТВ3-117ВМА-СБМ turboprop engine starting system?
8. What are the purposes of the engine bled air used in aircraft?
9. Name a list of engine performances which are mentioned above.
10. How does the turboprop engine output power change when the flight altitude is increased?
11. Why does the turboprop engine output power change when the flight altitude is increased?

Chapter 6. ENGINE ТВ3-117ВМА-СБМ TRANSMISSION

6.1. Introduction

The propeller drive transmission is intended to drive a propeller at a required rotational speed from a power turbine. The transmission provides for a power turbine RPM -to-propeller RPM gear ratio n_{pt} / n_{prop} = 13.2 : 1. The exterior view of engine transmission is shown in Fig. 6.1 and its longitudinal section view is represented in Fig. 6.2. The propeller drive transmission is remotely located over the turbine engine drive and is used for engine mounting to the aircraft structure. The transmission structure consists of rear reduction gear *9* (Fig. 6.1), shafting *7* and front reduction gear *3*.

6.2. Rear reduction gear description and operation

The rear reduction gear is designed to transmit the torque from the power turbine of the gas-turbine drive to the shafting located above the drive, and to reduce the power turbine rotational speed as well. The reduction gear is made as an ordinary cylindrical gearing having a gear ratio 1.09 : 1.

The rear reduction gear consists of casing *23* (Fig. 6.2), which has a lateral joint in which three spur pinion gears *29*, *26*, *25* are running in ball bearings. Driving gear *29* is connected with splined coupling *30*, the latter connecting the rear reduction gear with the power turbine drive shaft. Intermediate gear *26* engages driving gear *29* and driven gear *25* and, via drive gear *27*, and via accessory drive gearbox *28* gets in mesh with generator accessory drive gearbox *28* gear. Driven gear *25*, via the splined coupling is connected with the drive shaft of the shafting.

The lower section of casing *23* of the rear reduction gear incorporates a turbine drive attached with the use of ball support *19* (Fig. 6.1), the upper section of the casing is provided with a flange for attaching shafting casing *19* (Fig. 6.2).

6.3. Shafting description

The shafting is intended for transmitting the torque from the rear reduction gear to the front one. The shafting consists of casing *19* (Fig. 6.2), which has two lateral joints to house intermediate bearings *17* and *21* of splined couplings *16* and *20* with drive shafts *18* and *22* installed between them.

6.4. Front reduction gear description and operation

The front reduction gear is intended for reducing the shafting rotational speed down to a value required for propeller rotation. The front reduction gear is a two-stage, co-axial differential planetary reduction gear with the gear ratio equal to 12.11 : 1.

The first stage of the reduction gear has driving gear *12* that is connected by means of drive shaft *14* with shafting coupling *16* and is engaged simultaneously with three planetary gears *11* installed in planetary gear casing *31*, which, in their turn are, engaged with rim gear *10*. Planetary gear casing *31* is connected by means of splines with propeller shaft *3*, while rim gear *10* is connected, also by means of splines, with driving gear *32* of the second stage. Driving gear *32* of the second stage is engaged simultaneously with four planetary gears *8* installed in second stage planetary gear casing *33*, which, in their turn, are engaged with second stage rim gear *7* installed on propeller shaft *3* with the help of splined hub *35*.

Planetary gear casing *33* is connected by splines with torquemeter rim *9*. Propeller shaft *3* is installed in the front reduction gear casing *4* on two bearings (front — roller bearing type, and rear — ball bearing. The shaft flange has openings for propeller attachment.

The propeller attachment sealing is provided by rubber sealing ring *1* placed on the centering belt of propeller shaft *3*. The oil cavity of the reduction gear is sealed by annular seal placed in cap *40* and by pressurized air flowing from the compressor five stage via connection on the reduction gear casing.

Installed in the upper section of casing *4* is driven gear 6 of the drive of torquemeter oil pump *5* and propeller speed governor *13*. The driven gear is engaged with driving gear *34* installed on hub *35*.

The oil for controlling the propeller is supplied from propeller speed governor *13* through the inner high-pitch, low-pitch, and pitch-lock passages in reduction gear casing *4* and through oil by-pass assembly *36* into oil by-pass bushing *2* mounted in propeller shaft *3*. Oil by-pass bushing *2* has three co-axial cylindrical cavities arranged in sequence; the cavities are connected with the respective passages of the propeller when mounting it on shaft *3*. Over the circumference of shaft *5* are positioned three pushing rods *39* connected, at the side of propeller, with the propeller pitch control mechanism. The pushing rods, located in the reduction gear, are connected with propeller blade angle sensor via bearing *38* and rack gear train *37* transforming the translational motion of the rods into the rotary motion. The angle sensor measures the angle proportional to the angle of propeller blade turning.

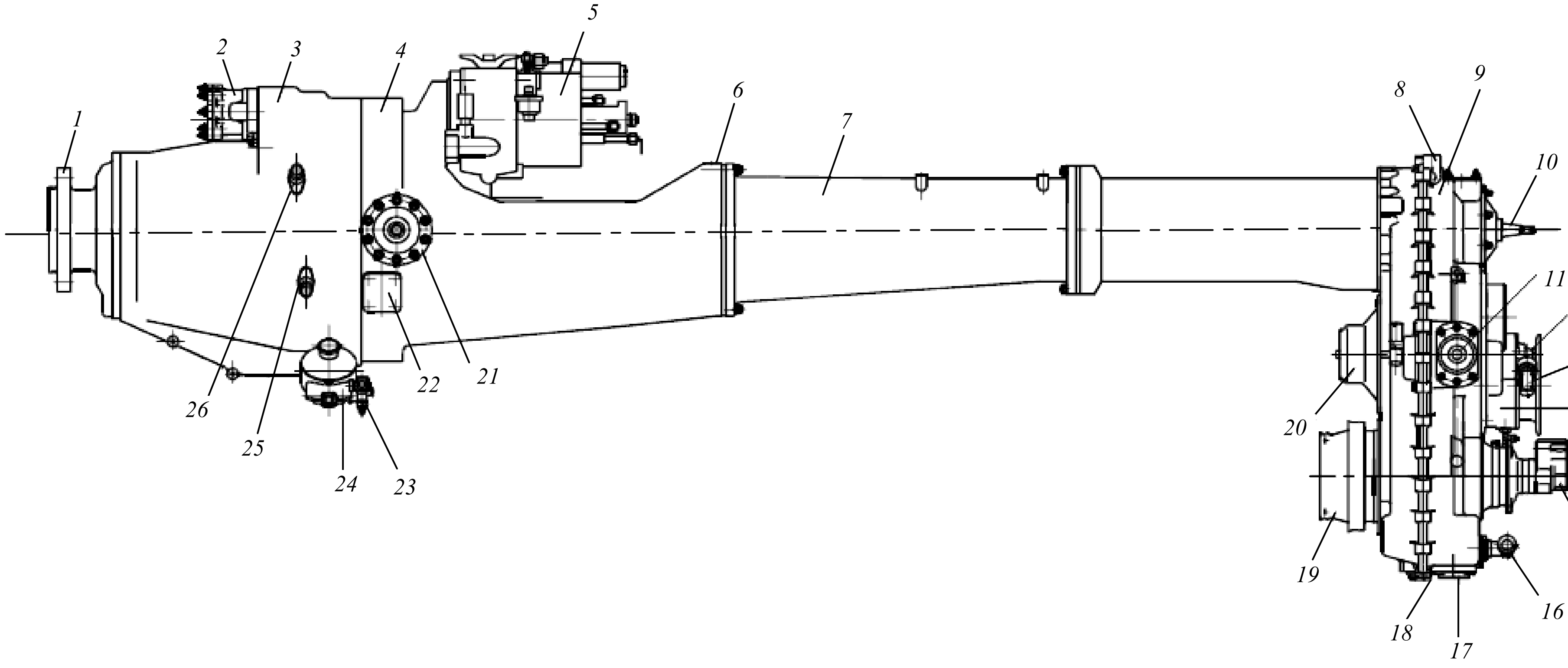


Fig. 6.1. Transmission — left side view:
1 — propeller shaft flange; *2* — torquemeter oil pump; *3* — front reduction gear; *4* — vibration pickup and mounting eye-bolts installation point; *5* — propeller speed governor; *6* — mounting eye-bolts installation point; *7* — shafting; *8* — mounting eye-bolts installation point; *9* — rear reduction gear; *10* — engine attachment rear trunnion; *11* — engine attachment spare trunnion; *12* — generator attachment adapter; *13* — centrimgal breather connection; *14* — accessory drive gearbox; *15* — propeller brake; *16* — rear reduction gear oil scavenge connection; *17* — rear reduction gear chip detector; *18* — reduction gear oil drain plug; *19* — turbine drive ball support; *20* — support cap; *21* — engine attachment front trunnion (both at left and right sides); *22* — turbine drive suspension brackets (both at left and right sides); *23* — front reduction gear oil drain valve; *24* — front reduction gear chip detector pan; *25* — propeller rotational speed sensors installation points; *26* — propeller rotary speed phase sensor installation point

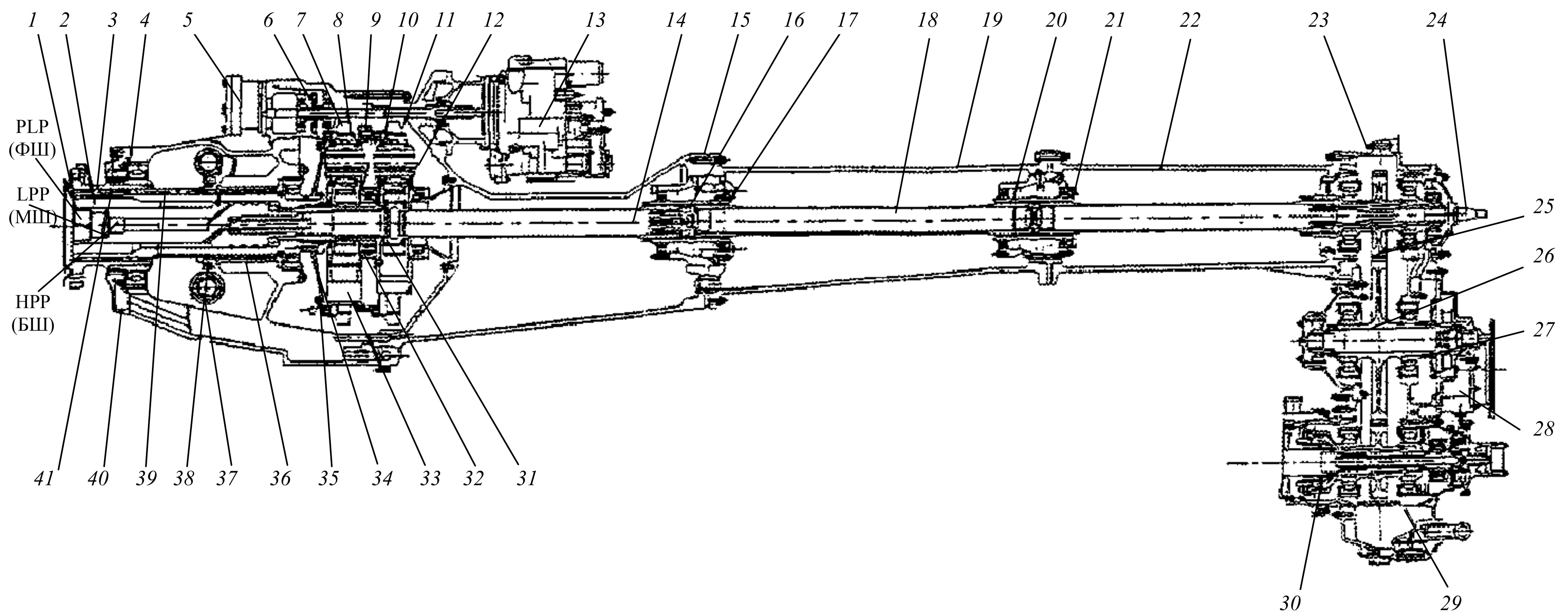


Fig. 6.2. Transmission-longitudinal section view:
1 — propeller sealing ring; *2* — oil by-pass bushing; *3* — propeller shaft; *4* — front reduction gear casing; *5* — torquemeter (TRQMT) oil pump drive;
6 — driven gear; *7* — second stage rim gear; *8* — second stage planetary gear; *9* — torquemeter rim; *10* — first stage rim gear; *11* — first stage planetary gear;
12 — first stage driving gear; *13* — propeller speed governor; *14* — drive shaft; *15* — mounting eye-bolts installation point; *16* — shafting coupling;
17 — shafting intermediate bearing; *18* — drive shaft; *19* — shafting casing; *20* — shafting coupling; *21* — shafting intermediate bearing; *22* — drive shaft;
23 — rear reduction gear casing; *24* — engine attachment rear trunnion; *25* — driven gear; *26* — intermediate gear; *27* — drive gear; *28* — accessory drive gearbox;
29 — rear reduction gear driving gear; *30* — splined coupling; *31* — first stage planetary gear casing; *32* — second stage driving gear; *33* — second stage planetary gear casing;
34 — driving gear; *35* — hub; *36* — oil by-pass assembly; *37* — rack gear train; *38* — bearing; *39* — pushing rod; *40* — seal cap;
41 — propeller anti-icier slip ring bracket; PLP (ФШ) — Propeller pitch lock passage; LPP (МШ) — Low-pitch passage; HPP (БШ) — High-pitch passage

6.5. Torquemeter description

The front reduction gear has a torquemeter. Torquemeter rim *7* (Fig. 6.3) is connected by its eyelets with six hydraulic actuators *6* which, by means of pistons *5* arranged in them, are connected with reduction gear casing *4*. The oil from torquemeter oil pump *1* is forced through the inner passages of reduction gear casing *4* via manifold *3*. The pressure in the torquemeter is measured by oil pressure transmitter *2* installed on front reduction gearbox casing.

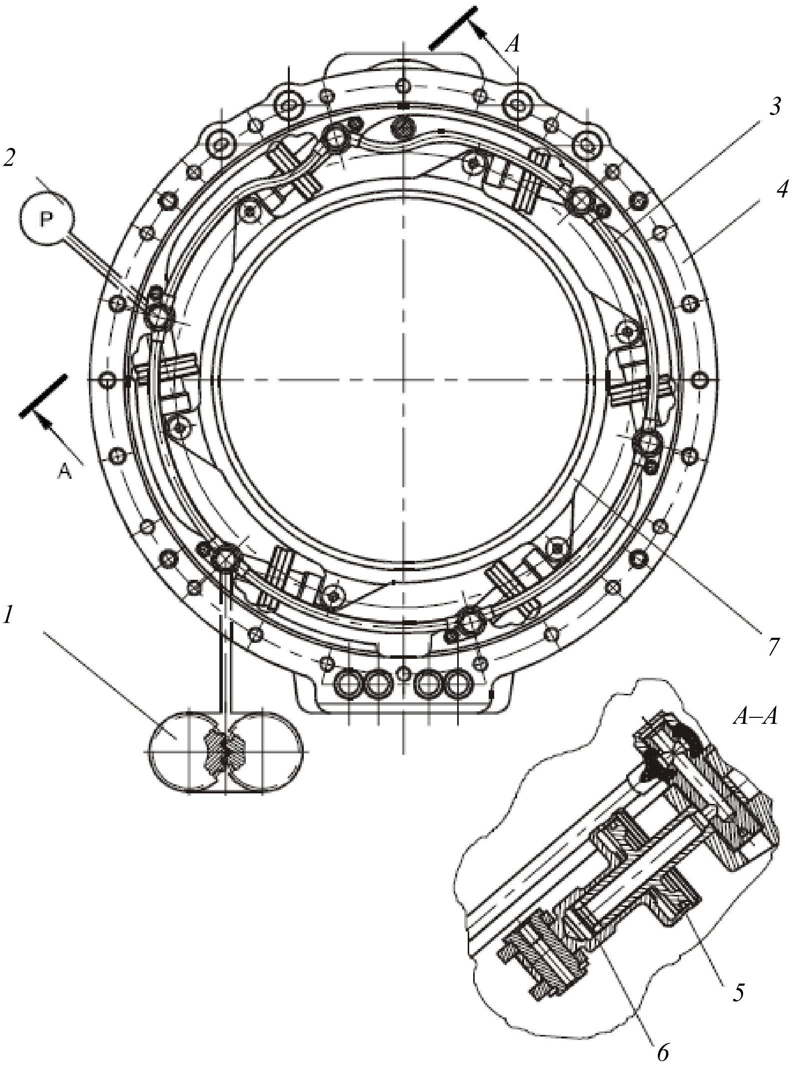


Fig. 6.3. Torquemeter:
1 — torquemeter oil pump; *2* — oil pressure transmitter; *3* — manifold; *4* — reduction gear casing;
5 — piston; *6* — hydraulic actuator; *7* — torquemeter rim

During operation of the reduction gear the torque arising on second stage planetary gear casing *33* (see Fig. 6.2) is transmitted by torquemeter rim *7* (Fig. 6.3) to casing *4* of the reduction gear via hydraulic actuators *7*, where oil pressure is built-up in proportion to the torque being transmitted. Oil pressure transmitter *2* forms an electrical signal which is used by the electronic engine control (EEC) monitor for computing the torque value. The torque value is indicated on the indicator in the flight compartment.

REVIEW QUESTIONS

1. What are the main purposes of the ТВ3-117ВМА-СБМ engine transmission?
2. What main components does the ТВ3-117ВМА-СБМ engine transmission comprise?
3. What is the ratio of power turbine rotor rotational speed to the propeller rotational speed of the ТВ3-117ВМА-СБМ engine transmission?
4. What purpose is the torquemeter used for?
5. Where is the torquemeter mounted?
6. Where is the torquemeter oil pressure transmitter installed?

the form of an oxidizer, they need no outside air in order to burn. For that reason rockets operate most efficiently in a complete vacuum because there is no air resistance to slow them down, and so it can propel aircraft or missiles at great altitudes where the air is very thin and drag is very low. All rocket engines may be subdivided into two basic types (Fig. 1.4): solid-propellant and liquid-propellant rocket engines.

In a liquid rocket, the propellants, the fuel and the oxidizer, are carried in separate tanks. The fuel circulates through the engine's cooling jacket before entering the combustion chamber. This circulation preheats the fuel for combustion and helps cool the rocket.

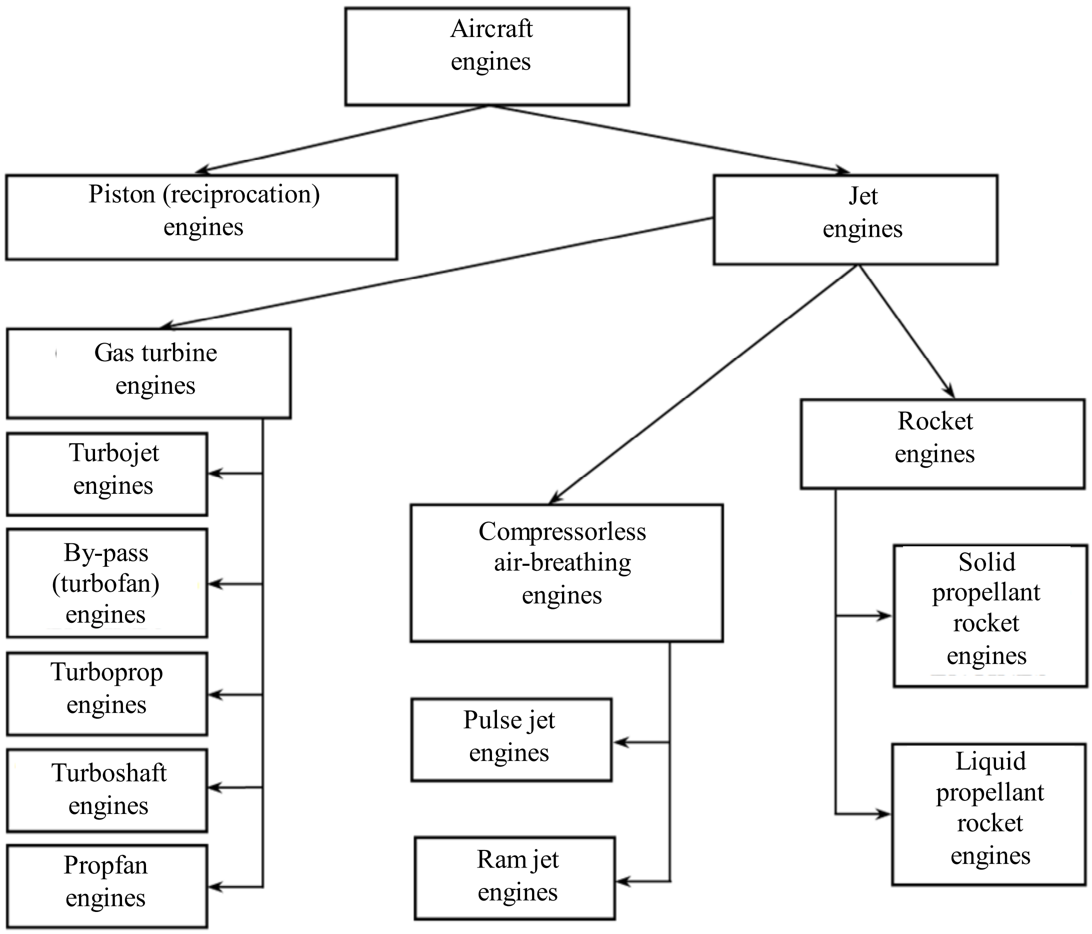


Fig. 1.1. Aircraft engines classification

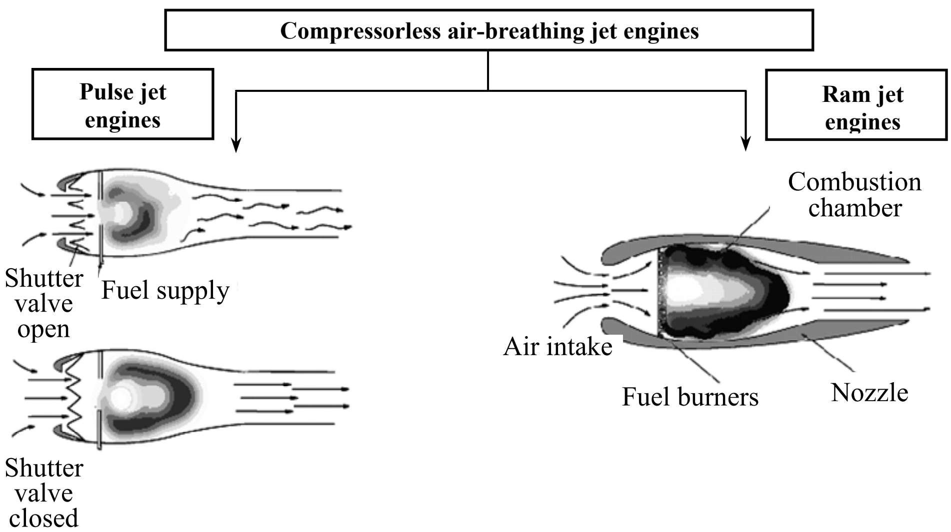


Fig. 1.2. Compressorless jet engines

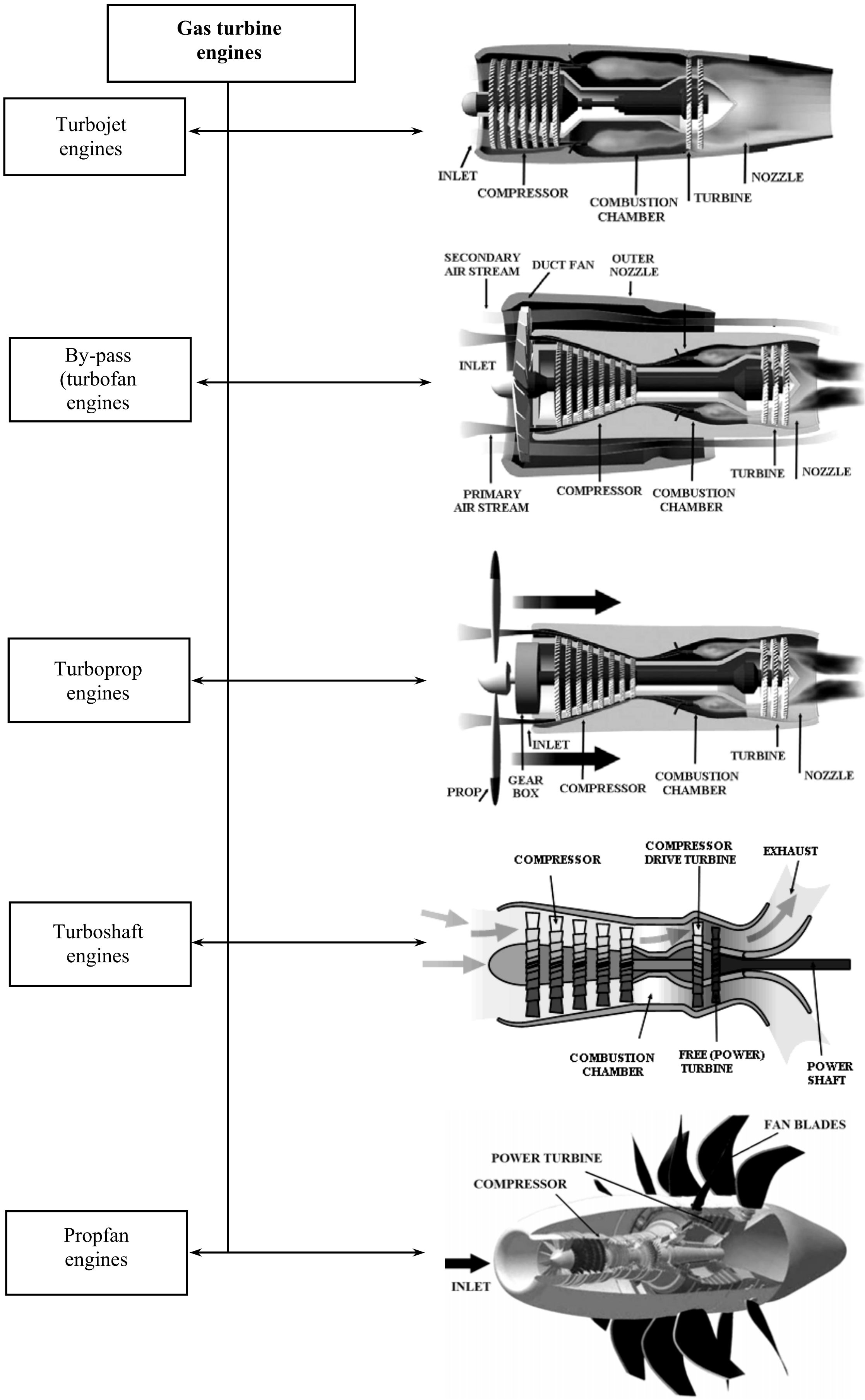


Fig. 1.3. Gas turbine engines classification

In a solid rocket, the propellants are mixed together; cast into a solid mass called the grain and packed into a solid cylinder. The grain, usually formed with a hole down the middle called the perforation, is firmly cemented to the inside of the combustion chamber. After ignition, the grain burns radically outward, and the hot combustion gases pass down the perforation and are exhausted through the nozzle.

Under normal temperature conditions, the propellants do not burn; but they will burn when exposed to a source of heat provided by an igniter. Once the burning starts, it proceeds until all the propellant is exhausted.

The absence of a propellant feed system in the solid-propellant rocket is one of its major advantages. Liquid rockets, on the other hand, may be stopped and later restarted, and their thrust may be varied somewhat by changing the speed of the fuel and oxidizer pumps.

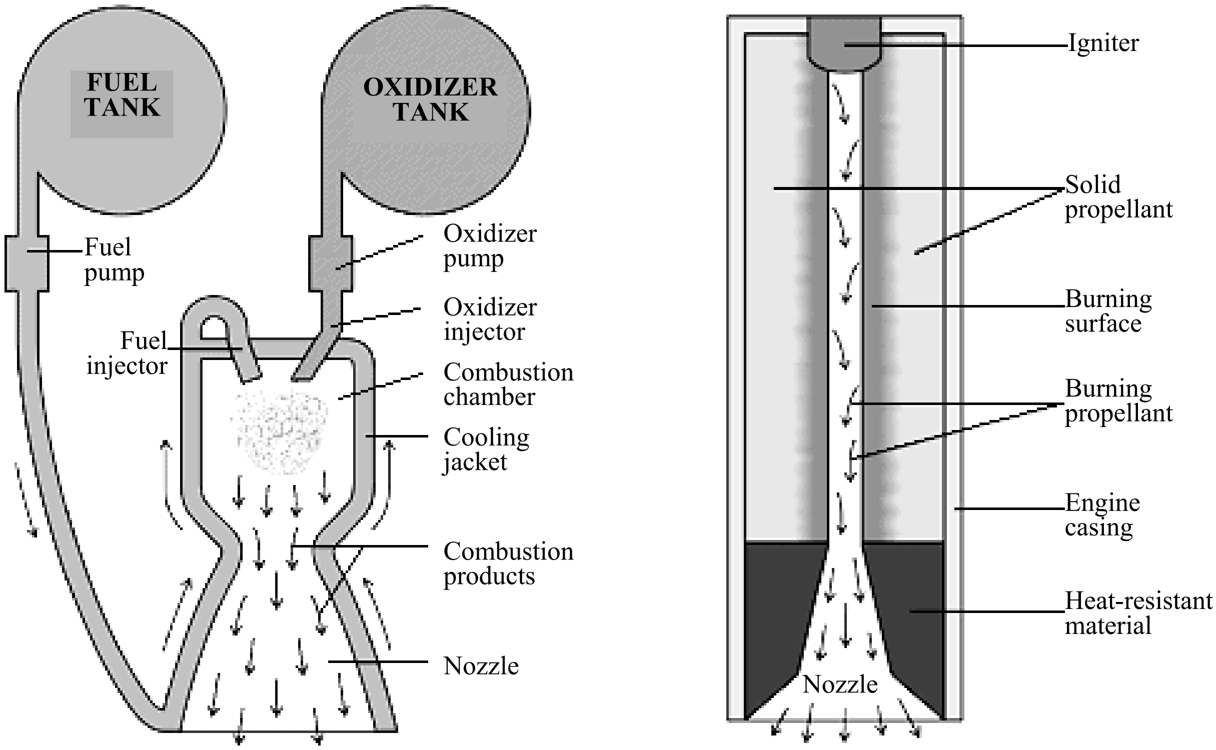


Fig. 1.4. Solid-propellant and liquid-propellant rocket engines

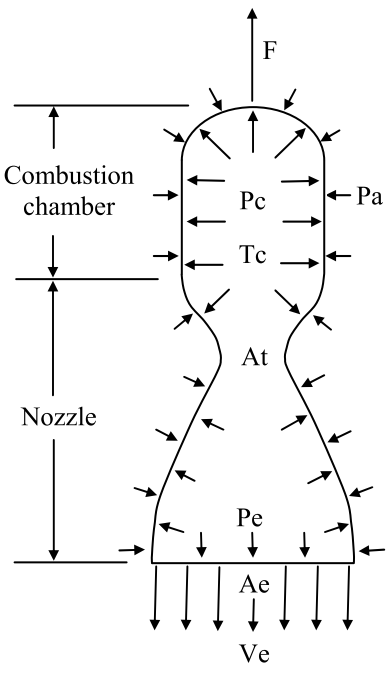


Fig. 1.5. Diagram of rocket engine

The simplest form of solid-propellant rocket is a tube made of metal or even paper, filled with gunpowder or some other sort of rapid burning mixture of chemicals. As the fuel burns, gases are expelled out the back of the tube and the rocket is pushed forward by the gas. This type of rocket dates back to the ancient Chinese who used the rockets for fireworks displays and as weapons.

1.2.1. The generation of thrust by a rocket engine

The function of a chemical rocket engine system is to generate thrust through combustion; i.e., release of thermal energy derived from the chemical energy of the propellants. The generated force (pressure) imparts a momentum to the combustion products. In accordance with the basic laws of motion, a momentum in the opposite direction is also imparted to the vehicle. In practice, high temperature, high-pressure gases are produced in combustion chambers through chemical reactions of either solid or liquid propellants. These gases are ejected through a nozzle at high velocity (Fig. 1.5). The operation of a rocket engine system is independent of its environment except for slight effects on performance caused by ambient air pressure. The rocket is the only practical device able to propel a vehicle in space.

Consider the process of thrust generation. We know that

$$F = ma\,.$$

Force (F) equals mass (m) times acceleration (a). We also know that the velocity (v) increase experienced by the accelerated mass, during the time (t) the force is imparted, is

$$v = at .$$

Combining these two fundamental relations, we obtain

$$F = \frac{m}{t} v .$$

This expression, known as the momentum theorem, is the basic thrust equation for rocket engines. When applied to rocket engines, the term for mass, m, and the term for velocity, v, may apply either to the vehicle or to the ejected gases (Fig. 1.5). The products of v and m, in opposite directions, must be equal, as prescribed by the law of action and reaction. This condition exists even in a "tie down" static rocket firing. In this case, however, the "vehicle mass" (the earth) is so large that reaction effects are undetectable.

Fig. 1.6, *a* shows the situation at time t. The rocket and fuel have a total mass m and this mass is moving with velocity v. At a time Δt later the configuration has changed to that shown in Fig. 1.6, *b*. A mass Δm has been ejected from the rocket and is moving with velocity u as seen by the observer. The rocket is reduced to mass $m - \Delta m$ and the velocity v of the rocket is changed to $v + \Delta v$.

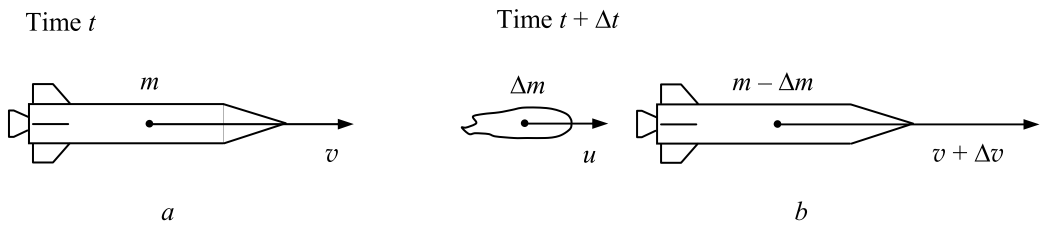


Fig. 1.6. Change mass and velocity at time

The vehicle designer is primarily interested in the utilization of the engine thrust available for the acceleration of the vehicle, which at any point of the trajectory may be expressed as

$$F = m_m \frac{v}{t} = m_m a . \tag{1.1}$$

The vehicle designer uses this equation for vehicle design and trajectory calculations, properly considering that thrust F and vehicle mass m_m change during flight.

In contrast, the engine designer and builder is primarily concerned with the generation of thrust. His attention, therefore, is focused on the efficient conversion of the chemical energy of the propellants into thermal energy, and thus into kinetic energy of the gaseous combustion products. His particular concern is to do this in the most efficient way. For the designer, the basic equation (1.1) may be rewritten as

$$\Sigma F = \frac{m}{t} v_e = \dot{m} v_e ,$$

where $\dot{m}$ is the mass flow rate of the gases, and v_e is their velocity at the nozzle exit. Even in this simple form, it becomes clear that, for a given mass flow rate, thrust will increase with increased gas velocities obtained.

1.2.2. Power cycles

Liquid propellant rocket engines are divided into two categories depending on the type of fuel feeding technology: pressure-fed or pump-fed. The first are the simplest, the latter require additional subsystems such as gas generators or preburners, turbopumps or heat exchangers. Typical pressure-fed engines have a rather lower combustion chamber pressure which limits both the attainable specific impulse and the thrust because they use gas stored in high- pressure tanks (typically 30 MPa helium). Most of the engines used in launchers work with turbomachines to provide the required mass flow rates at the design pressure to the combustion chamber. Pump-fed engines can be distinguished according to the method they use to generate the hot gases that drive the turbines.

Liquid bipropellant rocket engines can be categorized according to their power cycles, that is, how power is derived to feed propellants to the main combustion chamber. Described below are some of the more common types.

Gas-generator cycle: The gas-generator cycle (Fig. 1.7), also called *open cycle*, taps off a small amount of fuel and oxidizer from the main flow (typically 2 to 7 percent) to feed a burner called a gas generator. The hot gas from this generator passes through a turbine to generate power for the pumps that send propellants to the combustion chamber. The hot gas is then either dumped overboard or sent into the main nozzle downstream.

Increasing the flow of propellants into the gas generator increases the speed of the turbine, which increases the flow of propellants into the main combustion chamber, and hence, the amount of thrust produced. The gas generator must burn propellants at a less-than-optimal mixture ratio to keep the temperature low for the turbine blades. Thus, the cycle is appropriate for moderate power requirements but not high-power systems, which would have to divert a large portion of the main flow to the less efficient gas-generator flow.

As in most rocket engines, some of the propellant in a gas generator cycle is used to cool the nozzle and combustion chamber, increasing efficiency and allowing higher engine temperature.

Staged combustion cycle: In a staged combustion cycle (Fig. 1.8), also called *closed cycle*, the propellants are burned in stages. Like the gas-generator cycle, this cycle also has a burner, called a preburner, to generate gas for a turbine. The preburner taps off and burns a small amount of one propellant and a large amount of the other, producing an oxidizer-rich or fuel-rich hot gas mixture that is mostly unburned vaporized propellant. This hot gas is then passed through the turbine, injected into the main chamber, and burned again with the remaining propellants.

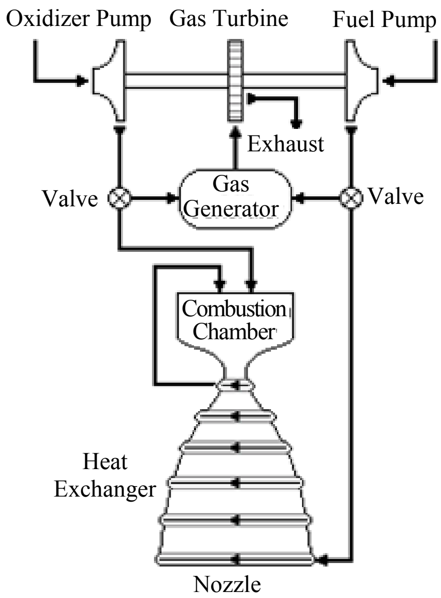


Fig. 1.7. Gas-generator cycle

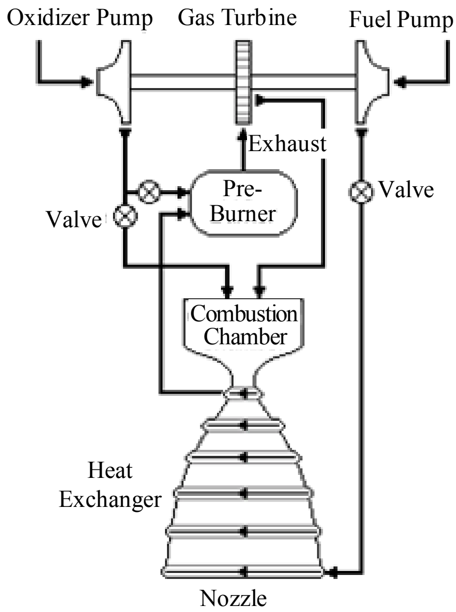


Fig. 1.8. Staged combustion cycle

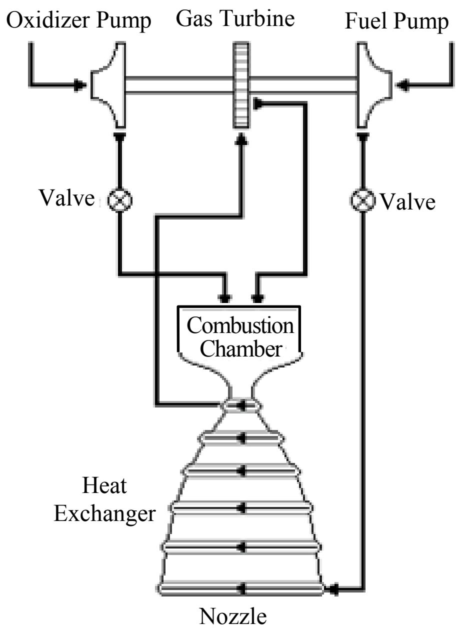


Fig. 1.9. Expander cycle

The advantage over the gas-generator cycle is that all of the propellants are burned at the optimal mixture ratio in the main chamber and no flow is dumped overboard. The staged combustion cycle is often used for high-power applications. The higher the chamber pressure, the smaller and lighter the engine can be to produce the same thrust. Development cost for this cycle is higher because the high pressures complicate the development process. Further disadvantages are harsh turbine conditions, high temperature piping required to carry hot gases, and a very complicated feedback and control design.

Staged combustion was invented by Soviet engineers and first appeared in 1960. In the West, the first laboratory staged combustion test engine was built in Germany in 1963.

Expander cycle: The expander cycle (Fig. 1.9) is similar to the staged combustion cycle but has no preburner. Heat in the cooling jacket of the main combustion chamber serves to vaporize the fuel. The fuel vapor is then passed through the turbine and injected into the main chamber to burn with the oxidizer. This cycle works with fuels such as hydrogen or methane, which have a low boiling point and can be vaporized easily.

As with the staged combustion cycle, all of the propellants are burned at the optimal mixture ratio in the main chamber, and typically no flow is dumped overboard; however, the heat transfer to the fuel limits the power available to the turbine, making this cycle appropriate for small to midsize engines. A variation of the system is the open, or bleed, expander cycle, which uses only a portion of the fuel to drive the turbine. In this variation, the turbine exhaust is dumped overboard to ambient pressure to increase the turbine pressure ratio and power output. This can achieve higher chamber pressures than the closed expander cycle although at lower efficiency because of the overboard flow.

Pressure-fed cycle: the simplest system, the pressure-fed cycle (Fig. 1.10), does not have pumps or turbines but instead relies on tank pressure to feed the propellants into the main chamber. In practice, the cycle is limited to relatively low chamber pressures because higher pressures make the vehicle tanks too heavy. The cycle can be reliable, given its reduced part count and complexity compared with other systems.

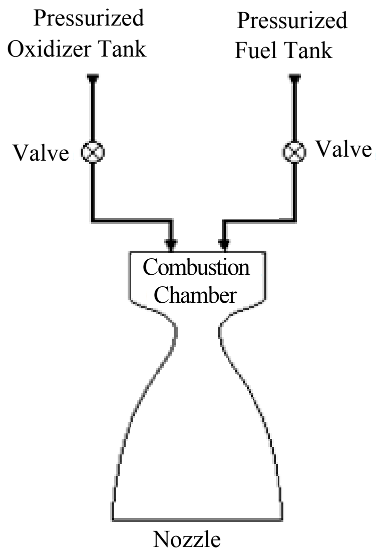


Fig. 1.10.Presure-fed cycle

1.2.3. Liquid rocket propellant

The term “liquid propellant” is used to define both liquid oxidizers (liquid oxygen, liquid fluorine, nitric acid, etc.) and liquid fuels (RP-1, alcohol, liquid hydrogen, etc.). In some cases additives are used (water, ferric chloride, etc.). The propellants furnish the energy and the working substance for the rocket engines. The selection of the propellants is one of the most important steps in the design of an engine. It greatly affects overall engine system performance as well as the design criteria for each engine component.

A good liquid propellant is one with a high specific impulse or, stated another way, one with a high speed of exhaust gas ejection. This implies a high combustion temperature and exhaust gases with small molecular weights. However, there is another important factor that must be taken into consideration: the density of the propellant. Using low-density propellants means that larger storage tanks will be required, thus increasing the mass of the launch vehicle. Storage temperature is also important. A propellant with a low storage temperature, i.e. a cryogenic, will require thermal insulation, thus further increasing the mass of the launcher. The toxicity of the propellant is likewise important. Safety hazards exist when handling, transporting, and storing highly toxic compounds. Also, some propellants are very corrosive; however, materials that are resistant to certain propellants have been identified for use in rocket construction.

Monopropellants

Liquid monopropellants may be either a mixture of oxidizer and combustible matter, or a single compound which can be decomposed with attendant heat release and gasification. A rocket monopropellant must be stable in a natural or controlled environment, yet should produce hot combustion or decomposition gases when pressurized, heated, or fed through a catalyst. A liquid monopropellant engine system usually does have the advantage of simplicity of tank age, feed plumbing, flow control, and injection. Unfortunately, most of the practical monopropellants, such as hydrogen peroxide (H_2O_2), have a relatively low performance. Thus, they are mainly used as secondary power sources in rocket engine systems, such as for turbopump gas generators and auxiliary power drives, and for attitude and roll control jets. Certain high-performance monopropellants, such as methyl nitrate (CH_3NO_3), are rather unstable and are considered unsafe for rocket applications.

Bipropellants

In a liquid bipropellant system, two different propellants are used, usually an oxidizer and a fuel. Separate tanks hold oxidizer and fuel which are not mixed until they reach the combustion chamber. Bipropellants offer higher performance, combined with safer operation. The combustion of many bipropellant combinations is initiated by ignition devices such as: a) chemical pyrotechnic igniters; b) electric spark plugs; c) injection of a spontaneously ignitable liquid fuel or oxidizer (“Pyrophoric fluid”) ahead of the propellant proper; d) small combustor where ignition is started.

Other bipropellant combinations ignite spontaneously upon mixing. Those combinations are defined as hypergolic and permit greatly simplified ignition, but pose certain hazards. For instance, accidental mixing of the fuel and oxidizer due to tank and other hardware failures could cause a violent explosion. These hazards must be considered when designing an engine system using hypergolic propellants.

Cryogenic Propellants

Some liquid propellants are liquefied gases with a very low boiling point (127 K to 20 K) at ambient pressure and a low critical temperature (260 K to 30 K). These propellants are defined as cryogenics. The most common cryogenic propellants for rocket applications are liquid oxygen (O_2), liquid hydrogen (H_2), liquid fluorine (F_2) and oxygen difluoride (OF_2) or mixtures of some of them. Cryogenic propellants pose storage and handling problems. Elaborate insulation must be provided in order to minimize losses due to boil-off (any vapor loss from the oxidizer or fuel in a rocket during countdown), the complexity depending on storage period and type of cryogenic. Recently, novel insulating techniques have been under development which should greatly reduce these losses. Adequate venting systems are needed for the developed gases. Storage and handling equipment and their components are extremely sensitive to atmospheric or other moisture; even minute quantities may cause a jamming of, for instance, a valve. Likewise, the design criteria, including materials selection for engine systems using cryogenic propellants, must consider the very low temperatures involved.

Storable Liquid Propellants

In contrast to the cryogenic propellants, certain other liquid propellants are stable over a reasonable range of temperature and pressure, and are sufficiently nonreactive with construction materials to permit storage in closed containers for periods of a year or more. These propellants are defined as storables. Storable liquid propellants permit almost instant readiness of the rocket engine and may result in greater reliability due to the absence of extremely low temperatures and the need to dispose of boil off vapors. Their application to military vehicles as well as to the upper stages of space vehicles has increased significantly during recent years.

Additives for Liquid Rocket Propellants

Sometimes additives are mixed into liquid propellants to improve cooling characteristics, to depress freezing point, to reduce corrosive effects, to facilitate ignition, and to stabilize combustion.

PROPERTIES OF ROCKET PROPELLANTS

Compound	Chemical Formula	Density	Melting Point	Boiling Point
Liquid Oxygen	O_2	1.14 g/ml	–218.8 °C	–183.0 °C
Liquid Fluorine	F_2	1.50 g/ml	–219.6 °C	–188.1 °C
Nitrogen Tetroxide	N_2O_4	1.45 g/ml	–9.3 °C	21.15 °C
Nitric Acid	HNO_3	1.55 g/ml	–41.6 °C	83 °C
Hydrogen Peroxide	H_2O_2	1.44 g/ml	–0.4 °C	150.2 °C
Nitrous Oxide	N_2O	1.22 g/ml	–90.8 °C	–88.5 °C
Chlorine Pentafluoride	ClF_5	1.9 g/ml	–103 °C	–13.1 °C
Ammonium Perchlorate	NH_4ClO_4	1.95 g/ml	240 °C	N/A
Liquid Hydrogen	H_2	0.071 g/ml	–259.3 °C	–252.9 °C
Liquid Methane	CH_4	0.423 g/ml	–182.5 °C	–161.6 °C
Ethyl Alcohol	C_2H_5OH	0.789 g/ml	–114.1 °C	78.2 °C
n-Dodecane (Kerosene)	$C_{12}H_{26}$	0.749 g/ml	–9.6 °C	216.3 °C
RP-1	$C_nH_{1.953n}$	0.820 g/ml	N/A	177–274 °C
Hydrazine	N_2H_4	1.004 g/ml	1.4 °C	113.5 °C
Methyl Hydrazine	CH_3NHNH_2	0.866 g/ml	–52.4 °C	87.5 °C
Dimethyl Hydrazine	$(CH_3)_2NNH_2$	0.791 g/ml	–58 °C	63.9 °C
Aluminum	Al	2.70 g/ml	660.4 °C	2467 °C
Polybutadiene	$(C_4H_6)_n$	≈0.93 g/ml	N/A	N/A

Fuel combinations used in rocket engines

Oxidizer	Fuel
Liquid Oxygen	Liquid Hydrogen
	Liquid Methane
	Ethanol + 25 % water
	Kerosene
	Hydrazine
	MMH
	UDMH
	50–50
Nitrogen Tetroxide	Kerosene
	Hydrazine
	MMH
	UDMH
	50–50

Oxidizer	Fuel
Liquid Fluorine	Liquid Hydrogen
	Hydrazine
Red-Fuming Nitric Acid (14 % N_2O_4)	Kerosene
	Hydrazine
	MMH
	UDMH
	50–50
Hydrogen Peroxide (85 % concentration)	Kerosene
	Hydrazine
Chlorine Pentafluoride	Hydrazine
FLOX-70	Kerosene

Propellants advantages and disadvantages

Propellants	Advantages	Disadvantages
LOX/LH2	High performance, commonality with SLS (Space Launch System) upper stage	Hydrogen boil-off, low fuel density
LOX/LCH4	Low boil-off fuel and oxidizer, good fuel density	Few heritage engines, low performance (compared to LOX/LH2)
LOX/RP	Storable fuel, low boil-off oxidizer, heritage engines, great fuel density	Low performance (compared to LOX/LH2 or LOX/LCH4)
NTO/MMH	Storable propellants, great fuel and oxidizer densities	Lowest performance of all options, toxicity necessitates special handling

Note: Propellant combinations considered are: liquid oxygen and liquid hydrogen (LOX/LH2), liquid oxygen and liquid methane (LOX/LCH4), liquid oxygen and kerosene (LOX/RP), and nitrogen tetroxide and hydrazine (NTO/MMH)

1.2.4. The pre-history of spaceflight

The pre-history of spaceflight — that is, before the launch of Sputnik inaugurated the Space Age in October 1957-features many well-known individuals: Sergei Korolev and Wernher von Braun, working on rockets; before them, Robert Goddard and Hermann Oberth with their early designs and tests; and before them, Konstantin Tsiolkovsky and his famous rocket equation.

The names of the pioneers are well known: Tsiolkovsky, Goddard, Oberth, von Braun, Korolev, Glushko, Chelomei, Some were practical engineers, some were mathematicians and others were dreamers. There are two strands. The first is the imaginative concept of the rocket as a vehicle for gaining access to space, and the second is the practical development of the rocket.

Konstantin Tsiolkovsky (1857–1935), a mathematics teacher, wrote about space travel, including weightlessness and escape velocity, in 1883, and he wrote about artificial satellites in 1895. In a paper published in 1903 he derived the rocket equation, and dealt in detail with the use of rocket propulsion for space travel; and in 1924 he described multi-stage rockets. His writings on space travel were based on mathematics.

He laid the mathematical foundations of spaceflight. K. Tsiolkovsky never experimented with rockets; his work was almost purely theoretical. He identified exhaust velocity as the important performance parameter; he realized that the higher temperature and lower molecular weight produced by liquid fuels would be important for achieving high exhaust velocity; and he identified liquid oxygen and hydrogen as suitable propellants for space rockets. He also invented the multi-stage rocket.

Tsiolkovsky's counterpart in the German-speaking world was the Rumanian **Herman Oberth** (1894–1992). Herman Oberth published his doctoral thesis in 1923, as a book in which he examined the use of rockets for space travel, including the design of liquid-fuelled engines using alcohol and liquid oxygen. His analysis was again mathematical, and he himself had not carried out any rocket experiments at the time.

Robert Goddard (1882–1945), a professor of physics at Clark University in Massachusetts, was, as early as 1914, granted patents for the design of liquid-fuelled rocket combustion chambers and nozzles. In 1919 he published a research on rocket vehicles called A Method of Reaching Extreme Altitudes, which contained not only the theory of rocket vehicles, but also detailed designs and test results from his own experiments. He was eventually granted 214 patents on rocket apparatus.

Goddard's inventions included the use of gyroscopes for guidance, the use of vanes in the jet stream to steer the rocket, the use of valves in the propellant lines to stop and start the engine, the use of turbo-pumps to deliver the propellant to the combustion chamber, and the use of liquid oxygen to cool the exhaust nozzle, all of which were crucial to the development of the modern rocket. He launched his first liquid-fuelled rocket from Auburn, Massachusetts, on 16 March 1926. It weighed 5 kg, was powered by liquid oxygen and petrol, and it reached a height of 12.5 meters. At the end of his 1919 paper Goddard had mentioned the possibility of sending an unmanned rocket to the Moon, and for this he was ridiculed by the Press. Because of his rocket experiments he was later thrown out of Massachusetts by the fire officer, but he continued his work until 1940, launching his rockets in New Mexico. In I960 the US government bought his patents for two million dollars.

Wernher von Braun (1912–1977) was one of the most important rocket developers and champions of space exploration during the period between the 1930s and the 1970s. Von Braun was one of the enthusiastic engineers who joined the military research station at Peenemünde. Von Braun is well known as the leader of what has been called the “rocket team” which developed the V-2, technical name A4, ballistic missile for the Nazis during World War II.

When the war ended in 1945, von Braun and most of his Peenemunde colleagues surrendered to the U.S. military. In 1960, von Braun and his team left the employ of the Army to join the newly formed National

Aeronautics and Space Administration (NASA). Serving as director of NASA's Marshall Space Flight Center in Alabama, von Braun oversaw the development of the “Saturn” I, IB, and V. The “Saturn” V rocket lifted all of the Apollo lunar missions into space.

The USSR space programmed has been the most active and focused in history: the first artificial satellite, the first man in space, the first spacecraft on the Moon, the first docking of two spacecraft, and the first space station. All of these are the achievements of the Soviet Union. In the period from 1957 to 1959, three satellites and two successful lunar probes had been launched by the USSR. In 1961, Yuri Gagarin became the first man in space, and at the same time, several fly-bys of Mars and Venus were accomplished. In all there were 12 successful the Soviet Union lunar probes launched before the American “Saturn V”. Apart from the drive and vision of the Soviet engineers — particularly Sergei Korolev — the reason for this success lay in the fact that the Soviet rockets were more powerful, and were better designed. The “Vostok”, “Soyuz”, and “Molniya” rockets were the brainchildren of S. Korolev and V. Glushko.

As the epitome of the practical engineer, **Sergei Korolev** (1906–1966) and his colleague **Valentin Glushko** (1908–1989) should be credited with much of the Soviet success. V. Glushko was the engine designer, and S. Korolev was the rocket designer.

Valentin Glushko was the preeminent Soviet rocket engine designer of the 20th century and, from 1974 until his death, the head of NPO “Energiya” — de facto head of the Soviet space program.

V. Glushko was put in charge of building the Russian version of the V-2 engine, the RD-100. This led to the RD-105 and RD-106 (neither of which was very successful) and then, from the standpoint of history the most important, the RD-107. Every manned Soviet spacecraft has been pushed into orbit by Glushko’s RD-107s or a derivative of it.

Glushko's engines, the RD 100, 200 and 300 series, were and still are used in launchers. It is significant that the 100 series, using liquid oxygen and alcohol, was a Soviet replacement for the A4 engine. Soviet rocket engines for new launchers are being made under licence in the United States. The desire to use a purely Soviet engine was already strong. In fact, in the 1930s V. Glushko had developed liquid-fuelled engines which used regenerative cooling, turbo-pumps, and throttles.

Vladimir Chelomei (1914–1984) was perpetually the second-most important designer of Soviet spacecraft and rockets, yet by the end of his life his Proton rocket had become a workhorse of the Soviet (and later Russian) space program and he was the godfather USSR’s ultimate achievement, the space station “Mir”.

His the Experimental Design “Bureau” number 52, or OKB-52, worked closely with V. Glushko to produce heavy-lift vehicles that competed with Korolev's R-7 and N-l designs. One of those was the successful UR-500 Proton heavy-lift launcher originally built as a military launcher for large surveillance satellites and manned observation platforms.

With time and experience, however, progress was made. But the modern rocket engines which propel some missiles are similar to the first rockets. The solid fuel booster rockets mounted on the sides of the Space Shuttle are of this type. On the Shuttle, each rocket engine is over 140 ft [42.67 m] long and able to exert a thrust of over 3 000 000 lb [13 350 kN]. The small solid fuel booster rocket engines are used for acceleration of military aircraft during starting.

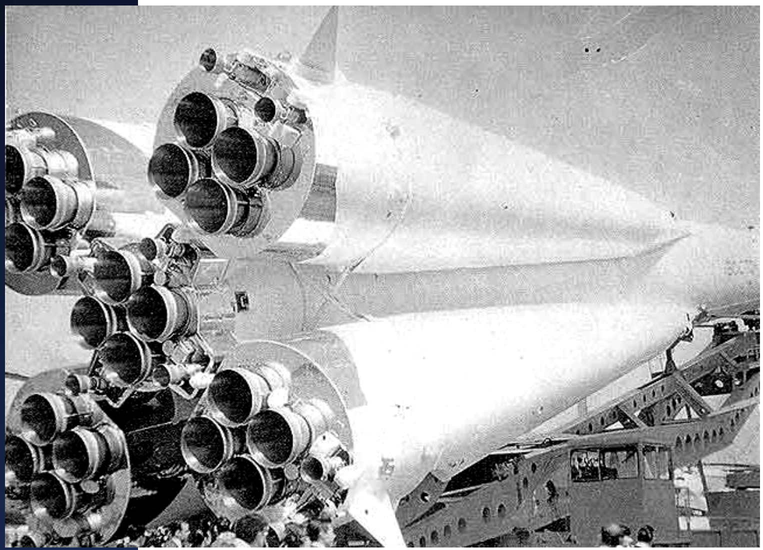
As it is known, the first in human history space flight was accomplished (realized) by Jury Gagarin on 12 April 1961 in space ship “Vostok”.

This space ship (Fig. 1.11, *a*) was launched into orbit by means of three stage carrier rocket. The first stage of the carrier rocket is four liquid-propellant rockets mounted on the lower part of the space ship. Each of these four rockets is over 30 m long. Appearance of the first and second stages of the carrier rocket “Vostok” is shown in Fig. 1.11, *b*. Four chamber liquid-propellant rocket engine РД-107 of this rocket (Fig. 1.11, *c*) is able to exert the start conditions thrust of 398 000 N. The modern liquid-propellant rocket engines such as engines for “Proton”(РД-276; РД-0210; РД-0211; РД-0213) and “Zenit” (РД-120; РД-170) have a thrust considerably greater, than “Vostok” had.

The RD-120 engine is a liquid fuel (liquid–oxygen/kerosene) rocket engine. The engine has been developed from 1976 to 1982 by NPO Energomash with V. P. Radovskiy leading the development. It is manufactured by Yuzhmash in Ukraine along most of the rocket.

The engine is designed to operate at altitude as an upper stage and incorporates a large exit area ratio nozzle to accommodate this. The engine was used on the “Zenit” launch vehicle second stage. It has a non-vectorable nozzle. The propellant delivery is provided via one main and two boost turbopumps. Ignition and turbopump operation is accomplished by using a single preburner. The engine was extremely reliable during its service. A total of 481 tests have been conducted using 150 engines, demonstrating a statistical reliability of 0.995.

Today the thrust of the LRE (liquid propellant rocket engine) is 100–800 t. Because engines work at sea level, the pressure of the products of combustion at the level of the buzzard is limited: it cannot be much more lower than the atmospheric pressure. It means that to increase the specific impulse it is necessary to increase the volume of products combustion in the buzzard. The powerful LRE of the first stages takes into account this characteristic by increasing of the pressure in the combustion chamber.



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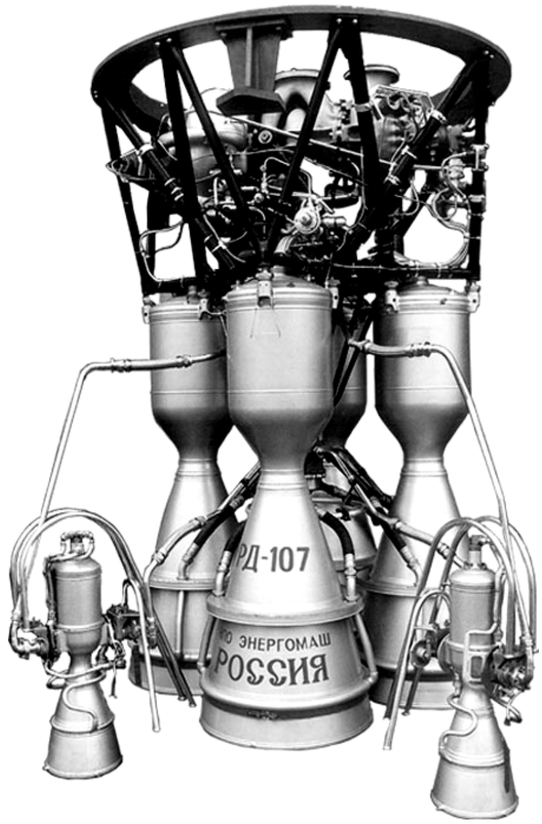


a *c*

Fig. 1.11. Space ship “Voskhod” and its liquid — propellant rocket engine RD-107:
a — outside view rocket — carrier “Voskhod”;
b — “Voskhod” in Air Show Le Bourget (1967);
c — rocket engine RD-107

RD-120 Engine Dimensions

Engine Dimensions	
Characteristic	Dimension (mm, kg)
Length	3872
Diameter	1954
Dry Mass	1125
Wet Mass	1285
Chamber Diameter	320
Characteristic Length, L	1274
Contraction Ratio	1.74 (unitless)
Throat Diameter	183.5
Exit Diameter	189S
Exit Area Ratio	106.7 (unitless)
Chamber Length	2992

RD-120 Performance Parameters

Engine Performance Parameters	
Parameter	Value
Mixture Ratio Control, %	±10
Throttling, %	85
Thrust (vacuum), N	833.565.25
Burn Time, sec	315
Specific Impulse (vacuum), sec	350
Propellent Mass Flow, kg/sec	242.9
Mixture Ratio, 0/F	2.6
Combustion Flame Temperature,	3670
Chamber Pressure, bar	162.7962
Nozzle Exit Pressure, bar	0.127491

The implementation of the high pressures imposed the used of the closed cycle instead of the open cycle. The difference between those *2* principles is that in an opened cycle (Fig. 1.12) the *2* components (the oxidizer and the fuel) are given by pumps to the combustion chambers in the liquid state. Furthermore, a small quantity is taken and steered towards the gas turbine to be able to activate the various pumps then these gases are ejected.

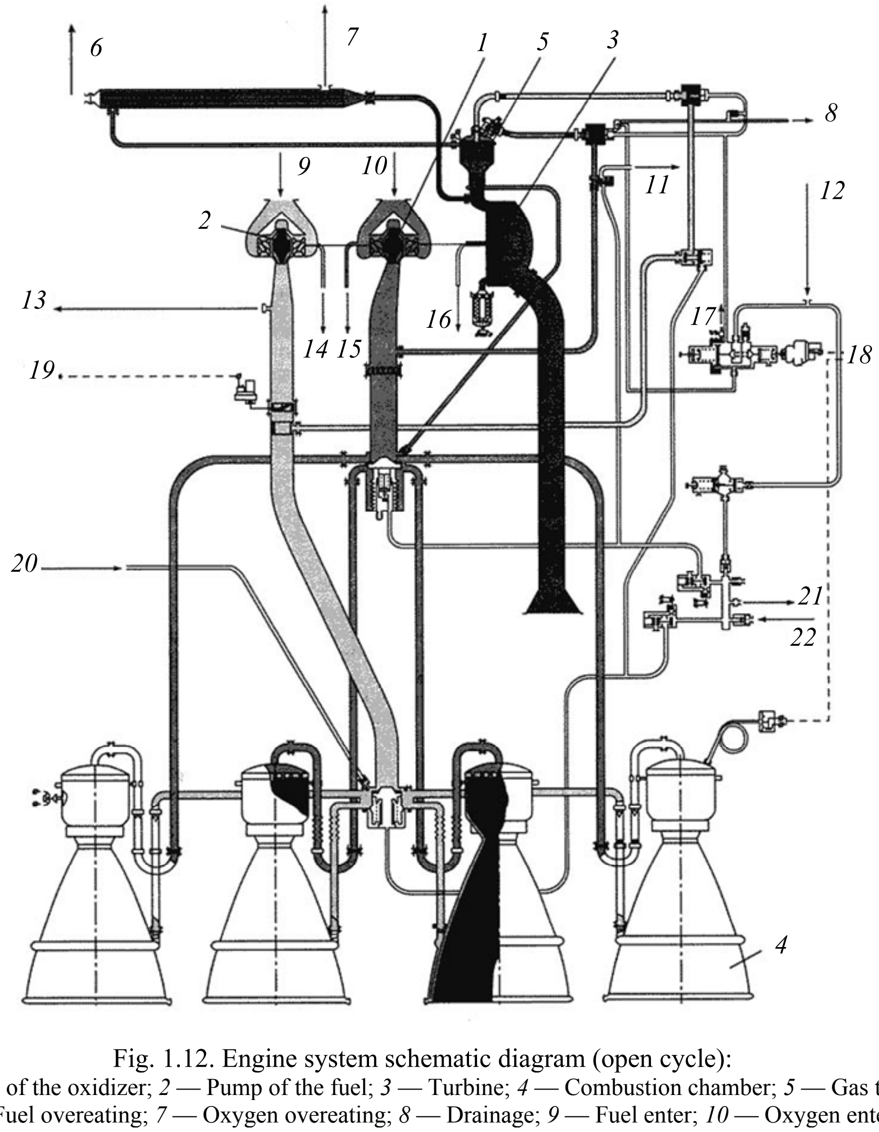


Fig. 1.12. Engine system schematic diagram (open cycle):
1 — Pump of the oxidizer; *2* — Pump of the fuel; *3* — Turbine; *4* — Combustion chamber; *5* — Gas turbine; *6* — Fuel overeating; *7* — Oxygen overeating; *8* — Drainage; *9* — Fuel enter; *10* — Oxygen enter; *11* — Derivation to the tanks; *12* — Air embarked from balloons; *13* — Fuel taking away for the pumps mechanism of the combustion chamber; *14* — Drainage of the fuel; *15* — Drainage of the oxygen; *16* — Drainage of the turbopump; *17* — To the manometer; *18* — Combustion chamber; *19* — To the melange regulator; *20* — Purge of the fuel duct; *21* — To the manometer; *22* — Outside air intake

The main example of the LRE opened cycle are the RD-107 and RD-108 which use as fuels liquid oxygen and some petroleum, they are installed on the rockets Sputnik, Luna, Vostok, Voskhod, Molniya, Soyouz, as well as engines using some acid nitrogen for the rockets Cosmos, Tcyklon, and the others strategic launchers. Engines with opened cycle have a pressure of 50–80 atm in the combustion chamber. The supplementary increase of the pressure is ineffective because of the losses of growth of the specific impulse caused by a not total use of the chemical energy of fuels.

In engines with closed cycles (Fig. 1.13) components burn completely in the combustion chamber. One of them, for example the oxidizer is sent to the gas turbine by the pump, there it is melted with a small quantity of fuel, the process of combustion is not total to insure an acceptable temperature of functioning for the turbine. Later, these gases go to the main combustion chamber where they are added with the rest of the fuel. In this type of engines all the chemical energy of the fuel is used and allows increasing the pressure in its maximum. Today the level of the pressure is 150–270 atm.

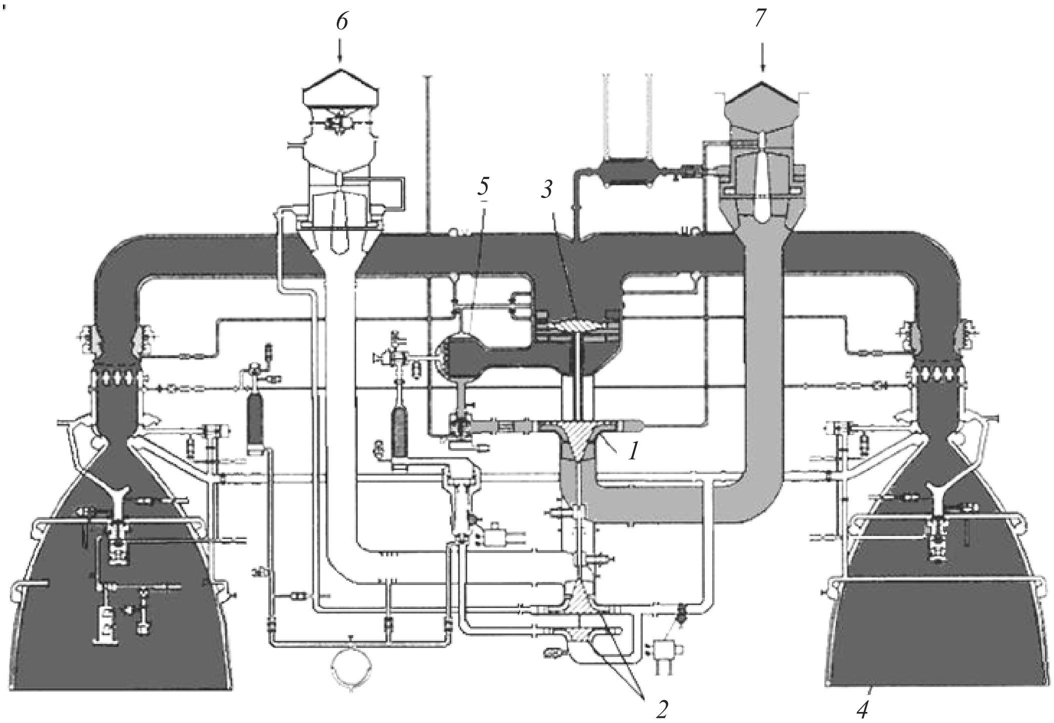


Fig. 1.13. Closed cycle. Engine system schematic diagram:
1 — Pump of the oxidizer; *2* — Pump of the fuel; *3* — Turbine; *4* — Chamber of combustion;
5 — Gas turbine; *6* — Fuel; *7* — Oxygen

The idea of an engine with closed cycle was realized for the first time in 1950 in the IRS (at the moment ФГУП "The Scientific center M. V. Keldysha"), about the engines of great thrusts (150 t at sea level) with a high pressure in the combustion chamber (150 atm) they were realized in the middle of the 60s in ОКБ-456 (at the moment NPO Energomash). The fuel used in this engine, RD-253, intended for the first Proton rocket was nitrogen tetroxide (N_2O_4) and unsymmetrical dimethyl hydrazine (UDMH). On the Proton rocket the first floor included 6 RD-253 engines, which have already made 300 flights.

The use of the closed cycle and the high pressures to increase the specific impulse became the main direction for the liquid propellant rocket engines for spatial and strategic launchers. So, on the P-36M Satan rocket is installed the RD-264 engine with a chamber's pressure of 210 atm, but on "Zenit" and "Energia" rockets RD-171 and RD-170 engines have a pressure in the combustion chamber of 250 atm.

The advantage of this kind of engines is that they are ecologically sure, have a high level on functioning, have a good flexibility of use and consume cheap fuel, such as oxygen and petroleum.

1.3. Compressorless air-breather jet engines

There are two basic types of compressorless air-breather jet engines: ram jet and pulse jet engines.

The ram jet engine (sometimes named as athodyd — that means "aero-thermodynamic duct") is the simplest of the jet engines in that it contains no moving parts (see Fig. 1.2). In general, the ram jet construction resembles a pipe whose diameter enlarges in the combustion area and then gradually decreases toward the rear to reach its minimum size at the exit. Located in the combustion area are flame holders, fuel burners, and an igniter. The ram jet must be accelerated to attain a velocity higher than 250 mph [402.3 km/h] before starting. Once this speed is reached, there is sufficient combustion pressure to continue combustion inside the engine.

The flame holders located in the combustion chamber provide the necessary blockage in the passage to slow down the airflow so that the fuel and air can be mixed and ignited. A fuel metering system is also required to supply the proper amount of fuel to the engine. When forward motion is imparted to ram jet engine from an external source, air is forced into the air intake where it loses velocity or kinetic energy and increases its pressure energy as it passes through the diverging duct. The total energy of fluid is then increased by the combustion of fuel, and expanding accelerating gases expel to the atmosphere through the propelling nozzle.

A ram jet is often the powerplant for missiles and target vehicles, but is unsuitable as an aircraft power plant because it requires forward motion imparting to it before any thrust is produced. Ram jets may be used as augmenters or afterburners on turbojet engines. The augmenter is attached to the rear of the turbojet engine so that the jet exhaust passes through it. Additional fuel is sprayed into the exhaust gases which still contain some oxygen. The fuel burns, creating additional thrust.

1.4. Pulse jet engines

A pulse jet engine is a jet engine to which the intake of air is intermittent (through valves), resulting in a pulsating thrust. It is a jet propulsion engine containing neither compressor nor turbine, equipped with valves in its front that open and shut and so create thrust intermittently, rather than continuously.

The pulse jet engine (see Fig. 1.2) uses the principle of intermittent combustion and unlike the ram jet it can be run at a static condition. The engine is formed by an aerodynamic duct similar to the ram jet but, due to the higher pressures involved it is of more robust construction.

The duct inlet has a series of inlet “shutter valves” that are spring-loaded into the open position. Air drawn through the open valves passes into the combustion chamber and is heated by the burning of fuel injected into the chamber. The resulting expansion causes a rise in pressure, forcing the valves to close, and the expanding gases are then ejected rearwards. A depression created by the exhausting gases allows the valves to open and repeat the cycle. The length of the tailpipe of the pulse jet regulates the frequency of the engine. Fuel flow is continuous, but flame propagation is intermittent, since the pulse jet operates in a step-by-step cycle. This is the only form of jet propulsion that operates by intermittent power surges, utilizing explosive rather than progressive or continuous combustion. However, in most pulse jet engines, the cycles per second are rather high and the net effect is practically continuous thrust.

Pulse jet engines provide thrust for some guided missiles. Pulse jets have been also designed for helicopter rotor propulsion and some dispense with inlet valves by careful design of the ducting to control the changing pressures of the resonating cycle. The pulse jet engine is unsuitable as an aircraft power plant because it has high fuel consumption and is unable to equal the performance of the modern gas turbine engine.

1.5. Gas turbine engines

Gas-turbine engines come in various mechanical arrangements. Aircraft turbine engines can generally be classified into four types of engines: turbojet, turbofan, turboprop, and turboshaft engines. The basic components of all these engines are essentially the same (see Fig. 1.3): an engine air inlet, a compressor, a combustion chamber (in American literature — Combustor), a turbine to drive the compressor, and an exhaust nozzle. The difference lies in the type and arrangement of these components.

The basic operation of the gas turbine or turbojet engine is relatively simple. Air is brought into the front of the turbine engine and compressed, fuel is mixed with this air and burned, and the heated exhaust gases rush out the back of the engine. The parts of a turbojet engine work together to change fuel energy to energy of motion, to cause the greatest thrust for the fuel used. A basic turbine engine is illustrated in Fig. 1.14.

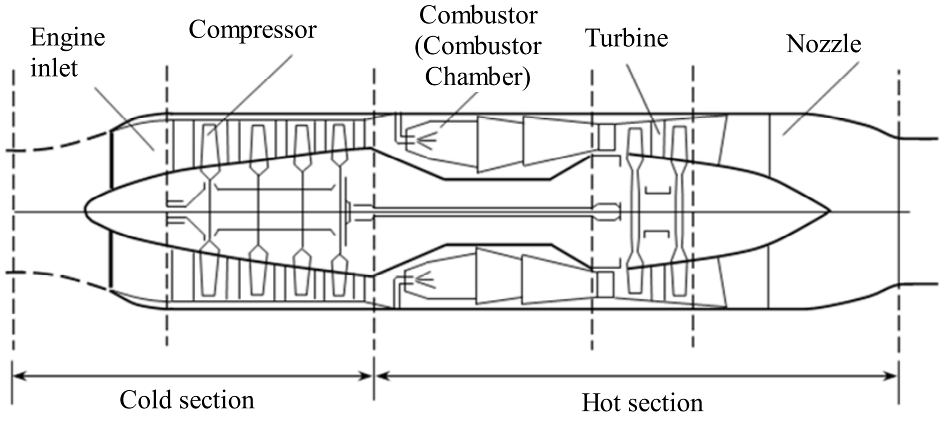


Fig. 1.14. Basic sections and parts of a gas-turbine engine

A gas turbine engine has three major sections: an air compressor, a combustion section, and a turbine section. The engine may also be divided into the cold section and the hot section. The forward or front part of the engine contains the air compressor, which is the cold section. The combustion and turbine sections make up the hot section of the engine. The compressor packs several tons of air into the combustion chamber every minute and works somewhat like a series of fans. Then fuel is forced into the combustion chamber through nozzles, a spark provides ignition, and the mixture burns in a process similar to a blowlamp, creating hot exhaust gases. These gases expand and are ejected from the rear of the engine. As the gases leave, they spin a turbine which is located just behind the combustion chamber. By means of an interconnecting shaft, the rotating turbine is connected to and turns the compressor, completing the cycle. After rushing by the turbine, the hot gases continue to expand and blast out through the exhaust nozzle at a high velocity, creating the force which propels a jet aircraft.

Gas turbines produce work in proportion to the amount of heat released internally. Therefore, it is necessary to study the production of heat in the engine, most of which is obtained by the burning of fuel (although some is obtained by compressing the air in the compressor). An ordinary thermometer indicates the temperature of the gases, but does not tell the quantity of heat that is available. Heat can not be measured directly, but must be calculated from three known quantities: temperature, mass (or weight), and specific heat.

Imagine two turbojet engines, one using 10.000 Ib/h [4540 kg/h] of fuel and the other using 1000 Ib/h [454 kg/h]. Both engines are operating at the same turbine inlet temperature. However, the larger engine can do approximately ten times the work of the smaller because ten times more heat is released at the same temperature. This example illustrates that heat and temperature are interconnected, but they are not the same.

Both the reciprocating engine and the gas turbine engine are considered **heat engines.** Both develop power or thrust by burning a combustible mixture of fuel and air. Both convert the energy of expanding gases to propulsive force. The reciprocating engine does this by changing the energy of combustion to mechanical energy which is used to turn a propeller. Aircraft propulsion is obtained as the propeller imparts a relatively small amount of acceleration to a large mass of air. The gas turbine, in its basic turbojet configuration, imparts a relatively large amount of acceleration to a smaller mass of air and thus produces thrust or propulsive force directly.

The turbojet engine is a heat engine and the higher the temperature of combustion the greater the expansion of the gases. The combustion temperature, however, must not exceed a value that provides a turbine gas entry temperature suitable for the design and materials of the turbine assembly.

The mechanical arrangements of various types of gas-turbine engines are shown in Fig. 1.15–1.21.

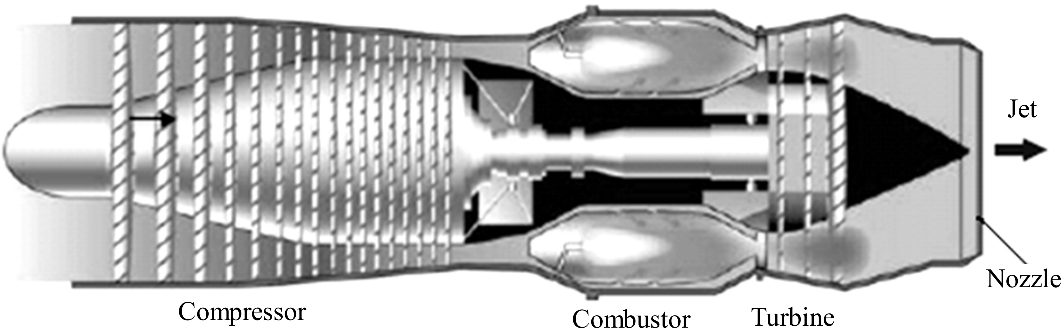


Fig. 1.15. Schematic diagram of a single-shaft turbojet engine

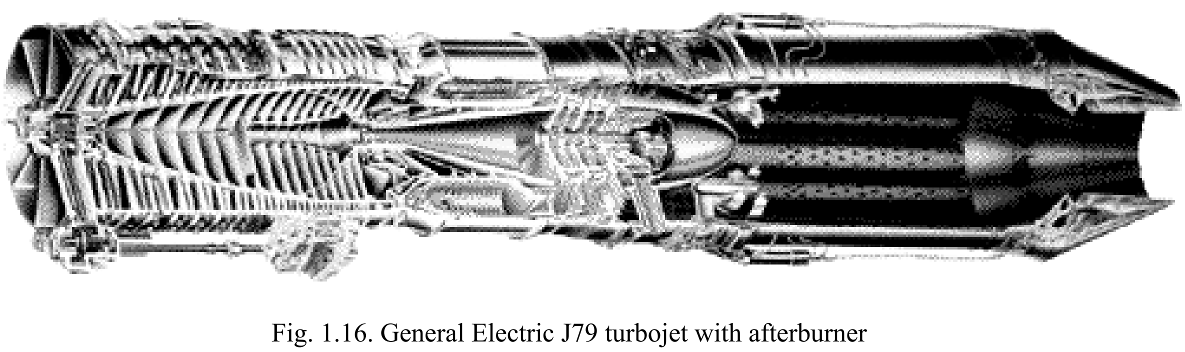


Fig. 1.16. General Electric J79 turbojet with afterburner

1.6. Turbojet engines

A turbojet engine is a type of gas turbine engine which produces thrust through the hot gases exiting the exhaust section of the engine. The first turbo jet engine was designed and produced by British engineer Frank Whittle. In 1930 Frank Whittle was granted his first patent for using a gas turbine to produce a propulsive jet, but it was eleven years before his engine completed its first flight.

In the Soviet Union, the first turbojet engine was designed by Ukrainian engineer Arkhip Mikhaylovich Lulka, a graduate of Kiev Polytechnic institute in 1930. He began work upon his project of turbojet engine with axial flow compressor during 1939–1940. But, the Great Patriotic war of 1941 interrupted this work. So, the first turbojet engine in the Soviet Union, named TP–1, was able to be manufactured only after the war. And its first test flight was completed in 1946. Later on, A. M. Lulka designed many advanced turbojet engines such as АЛ-5, АЛ-7, etc. Some of them are used in military airplanes even till present date. Most well-known, among soviet commercial turbojet engines, is РД-3М-500 designed by A. Mikulin during 1950–1955. This engine was used in the first passenger jet airliner Ту-104. Schematic diagrams of two turbojet engines are shown in Fig. 1.17. The first is a single-spool centrifugal flow turbo-jet engine (Fig. 1.17, *a)*, and the second is a twin-spool axial flow turbojet engine with afterburner (Fig. 1.17, *b*). Both, first and second engines consist of five main units (*1* — air inlet, *2* — compressor, *3* — combustor, *4* — turbine, *5* — nozzle) but, the second includes additional unit *6* — afterburner chamber. This device permits to increase the value of engine thrust burning additional fuel in afterburning jet pipe with a variable nozzle at the rear. Other difference between these two engines is different types of theirs compressors. The first engine (Fig. 1.17, *a*) includes the centrifugal compressor unlike the second engine which has axial flow compressor (Fig. 1.17, *b*).

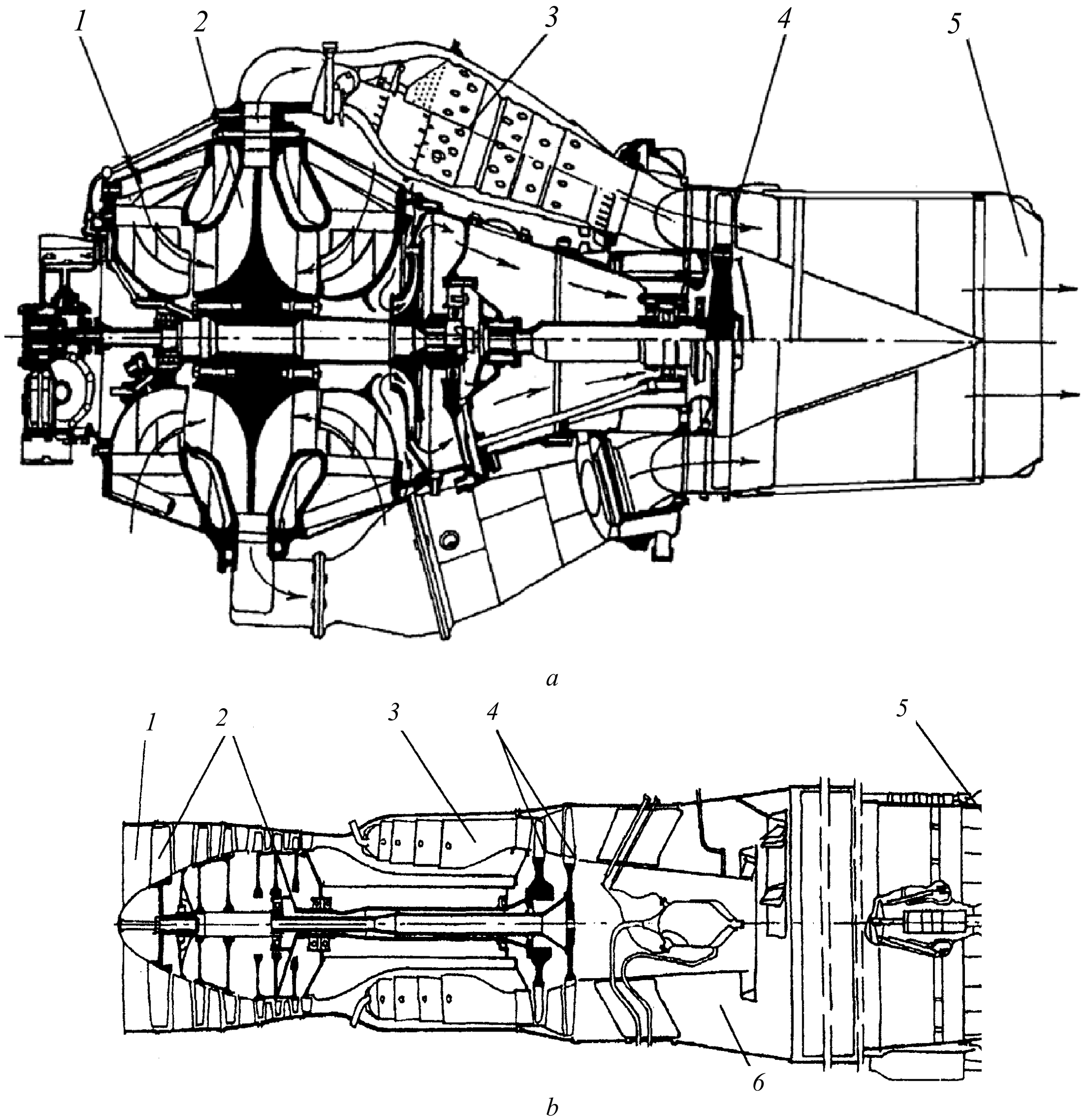
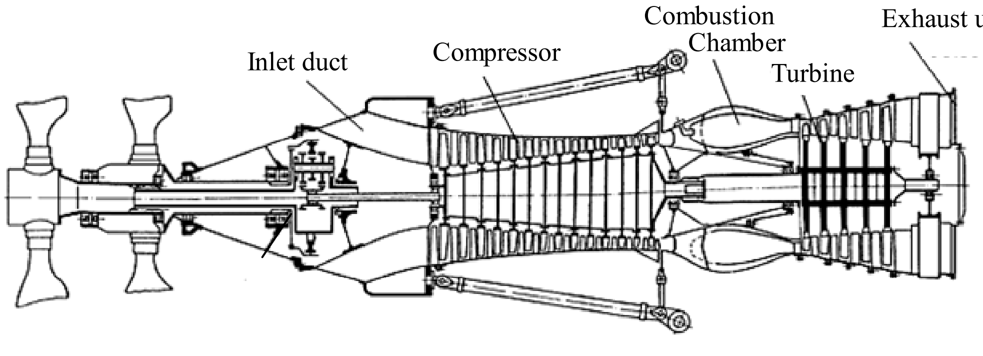


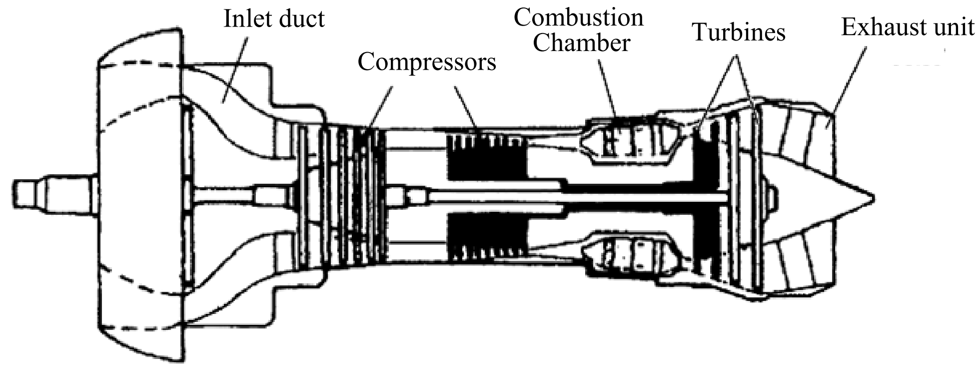
Fig. 1.17. Schematic diagrams of turbojet engines:
a — double-entry single-stage centrifugal flow turbojet engine; *b* — twin-spool axial flow turbojet engine with afterburner; *1* — air inlet; *2* — compressor; *3* — combustor; *4* — turbine; *5* — exhaust nozzle; *6* — afterburner chamber

1.7. Turboprop engines

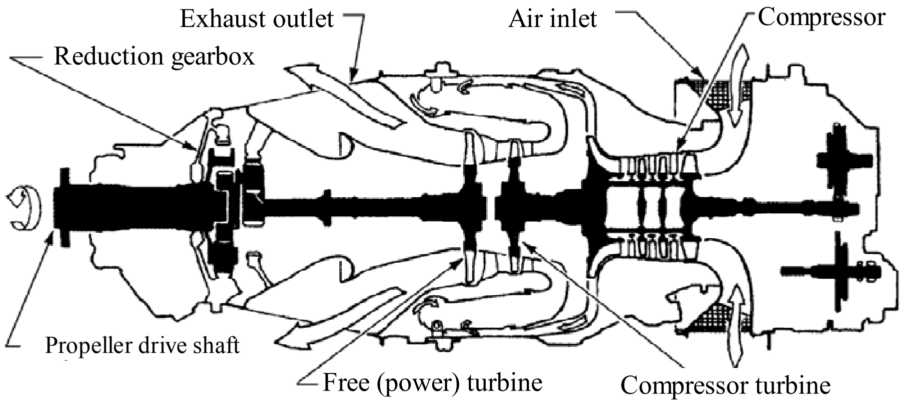
The turboprop (turbo-propeller) engine is a combination of a gas turbine and a propeller (Fig. 1.18).



a



b



c

Fig. 1.18. Turboprop engines:
a — single-spool axial flow turboprop engine; *b* — twin-spool turboprop engine;
c — countercurrent flow turboprop engine with free power turbine

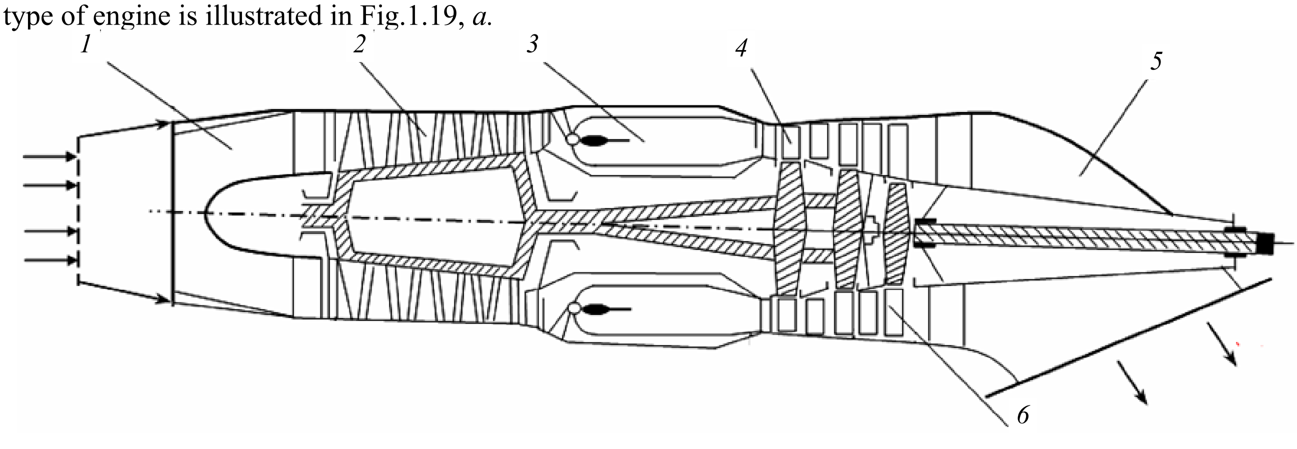
Turboprops are basically similar to turbojet engines in that both have a compressor, combustion chamber, turbine, and a jet nozzle, all of which operate in the same manner on both engines.

However, the difference is that the turbine in the turboprop engine usually has more stages than that in the turbojet engine. In addition to operating the compressor and accessories, the turboprop turbine transmits increased power, through a shaft and a reduction gear box, to drive the propeller. This engine uses almost all the exhaust-gas energy to drive the propeller and therefore provides very little thrust through the ejection of the exhaust gases.

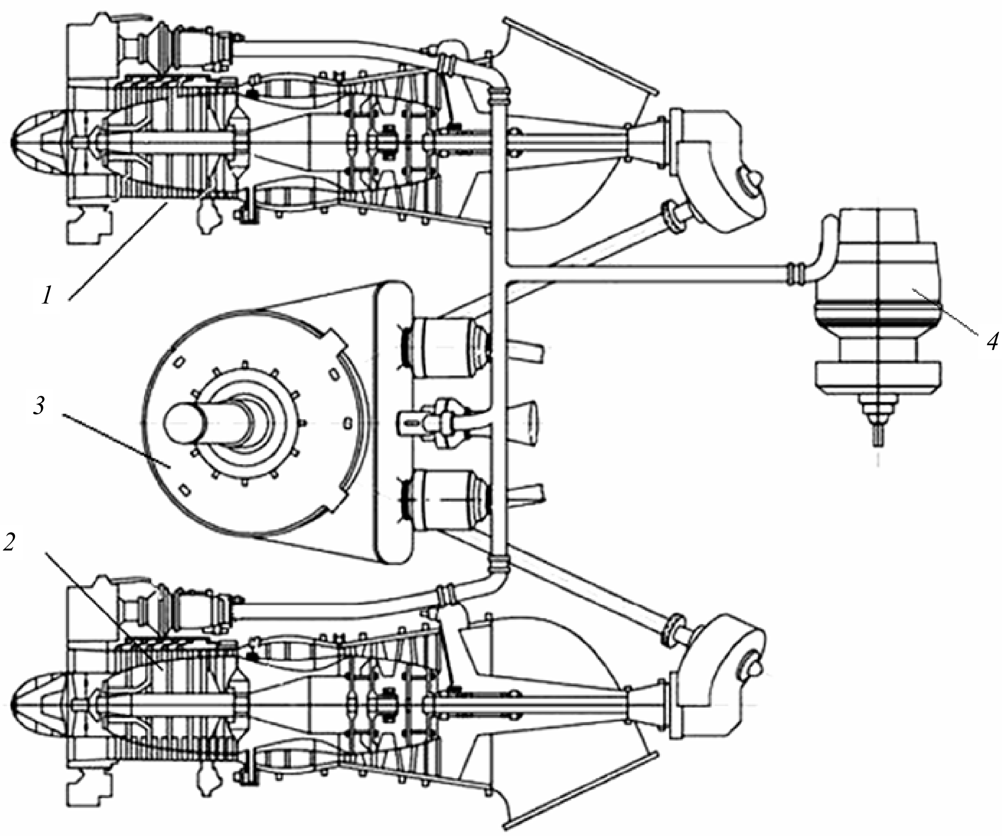
The exhaust gases represent only about 10 percent of the total amount of energy available. The other 90 percent of the energy is extracted by the turbines that drive the compressor and the propeller. Most well-known, among soviet commercial turboprop engines, are AI-20, and AI-24, designed by A. Ivchenko during 1958–1965, and most high-power turboprop HK-12, designed by N. D. Kuznetsov (see Fig. 1.16, *a*). Four turbo-propellers NK-12 were installed in the passenger airliner Tu-114.

1.8. Turboshaft engines

A gas turbine engine that delivers power through a shaft to operate something other than a propeller is referred to as a turboshaft engine (Fig. 1.19). Turboshaft engines, like the turboprop, use almost all the exhaust energy to drive the output shaft. The power takeoff may be coupled directly to the engine turbine, or the shaft may be driven by a turbine of its own (free turbine) located in the exhaust stream. The free turbine rotates independently. This principle is used extensively in current production turboshaft engines. An example of such type of engine is illustrated in Fig.1.19, *a*.



a



b

Fig. 1.19. Turboshaft engines:
a — simplest turbofhaft engine; *b* — helicopter powerplant with two turboshaft engines

The main parts of this engine (air inlet *1*, compressor *2*, combustor *3*, turbine *4* and exhaust pipe *5*) are identical to those of the mentioned above turbo-propeller, and the only difference is additional turbine *6*, which is called free or power turbine and is mounted in the rear end of engine to drive an output shaft.

The turboshaft engines are currently used in aviation mainly on helicopters and for as auxiliary power units on large transport aircraft. An example of helicopter powerplant, that consists of two turboshaft engines is illustrated in Fig. 1.19, *b*. Two engines *1* and *2* in this diagram are connected with common helicopter reduction gearbox *3* to drive the main lift rotor. To start main engines this type of powerplant includes auxiliary power unit *4*. Such powerplant is used in the majority of modern domestically produced helicopters such as Mi-8, Mi-26, etc (Fig. 1.20).

b

a *c*

Fig. 1.20. Helicopters:
a — Mil Mi-26 recovering an American CH-47 Chinook helicopter;
b — Mil Mi-10; *c* — Mil V-12, largest helicopter in the world

Most well-known, among domestically produced turboshaft engines, are TV2-117, TV3-117, developed by Klimov J.-S.Co (the city of Saint-Petersburg) and most high-power D-136, developed by the Ivchenko PROGRESS Designing Bureau (Zaporizhia).

1.9. By-pass engines

A by-pass (or turbofan) engine may be considered a cross between a turbojet engine and a turboprop engine. The turboprop engine drives a conventional propeller through reduction gears to provide a speed suitable for the propeller. The propeller accelerates a large volume of air in addition to that which is accelerated by the engine itself. The turbofan engine accelerates a smaller volume of air than the turboprop engine but a larger volume than the turbojet engine.

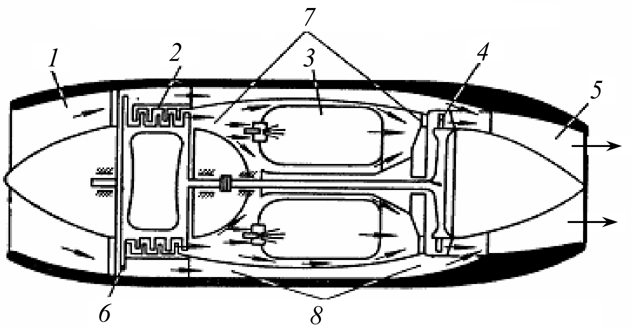


Fig. 1.21. Lulka's conception of by-pass engine

The first concept of by-pass engine was proposed by Ukrainian engineer Arhip M. Lulka, a graduate of Kiev Polytechnic institute. Fig. 1.21 shows the copy of Lulka's patent application, concerning this type of engine. As we can see, proposed conceptual design of the new engine includes five basic (main) units that are in any turbojet engine: air intake *1*, compressor *2*, combustor *3*, turbine *4* and exhaust nozzle *5*. The basic difference of the new engine in the comparison with conventional turbojet engine is in presence of fan unit *6* and two air ducts: inner duct *7* and external duct *8* (by-pass air duct). Fan unit *6* driven by turbine, 4 provides additional air flow thus increases mass of air accelerated in engine and so increases the thrust.

A. M. Lulka patented this new type of a gas turbine engine in 1937, but at that period it was impossible to manufacture it, since suitable heat resisting materials had not then been developed. The point is that this type of engine may be efficient only if the gas temperature level is higher than that in a turbojet and turboprop engine. It was clear to A. M. Lulka, so he paid his attention to developing a turbojet engine. But the first by-pass engines for commercial aircraft were developed, and manufactured in Great Britain in 1959. Nowadays this engine type is widely used for transport aircraft. Schematic diagram and a cutaway view of a two-shaft high-bypass turbofan engines are shown in Fig. 1.22, 1.23.

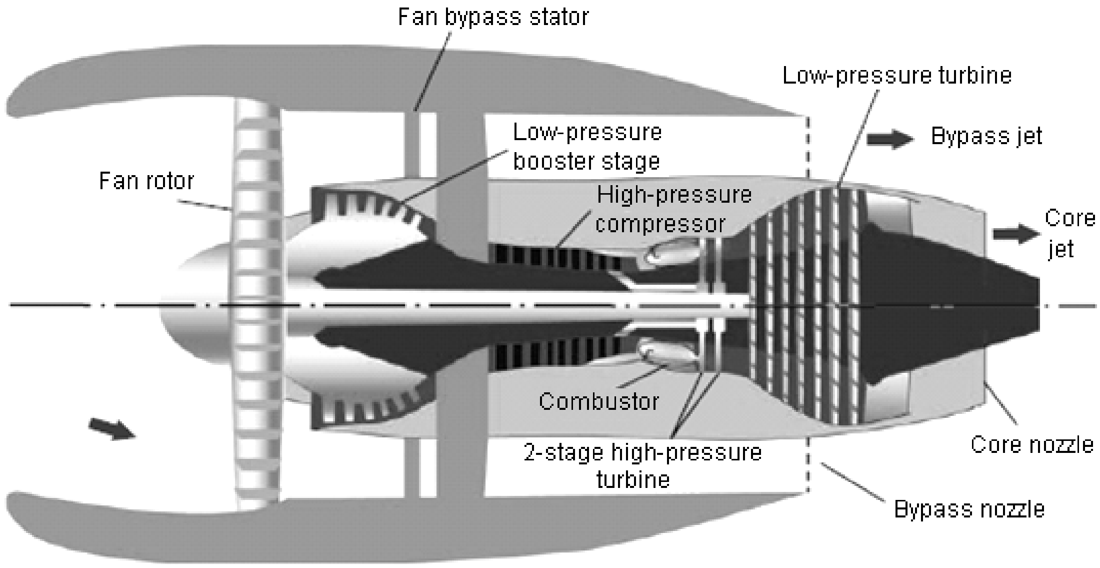


Fig. 1.22. Schematic diagram of a two-shaft high-bypass turbofan engine

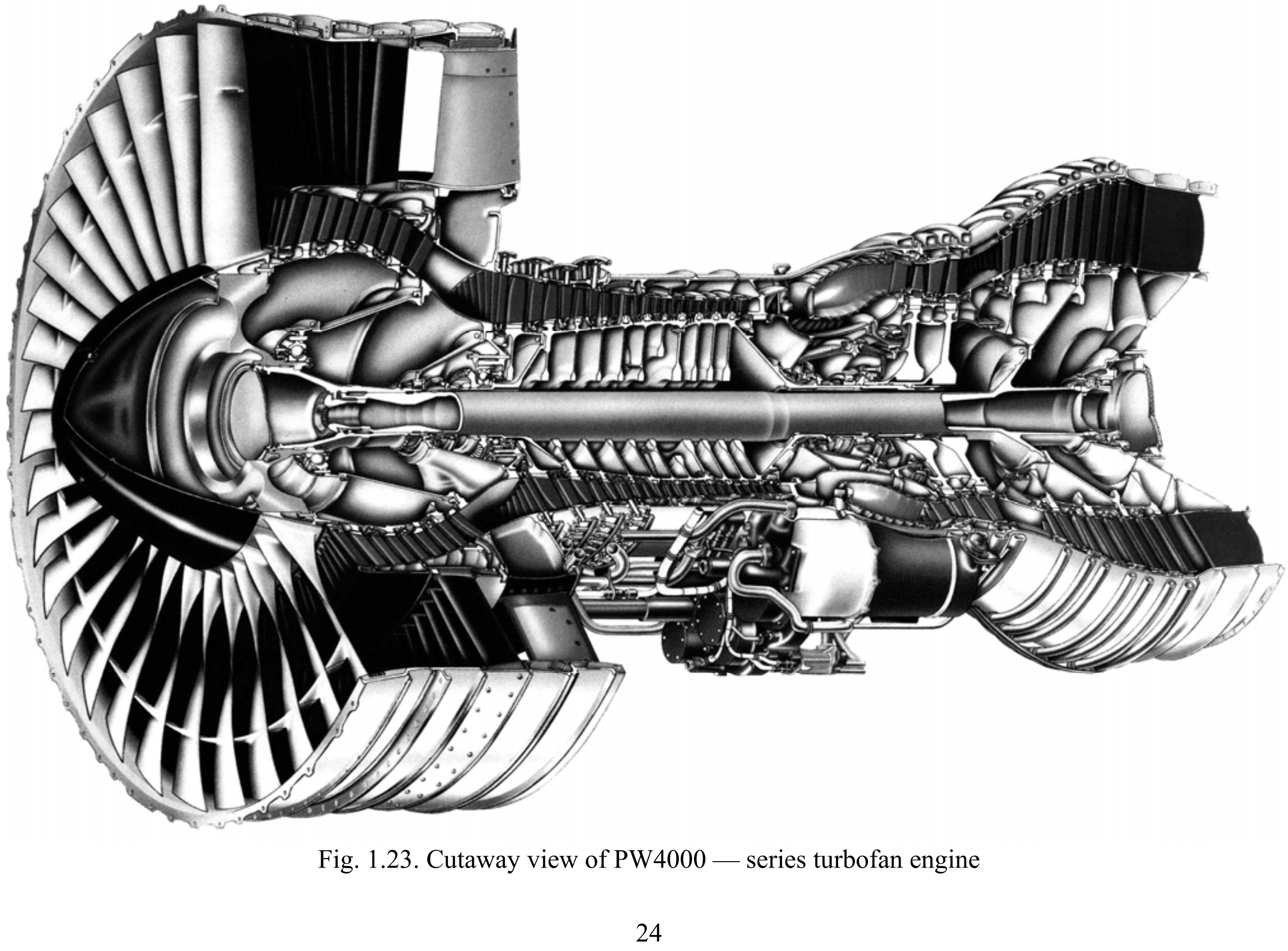


Fig. 1.23. Cutaway view of PW4000 — series turbofan engine

The arrangement of a forward turbofan engine with a dual compressor is shown in Fig. 1.24. This bypass engine, like the mentioned above turbojet and turboprop engines, consists of five basic units: inlet unit *1*, compressor *2*, combustor *3*, turbine unit *4* and exhaust nozzle *5*. But the compressor unit *2* in this case consists of two units: low pressure compressor *9* and high pressure compressor *8*. Each of these compressors is driven by its own turbine. The low pressure turbine *6* is connected with and drives the low pressure compressor. The high pressure turbine *7* is connected with high pressure compressor and drives it.

During operation, air from the fan section of the forward blades is carried outside to the rear of the engine, through ducting. The by-pass engine has two gas streams: cool by-pass airflow and hot turbine discharge gases which pass through the core of the engine. The by-pass air or fan air is cool because it has not passed through the actual gas-turbine engine.

Two different duct designs are used with forward-fan engines. The air leaving the fan flows along the outer case of the basic engine is discharged through the jet nozzle (Fig. 1.24), or it can be exhausted overboard through the separate — fan duct nozzle (Fig. 1.25). The fan air is either mixed with the exhaust gases before it is discharged or it passes directly to the atmosphere without prior mixing.

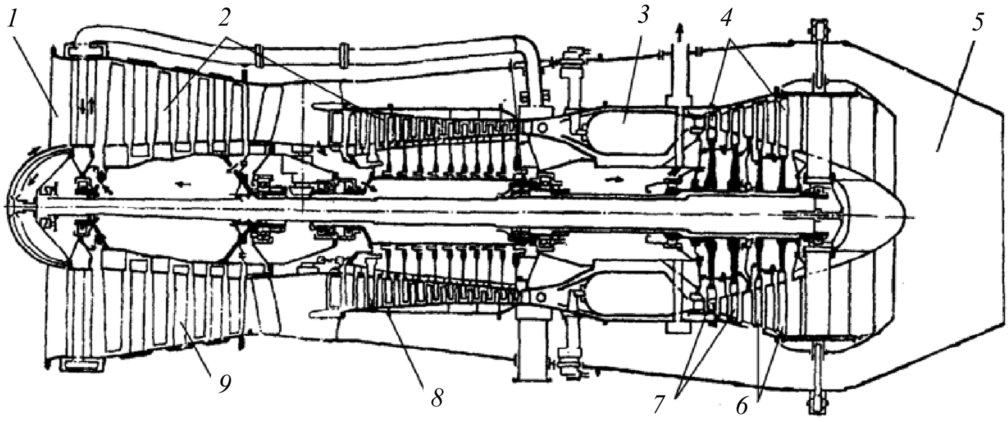


Fig. 1.24. Schematic diagrams of twin spool by-pass turbofan (low by-pass ratio)

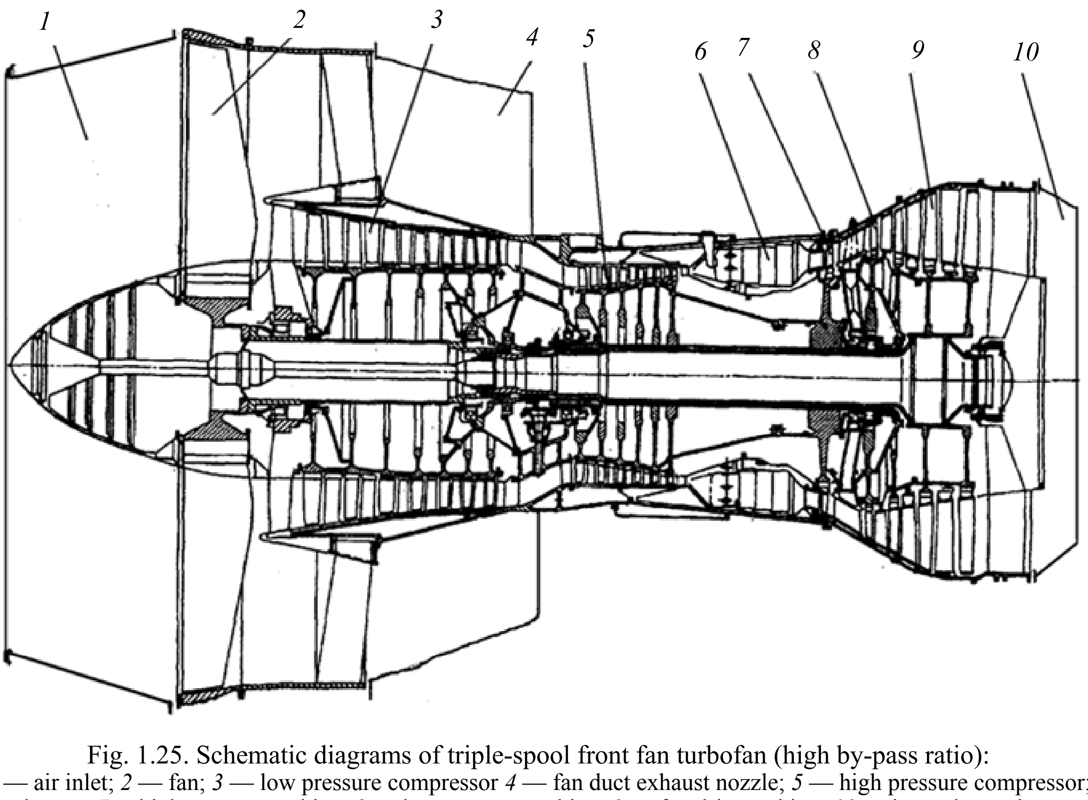


Fig. 1.25. Schematic diagrams of triple-spool front fan turbofan (high by-pass ratio):
1 — air inlet; *2* — fan; *3* — low pressure compressor *4* — fan duct exhaust nozzle; *5* — high pressure compressor; *6* —combustor; *7*— high pressure turbine; *8* — low pressure turbine; *9* — fan drive turbine; *10* — inner duct exhaust nozzle

Turbofan engines may be high by-pass or low-bypass engines. The ratio of the amount of air that by-passes (passes around) the core of the engine to the amount of air that passes through the core is called the bypass ratio. The low bypass engine does not by-pass as much air around the core as the high-bypass engine.

During recent years, the high bypass turbofan engine has become one of the principal sources of power for large transport aircraft. Among such engines are the Pratt & Whitney JT9D, the General Electric CF6, the Rolls Royce RB 211 and the Ivchenko Zaporizhia PROGRESS designing bureau D-18 and D-36. The by-pass ratio of these engines is near 5:1 or even more (Fig. 1.25). In this case fan *2* is driven by separate turbine *9*, which is called the fan turbine. Low pressure compressor *3* is driven by low pressure turbine *8* and high pressure compressor *5* is driven by high pressure turbine *7*. The air leaving the fan flows in the short outer duct and is discharged to atmosphere through separate nozzle *4*. The gases after passing through all turbines are discharged through exhaust nozzle *10*.

In this case fan air can account for around 80 percent of the engine's total thrust. The effect of the turbofan design is to greatly increase the power weight ratio of the engine and to improve the thrust specific fuel consumption.

Principal advantages of the high-bypass engine are greater efficiency and reduced noise. The high-bypass engine has the advantages of the turboprop engine but does not have any problems of propeller control. The design is such that the fan can rotate at its most efficient speed, depending on the speed of the aircraft and the power required engine.

REVIEW QUESTIONS

1. What are two main propulsive devices and what is the main difference between them?
2. Give classification of the air-breathing engines and make a list of main components of each engine.
3. Why does not a ram jet engine require a compressor and a turbine?
4. How do rocket engines work?
5. What are the types of rocket propulsion?
6. How does rocket propulsion differ from jet propulsion?
7. What is a propellant?
8. Give the description of rocket engine design.
9. What are some rocket propellants?
10. What is a pulsejet?
11. What are the main components of GTE?
12. What main components does the gas generator consist of?
13. What components does the hot section consist of?
14. Which place in the turbojet engine is subjected to the highest temperature?
15. What is a turboprop engine?
16. What is the difference between a turboprop engine and a turbofan engine?
17. In what engine does not the most of the air entering the engine pass through the gas generator?
18. What type of gas turbine engines is most suitable for regional medium range airliners?
19. Which type of gas turbine is used for helicopters?

Chapter 2. ENGINE OPERATING FUNDAMENTALS

2.1. Concept

The gas turbine engine is essentially a heat engine using air as a working fluid to provide thrust. To achieve this, the air passing through the engine has to be accelerated; this means that the velocity or kinetic energy of the air is increased. To obtain this increase, the pressure energy is first of all increased, followed by the addition of heat energy, before final conversion back to kinetic energy in the form of a high velocity jet efflux. These changes of state of working fluid which take place in any heat engine are named operation processes. Taken as a whole all this, processes are named as engine **operating cycle**. **A cycle** is a complete sequence of events returning to the original state. That is, a cycle is an interval of time occupied by one round, or course, of events repeated in the same order in a series-such as the cycle of the seasons, with spring, summer, autumn, and winter following each other and then recurring.

An engine cycle is the series of events that an internal-combustion engine goes through while it is operating and delivering power. In a four-stroke five-event cycle these events are intake, compression, ignition, combustion, and exhaust. An **internal-combustion engine,** whether it be a piston-type or gas-turbine engine, is so called because the fuel is burned inside the engine rather than externally, as with a steam engine. Since the events in a piston engine occur in a certain sequence and at precise intervals of time, they are said to be timed.

Most piston-type engines operate on the four-stroke five-event-cycle principle originally developed by August Otto in Germany. There are four strokes of the piston in each cylinder, two in each direction, for each engine operating cycle. The five events of the cycle consist of these strokes plus the ignition event. The four-stroke five-event cycle is called the **Otto cycle.** Other cycles for heat engines are the **Carnot cycle,** named after Nicolas-Leonard-Sadi Carnot, a French engineer; the **Diesel cycle,** named after Dr. Rudolf Diesel, a German scientist; and the **Brayton cycle**, named for George B. Brayton, a U.S. engineer. All the cycles mentioned pertain to the particular engine theories developed by the men whose names are given to the various cycles. All the cycles include the compression of air, the burning of fuel in the compressed air, and the conversion of the pressure and heat to power.

2.2. A Piston Engine

Piston engines are used for low-speed flight. An airplane piston engine works on the same principle as an automobile engine. It releases the energy stored in fuel by burning the fuel in a «four-stroke» cycle.

The parts of an engine vary depending on the engine's type. For a four-stroke engine, key parts of the engine include the **cylinder, piston, connecting rod,** and **crankshaft.**

These are shown in Fig. 2.1. The cylinder has a smooth surface such that the piston can, with the aid of piston rings and a lubricant, create a seal so that no gases can escape between the piston and the cylinder walls. The piston is connected to the crankshaft by means of the connecting rod so that the rotation of the crankshaft causes the piston to move with a reciprocating motion up and down in the cylinder.

The distance through which the piston travels is called the **stroke.** During each stroke, the crankshaft rotates 180°. The limit of travel to which the piston moves into the cylinder is called **top dead center (TDC),** and the limit to which it moves in the opposite direction is called **bottom dead center (BDC).**

For each revolution of the crankshaft there are two strokes of the piston, one up and one down, assuming that the cylinder is in a vertical position.

It is important to understand top dead center and bottom dead center because these positions of the piston are used in setting the timing and determining the valve overlap. Top dead center TDC may be defined as the point a piston has reached when it is at its maximum distance from the centerline of the crankshaft. Top dead center is the position of the piston when it forms the smallest volume in the cylinder.

The minimum volume formed in the cylinder when the piston is at TDC is called the **clearance volume**. The bottom dead center BDC may be defined as the point a piston has reached when it is at a minimum distance from the centerline of the crankshaft. Bottom dead center is the position of the piston when it forms the largest volume in the cylinder. Fig. 2.2 illustrates the piston positions at TDC and at BDC.

The distance between TDC and BDC is the largest distance that the piston can travel in one direction, and it is called the **stroke**. The diameter of the piston is called the **bore**.

Compression Ratio. The ratio of the maximum volume formed in the cylinder to the minimum (clearance) is called the compression ratio of the engine (it is determined by ***r*** or ε).

$$r = \varepsilon = \frac{V_{\max}}{V_{\min}} = \frac{V_{\mathrm{BDC}}}{V_{\mathrm{TDC}}}.$$

For example, if the volume of the space in a cylinder is 120 in^3 [0.00197 m^3] when the piston is at the bottom of its stroke and the volume is 20 in^3 [0.00033 m^3] when the piston is at the top of its stroke, the compression ratio is r = 120:20 = 6:1. This is the usual manner for expressing a compression ratio.

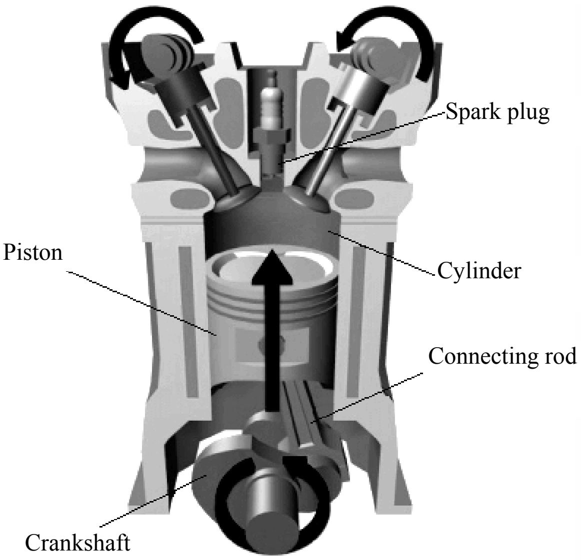


Fig. 2.1. Key components in a typical four-stroke engine

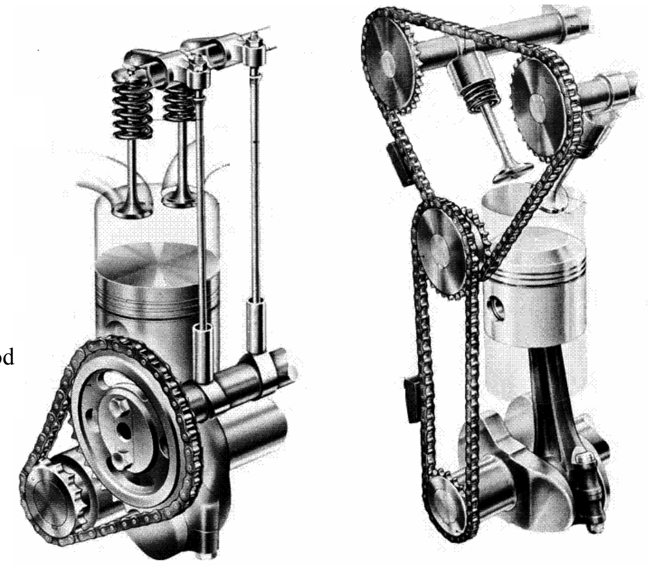


Fig. 2.2. Displacement of piston

2.3. The Four-Stroke Five-Event Cycle

The four strokes of a four-stroke-cycle engine are the intake stroke, the compression stroke, the power stroke, and the exhaust stroke. In a four-stroke-cycle engine, the crank shaft makes 2 revolutions for each complete cycle. The names of the strokes are descriptive of the nature of each stroke (Fig. 2.3).

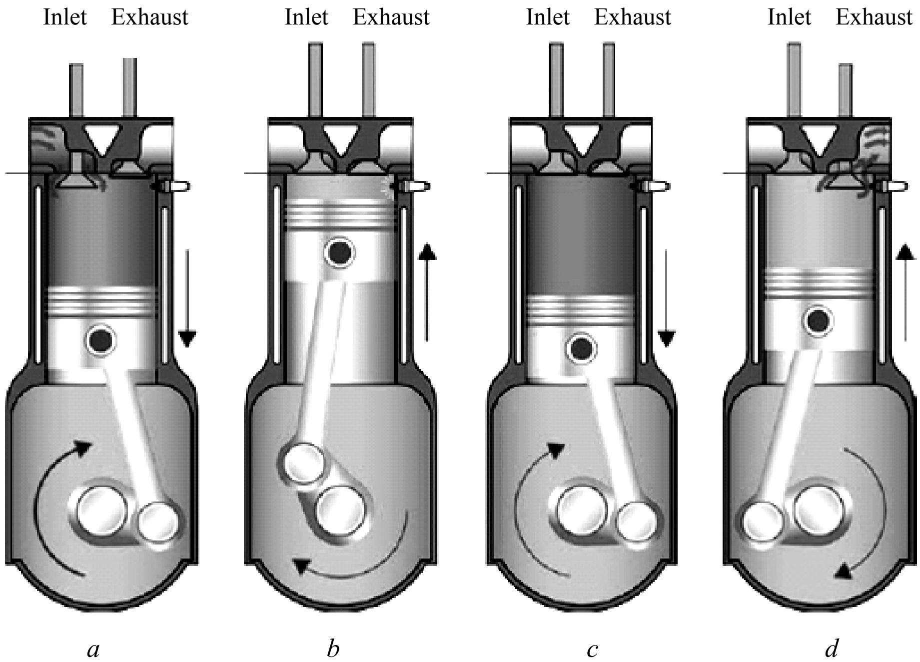


Fig. 2.3. Strokes of four-stroke-cycle engine:
a — intake; *b* — compression; *c* — expansion; *d* — exhaust

During the **intake stroke**, the piston starts at TDC with the intake valve open and the exhaust valve closed. As the piston moves downward, a mixture of air and fuel (sometimes called the **working fluid** or working body) is drawn from the carburetor into the cylinder. At the end of the intake stroke the piston reaches BDC, the intake valve closes. Now both valves have closed. The piston moves upward, compressing the air-fuel mixture in the cylinder. This event is called the **compression stroke**. The amount that the mixture is compressed is determined by the compression ratio of the engine. Shortly before the piston reaches TDC **ignition** takes place. The spark plug fires and the mixture ignites, increasing the pressure and temperature in the cylinder. The high-pressure gas pushes the piston downward toward BDC with great force turning the crankshaft to provide the power to propel the vehicle. This stoke is called the **power stroke**. The power stroke is also called the **expansion stroke** because the gas expansion takes place at this time.

Well before the piston reaches BDC on the power stroke, the exhaust valve opens, and the hot gases begin to escape from the cylinder. The pressure differential across the piston drops to zero, and the gases that remain in the cylinder are forced out through the open exhaust valve as the piston moves back toward TDC. This is the **exhaust stroke** and is also called the **scavenging stroke** because the burned gases are scavenged (removed from the cylinder) during the stroke.

We may summarize the complete cycle of the four-stroke-cycle engine as follows: **intake stroke** — the intake valve is open and the exhaust valve is closed, the piston moves downward, drawing the air-fuel mixture into the cylinder, and the intake valve closes; **compression stroke** — both valves are closed, the piston moves toward TDC, compressing the air-fuel mixture, and ignition takes place near the top of the stroke; **power stroke** — both valves are closed, the pressure of the expanding gases forces the piston toward BDC, and the exhaust valve opens well before the bottom of the stroke; **exhaust stroke** — the exhaust valve is open and the intake valve closed, the piston moves toward TDC, forcing the burned gases out through the open exhaust valve, and the intake valve opens near the top of the stroke.

The five-event sequence of intake, compression, ignition, power, and exhaust is a cycle which must take place in the order given if the engine is to operate at all, and it must be repeated over and over for the engine to continue operation. Engines based on the four-stroke cycle have one power stroke for every four strokes (up-down-up-down). None of the events can be omitted, and each event must take place in the proper sequence. For example, if the gasoline supply is shut off, there can be no power event. The mixture of gasoline and air must be admitted to the cylinder during the intake stroke. Likewise, if the ignition switch is turned off, there can be no power event because the ignition must occur before the power event can take place.

Note at this point that each event of crankshaft rotation does not occupy exactly 180° of crankshaft travel. The intake valve begins to open substantially before TDC, and the exhaust valve closes after TDC. This is called **valve overlap** and is designed to take advantage of the inertia of the out flowing exhaust gases to provide more complete scavenging and to allow the entering mixture to flow into the combustion chamber at the earliest possible moment, thus greatly improving volumetric efficiency (volumetric efficiency of an engine is the ratio of the actual volume of air at atmospheric pressure and ambient temperature that gets into the engine on the intake stroke over the total volume at BDC).

2.4. Gas turbine engine

Main parts of any gas turbine engine were considered in previous chapter. The operating processes which take place in gas turbine engine are similar to those in piston engine (Fig. 2.4).

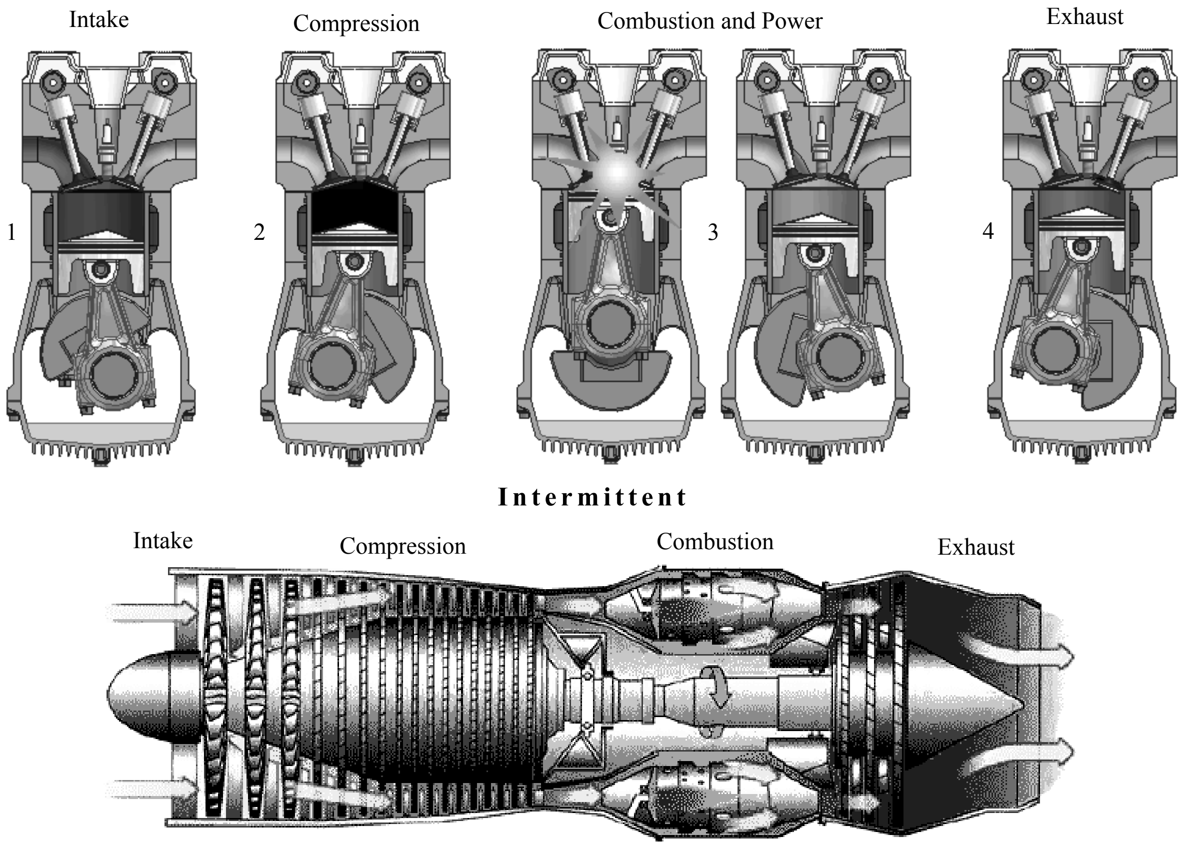


Fig. 2.4. Comparison between a piston engine and a gas turbine engine working cycles

The difference between these two working cycles is in the way and places where operating processes take place. In the piston engine all processes occur intermittently inside the cylinder, whereas in a gas turbine engine compression, combustion, expansion take place continuously in different parts of engine, namely: in compressor, combustor, turbine and exhaust pipe and propelling nozzle.

2.5. Working cycles

The Constant Volume Combustion Cycle. The investigation of the work and power of a piston engine is best accomplished by the use of the $p - V$ diagram in which the abscissas show the volumes occupied by the working body in the cylinder, and the ordinates give the corresponding absolute pressures.

Since the changes in the volume of the working body in the cylinder are directly proportional to the shifting of the piston, this diagram must show the changes in pressure in the cylinder depending on the shifting of the piston. These diagrams for piston engines are known as **indicator diagrams**.

The indicator diagram is a trace made by a recording pressure gauge, called the indicator, attached to the cylinder of a reciprocating engine. The indicator produces the closed curve of the pressure changes in the cylinder depending on the position of the piston. The area of the figure in the indicator diagram gives the value of the indicated work of one engine cycle, drawn to a definite scale.

The indicator and theoretical diagrams of an engine employing combustion at constant volume are given in Fig. 2.5 and there is a schematic diagram of each stroke.

The piston, in moving from one position of rest at point B_1 (TDC), to the other position of rest at point B_2 (BDC), sucks a mixture of fuel and air into the left-hand part of the cylinder through a valve in the cylinder head (not shown in the diagram) at a constant pressure (slightly below the pressure of the outside air, i.e., the atmosphere); this is the **suction stroke** that on the diagram is represented by the horizontal line *ed.*

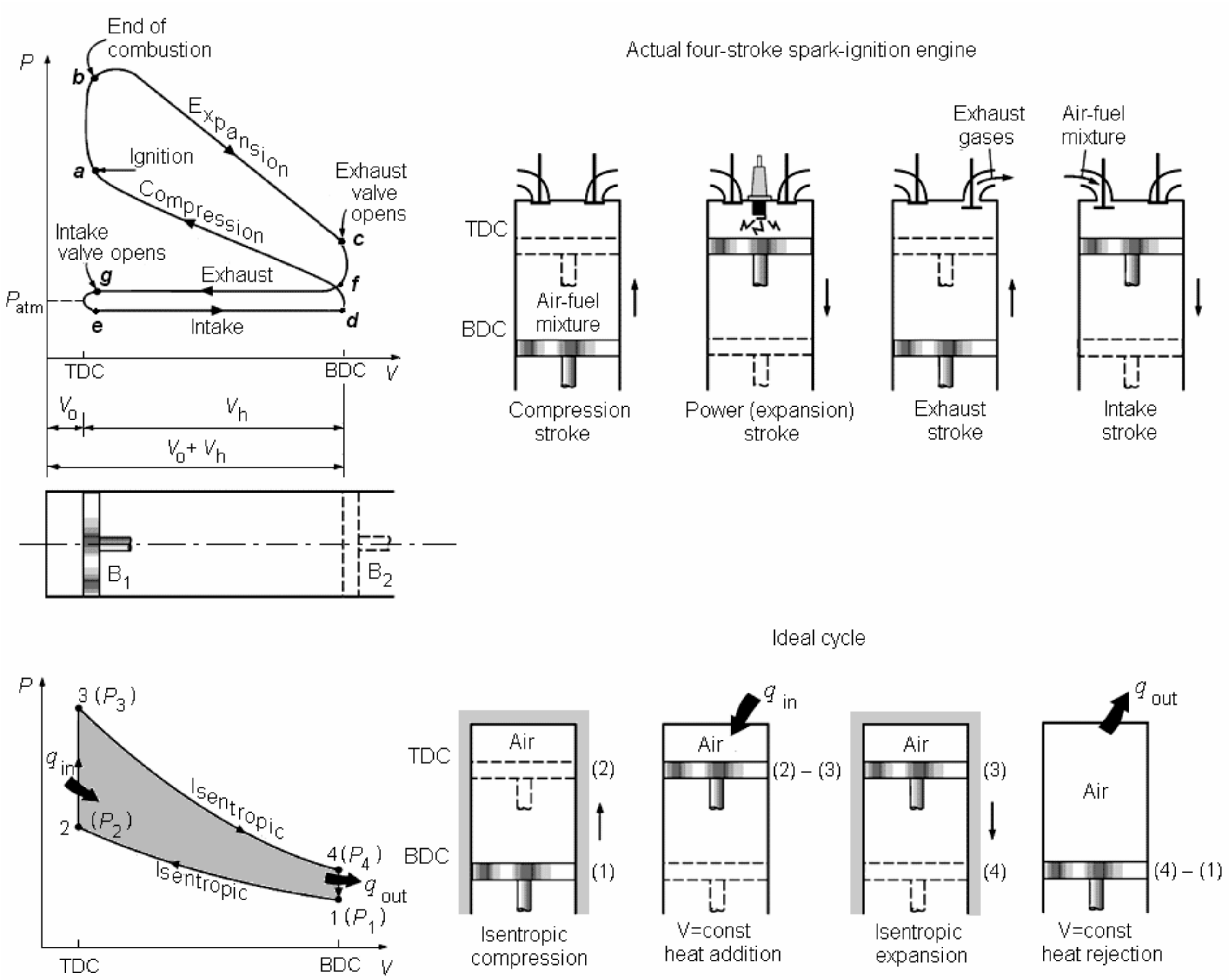


Fig. 2.5. Indicator and p-V diagram for constant volume combustion cycle

When the piston reaches BDC at point B_2, the inlet valve closes and the piston, moving to the left, compresses the gas in the cylinder according to adiabatic curve *da*, the pressure increases from p_1, at BDC to p_2 at TDC. This is the **compression stroke**; the adiabatic curve *da* indicates a change in the state of the gas because it refers to the same weight of the mixture in the cylinder throughout its entire length. Shortly before the piston reaches its position B_2 (TDC), the mixture in the cylinder is ignited by an electric spark. Theoretically combustion takes place at constant volume process, i.e. instantaneously. Owing to this combustion has the character of an explosion, the pressure in the cylinder increases from p_2 to p_3 along the vertical line *ab* showing a change in the state of the working body.

The explosion forces the piston to the right and the combustion products expand to the end of the stroke along adiabatic *bc,* and their pressure simultaneously drops from p_3 to p_4. The moving piston in turn forces the crankshaft to rotate, producing a useful work output during the **expansion** or **power stroke**, the adiabatic *bc* shows the change in the state of the combustion products. When the piston reaches BDC, a valve in the piston-head opens, connecting the interior of the cylinder with the outside atmosphere and theoretically the pressure in the cylinder instantaneously drops from p_4 to the pressure slightly above the atmospheric pressure, part of the combustion products exhausts of the cylinder. This process is shown on the diagram by the vertical line *cf*.

When the exhaust valve opens, the piston, moving to the left, ejects the combustion products at constant pressure, their volume, V_h being equal to the volume of the cylinder (the product of the interior cross-section area and the length of the piston stroke). This is the **exhaust stroke** (the horizontal line *fg* on the indicator diagram).

When the piston reaches TDC, the exhaust valve closes and the inlet valve opens, permitting a fresh portion of the air-fuel mixture to enter the cylinder*,* i. e., for the next cycle. Notice that pressure in the cylinder is slightly above the atmospheric value during the exhaust stroke and slightly below during the intake stroke.

The shaded area *1–2–3–4–1* of ideal cycle $p - V$ diagram (see Fig. 2.4) represents the magnitude of net work done in one engine cycle. In the given engine the complete working cycle is completed in four strokes, or two revolutions of the crankshaft. In this engine heat is imparted to the working body only during the combustion process *ab*, its quantity being $q_1 = c_v (T_3 - T_2)$, and is rejected only during process *fg*, the quantity being $q_2 = c_v(T_4 - T_1)$, where q_1 and q_2 are the input and output heat, c_v is the specific heat of a working body at constant volume T_1, T_2, T_3, and T_4 are the temperatures of working body in points *1*, *2*, *3* and*,* accordingly.

Theoretically net work of the cycle may be determined as the difference of q_1 and q_2**.** But the more suitable is determination of the cycle work using indicated power. The power developed inside the cylinder of the engine is called the indicated power (IP).

The indicated power produced by an engine is the power calculated from the indicated mean effective pressure (m.e.p.) p_m. The mean effective pressure is a fictitious pressure which, if it acted on the piston during the entire power stroke, would produce the same amount of net work as that produced during the actual cycle. Indicated power is the power developed without reference to friction losses within the engine. For a four-stroke-cycle engine the indicated horsepower can be calculated from the following formula, in which the letter symbols in the numerator are arranged to spell the word «plank» to assist memorizing the formula:

$$\text{INDICATED HORSEPOWER} = \frac{\text{PLANK}}{33{,}000} \text{ [h.p.]},$$

where P = Indicated mean effective pressure in p.s.i.; L = Length of the stroke in ft. or in fractions of a foot; A = Area of the piston head or cross-sectional area of the cylinder, in sq. in.; N = Number of power strokes per minute; $\mathrm{N} = \frac{\text{r.p.m.}}{2}$; K= Number of cylinders.

In the formula above, the product of the piston area and the indicated mean effective pressure gives the force (in pounds) acting on the piston. This force multiplied by the length of the stroke in feet gives the work performed in one power stroke, which, multiplied by the number of power strokes per minute, gives the number of ft.-lb. per minute of work produced by one cylinder. Multiplying this result by the number of cylinders in the engine gives the amount of work performed, in ft.-Ib., by the engine. Since hp. is defined as work done at the rate of 33,000 ft.-lb. per min., the total number of ft.-lb. of work performed by the engine is divided by 33,000 to find the indicated horsepower**.**

Example

Given: Indicated mean effective pressure (p_m) = 165 Ibs./sq. in.

Stroke (L) = 6 in. or .5 ft.

Bore = 5.5 in. (A = $1/4 \cdot \pi D^2 = 1/4 \cdot 3.1416 \cdot 5.5 \cdot 5.5 = 23.76$ sq. in.)

R.P.M. = 3,000 $\left(\mathrm{N} = 1/2 \text{ r.p.m.} = 1/2 \cdot 3000 = 1500 \text{ r.p.m.}\right)$

No. of cylinders (K) = 12.

Now, substituting in the formula, may calculate:

$$\text{Indicated h.p.} = \frac{165 \cdot 0.5 \cdot 23.76 \cdot 1500 \cdot 12}{33.000} = 1069.20 \text{ h.p.}$$

If we use other physical units then the formula for calculation of indicated power [KWt] is:

$$\text{IP} = \frac{p_m V_h n K}{122.5},$$

where p_m = indicated mean effective pressure in MPa; V_h = displacement volume of all cylinders in dm^3; n = rotations per minute; K = number of cylinders. Thermal efficiency of piston engine cycle can be definite as:

$$\eta_t = 1 - \frac{q_2}{q_1} = 1 - \frac{c_v \left(T_4 - T_3\right)}{c_v \left(T_1 - T_2\right)}.$$

Making the necessary substitutions in the efficiency expression, we get

$$\eta_t = 1 - \frac{1}{\varepsilon^{k-1}},$$

where ε — the adiabatic compression ratio; c_v — constant volume specific heat of working body; $k = c_p\ c_v$ — specific heat ratio; c_p — constant pressure specific heat of working body.

For hot gases in the cylinder k may be taken as 1.33.

As we see from the equation, the thermal efficiency of the cycle is fully determined by the adiabatic compression ratio (ε) and increases as the latter increases.

The thermal efficiency as the function of the compression ratio at k = 1.33 is given in the table:

ε	4	5	6	7	8
η_τ	36.7	41,2	44.6	47.4	49.6

Theoretic thermal efficiency of the cycle does not take into account friction and other losses of energy during engine operating. More successfully it may be done by means of Total Indicated Efficiency of engine. This coefficient for to-day's piston engines is about 25…30 %.

The development and perfection of internal combustion engines has always been connected with the problem of increasing the compression ratio; in the early eighties of the last century, pressure at the end of the compression stroke was no more than 0.2–0.25 MPa, whereas today it reaches 1.2 MPa and more with certain generator and blast-furnace gases. The composition of the combustible mixture, however, determines the limit to which ε may be increased because an increase in adiabatic compression gives a simultaneous rise in temperature from T_1 to T_2, which must always be below the temperature of spontaneous combustion in order to prevent the explosion occurring spontaneously before the piston reaches the end position. Such premature ignition (named detonation) is not only detrimental to the work of the engine from the standpoint of heat conversion but may lead to breakages or to the engine coming to a standstill.

The cycles of gas turbines and jet engines. A simple layout chart of gas turbine engine is shown in Fig. 2.6.

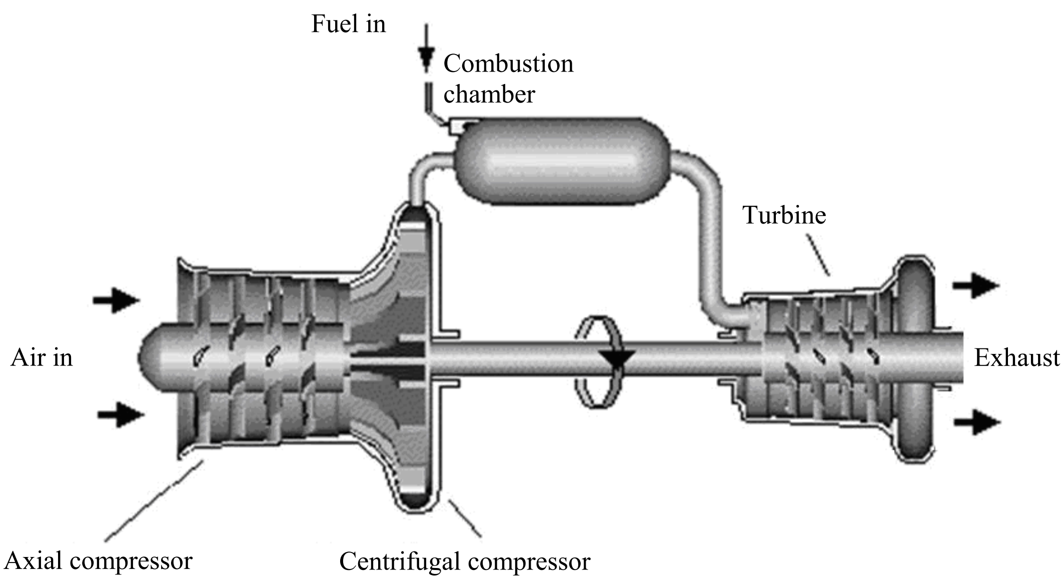
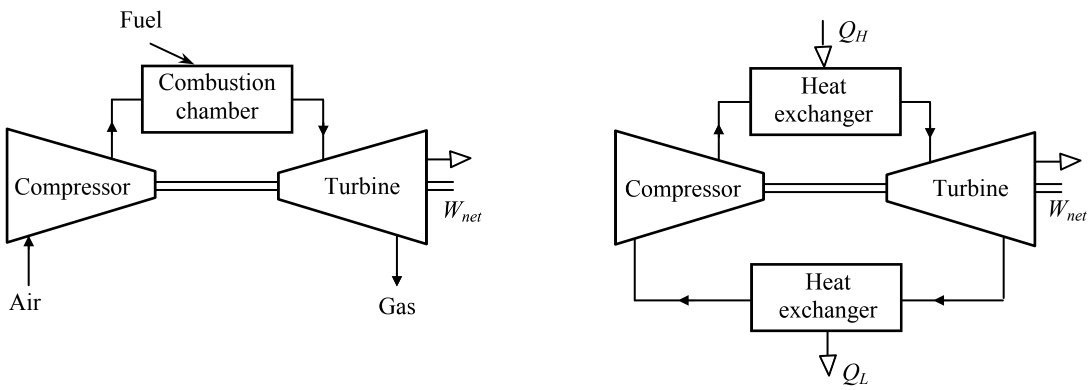


Fig. 2.6. Layout chart of gas turbine engine

The gas-turbine operates on the principle of the Brayton cycle (or Joule cycle). In practice, real cycle is "open" cycle (Fig. 2.7, *a*), where the working fluid is first compressed then it enters the combustion chamber where fuel is injected and air-fuel mixture is burned at constant pressure, and then products of combustion expand in the turbine and leave the engine (not recirculated). This is the way a jet propulsion cycle works. Fig. 2.7, *b* shows the alternative, a closed cycle, which recirculates the working fluid.

The Brayton cycle is the air standard cycle for the gas turbine engine. Air is first compressed reversible and adiabatically, heat is added to it at constant pressure, the air expands in the turbine adiabatically, then heat is rejected from the air at constant pressure to return it to the initial state.



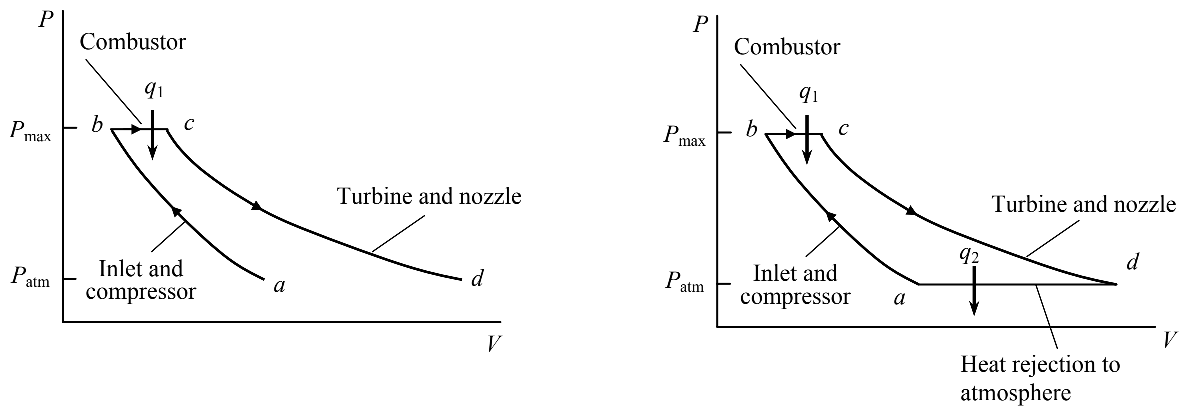
a *b*

Fig. 2.7. Open and closed cycles

Brayton cycle is similar to the four-stroke piston engine cycle. The cycle processes are intermittent in the case of the piston engine whilst they occur continuously in the gas turbine. However, in the gas turbine engine, combustion occurs at a constant pressure, whereas in the piston engine it occurs at a constant volume.

The air entering the inlet duct of the engine is at essentially ambient pressure. The portion of the curve *ab* (Fig. 2.8, *a* and *b*) shows the air pressure rise in an inlet and other portion of the curve *ab* shows the air pressure rise in a compressor.

The pressure rises from ambient at the inlet to the maximum pressure at the compressor exit plane. When energy is added to the air from the fuel burned in the combustors (line *bc*), the pressure remains relatively constant (it is because of this characteristic that the Brayton cycle is called a constant pressure cycle), but you will notice that the volume increases greatly.



a *b*

Fig. 2.8. State diagram of a gas turbine engine and Brayton cycle on *p-v* plot

When the heated air leaves the combustion chamber, it passes through the turbine where the pressure drops. But volume of the air continues to increase as we see in the portion of the curve *cd*.

The products of combustion expand greatly, and since there is little opposition to the flow of these expanding gases as they leave the engine, they are accelerated greatly. Some of the energy is extracted from the exiting gases by the turbine and this is used to drive the compressor and the various engine accessories.

After the gas leaves the turbine, it passes through an exhaust system where the pressure continues to drop to ambient and the velocity continues to increase. In this cycle, work is accomplished by increasing the velocity of the air as it passes through the engine.

2.6. Thrust Calculations

Thrust is the force which moves any aircraft through the air. Thrust is generated by the propulsion the aircraft. Different propulsion systems develop thrust in different ways, but all thrust is generated through some application of Newton's third law of motion. For every action there is an equal and opposite reaction. In any propulsion system, a working fluid is accelerated and the reaction to this acceleration produces a force.

The thrust of a turbojet engine is produced by compressing air in the inlet and compressor, mixing the air with fuel and burning in the combustor, and expanding the gas stream through the turbine and nozzle. The net thrust delivered by the engine is the result of converting internal energy to kinetic energy.

The thrust, produced when the engine is not in motion, is called **Gross thrust** and denoted by the symbol F_g. The acceleration of the gas within the engine is the difference in velocity between the air in the inlet duct, and the air as it leaves the exhaust nozzle. We can calculate the gross thrust by the formula:

$$F_g = \dot{m}_a \frac{(v_2 - v_1)}{g},$$

where F_g — Gross thrust in pounds; $\dot{m}_a$ — mass flow rate of air in pounds per second; v_1 — initial velocity of the air in feet per second; v_2 — final velocity of the air in feet per second; g — constant for acceleration due to gravity (9.81 meter per second^2 or 32.2 feet per second^2).

Let's see the way this formula works: assume that there is a business jet airplane on the runway with the engine producing takeoff power. However, the airplane has not yet begun to roll. The compressor is pulling 50 pounds of air per second through the engine (this is $\dot{m}_a$). Since the airplane is not moving, v_1 is zero, but we will assume the exhaust velocity v_2 to be 1.300 feet per second.

$$F_g = \dot{m}_a \frac{(v_2 - v_1)}{g} = 50 \ \frac{(1300 - 0)}{32{,}2} = 2{,}0866 \text{ pounds.}$$

Net thrust. When an aircraft is flying, inlet air has an initial momentum and the velocity change across the engine will be greatly reduced. We can consider the same airplane whose gross thrust we have just computed as flying at velocity 500 miles per hour (734 feet per second). Its net thrust can be found by using the same formula, only this time there is an initial velocity of air

$$F_n = \dot{m}_a \frac{(v_2 - v_1)}{g} = 50 \ \frac{(1300 - 734)}{32{,}2} = 878{,}9 \text{ pounds.}$$

From this consideration we see thrust equation indicates that net thrust equals gross thrust minus ram drag (ram drag is associated with slowing down the free stream air as air is brought inside the aircraft).

2.7. Efficiencies

One of the main measures of turbine engine efficiency is the amount of thrust produced or generated, divided by the fuel consumption. This is called **thrust specific fuel consumption,** or TSFC. The TSFC is the amount of fuel required to produce 1 Ib [0,004 45 kN] of thrust and can be calculated as follows:

$$TSFC = \frac{\dot{m}_f}{F_n},$$

where $\dot{m}_f$ — mass flow rate of fuel, [Ib/h] kg/h; F_n – net thrust, [Ib f] kgf.

This leads to the conclusion that the more thrust obtained per pound of fuel, the more efficient the engine is. Specific fuel consumption is made up of a number of other efficiencies. The two major factors affecting the *TSFC* are propulsive efficiency and cycle efficiency.

Propulsive efficiency is a measure of the performance of a thrust-producing device. This efficiency is defined to describe the transfer of kinetic energy in the exhaust stream to thrust power. It is the ratio of the thrust power to the jet power (the ratio of the useful power output to the total power output of the propulsion system):

$$\eta_p = \frac{\text{thrust power}}{\text{jet power}}.$$

In other words, the propulsive efficiency is the percentage of the total energy made available by the engine which is effective in propelling the engine. A comparison of the propulsive efficiencies of various types of gas turbine engines is shown in Fig. 2.9.

A simplified version of the propulsive efficiency formula for an unhooked engine is:

$$\eta_p = \frac{2v}{v + v_j},$$

where v_j — jet velocity at propelling nozzle, ft/s [m/s]; v — aircraft speed, ft/s [m/s].

If an aircraft is traveling at a speed (v) of 400 mph [644 km/h] and its jet velocity (v_j) was 1150 m.p.h [1 851 km/h], the propulsive efficiency can be calculated as follows:

$$\eta_p = \frac{2 \cdot 400}{400 + 1150} = 52\ \%.$$

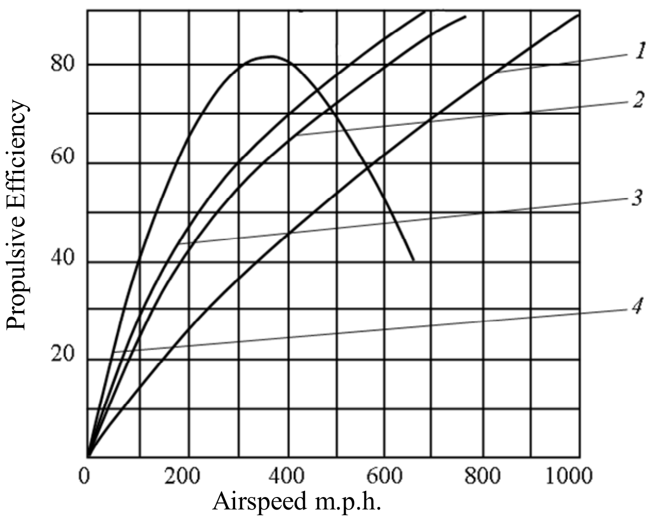


Fig. 2.9. Comparative propulsive efficiencies (Rolls-Royce)
1 — pure turbojet; *2* — bypass turbojet with low bypass ratio; *3* — bypass turbojet with high bypass ratio; *4* — turboprop

The formula for calculating propulsive efficiency for a turbofan engine using separate exhaust nozzles is

$$\eta_p = \frac{\dot{m}_{a1} v(v_{j1} - v) + \dot{m}_{a2} v(v_{j2} - v)}{\dot{m}_{a1} v(v_{j1} - v) + \dot{m}_{a2} v(v_{j2} - v) + \frac{1}{2}\dot{m}_{a1}(v_{j1} - v)^2 + \frac{1}{2}\dot{m}_{a2}(v_{j2} - v)^2},$$

where $\dot{m}_{a_1}$ — the by-pass air flow rate, Ib/s [kg/s]; $\dot{m}_{a_2}$ — the air flow rate through the core engine, Ib/s [kg/s]; v_{j1} — jet velocity of fan air at propelling nozzle, ft/s [m/s]; v_{j2} — jet velocity of core engine air at propelling nozzle, ft/s [m/s]; v — aircraft speed, ft/s [m/s].

Cycle efficiency is the amount of energy put into a usable form in comparison with the total amount of energy available in the fuel. It involves combustion efficiency, thermal efficiency, mechanical efficiency, compressor efficiency, etc. It is, in effect, the overall efficiency of the engine components starting with the compressor and going through the combustion chamber and turbine. The job of these components is to get the energy in the fuel into a form which the jet nozzle can turn into thrust.

Combustion efficiency takes into account the fact that there will be some heat loss due to radiation and conduction and that incomplete combustion might occur. Combustion efficiency is the actual thermal energy added to the working fluid divided by the thermal energy that should have been released had all the combustible constituents of fuel been completely oxidized in an adiabatic combustor.

Thermal efficiency is defined as the fraction of the heat input to a system during a cycle that is converted into net work output, or

$$\eta_{th} = \frac{\text{Net work output of the cycle}}{\text{Total heat input to the cycle}} = \frac{W_{net}}{Q_{in}}.$$

The thermal efficiency is depended on the **pressure ratio** $r_{\text{p}} = \pi = \frac{p_b}{p_a}$ or compression ratio $r = \varepsilon = \frac{v_a}{v_b} w$ and combustion temperature. Thermal efficiency increases with these parameters. Unfortunately, the turbine inlet temperature is limited by the thermal and mechanical stresses that can be tolerated by the turbine.

The development of new materials and techniques to minimize these limitations is continually being pursued.

Since $\frac{T_4}{T_1} = \frac{T_3}{T_2}$ and $\frac{T_1}{T_2} = \left(\frac{v_a}{v_b}\right)^{k-1} = \frac{1}{\varepsilon^{k-1}} = \frac{1}{r_p^{\frac{k-1}{k}}}$, we get, after the necessary substitutions (k — adiabatic exponent, for air k = 1.4; for exhaust gas k = 1.3)

$$\eta_{th} = 1 - \frac{1}{\varepsilon^{k-1}} = 1 - \frac{1}{r_p^{\frac{k-1}{k}}},$$

i. e., the efficiency of the ideal cycle is fully determined by the adiabatic compression or expansion ratio and increases as that ratio. When ε for piston engine with combustion at constant volume and gas turbine engine is identical, the efficiency of the cycles are the same, but Brayton cycle highest pressure is lower than that of the piston engine.

The average gas-turbine engine has a thermal efficiency under cruise conditions of 45 to 50 percent, whereas aircraft piston engines have 25 to 30 percent efficiency and rockets approximately 50 percent thermal efficiency.

TSFC, propulsive efficiency, thermal efficiency are typically used to express the performance of a given engine relative to other engines.

REVIEW QUESTIONS

1. What is an internal-combustion engine?
2. What kind of movement is called jet?
3. Name the strokes of a reciprocating engine.
4. Call the main parts of the reciprocating engine.
5. What is the position of the piston when it forms the smallest volume in the cylinder?
6. State the four processes of a gasoline engine.
7. Explain the operation of a piston engine?
8. What are three basic components of gas turbine engine?
9. What is the air cycle of a gas turbine engine?
10. What are four processes of a gas turbine engine?
11. Draw a schematic diagram and *p-v* diagram of an open cycle gas turbine.
12. Discuss the merits and demerits of the constant volume combustion cycle and constant pressure combustion cycles applied to reciprocating and rotating plants.
13. Define the propulsive power and propulsive efficiency.
14. What is compression ratio?
15. What is pressure ratio?
16. What is influence of pressure ratio increase on the gas turbine cycle thermal efficiency?
17. Can the thermal efficiency of turboprop engine be equal to 100 %?
18. What is the difference between propulsive efficiency and energy conversion efficiency of the gas turbine?
19. What is the gross thrust?
20. How is the thrust force generated by jet propulsion?

Chapter 3. GAS-TURBINE ENGINE PROGRESS

3.1. Modern achievements

The gas-turbine engine can be used as a **turbojet engine** (thrust developed by exhaust gases alone), a **turbofan engine** (thrust developed by a combination of exhaust gases and fan air), a **turboprop engine** (in which a gas-turbine engine-turns a gearbox for driving propellers), and a **turboshaft engine** (which drives helicopter rotors). Small gas-turbine engines called **APU's** (auxiliary power units) have been developed to supply transport-category aircraft with electrical and pneumatic power. Although these units can be used both on the ground and in the air, they are mainly used on the ground.

Gas-turbine engines are used for propulsion in many different types of aircraft. These aircraft include airliners, business aircraft, training aircraft, helicopters and agricultural aircraft. During the period of development of the gas-turbine engine, designers and engineers faced many challenges. Designers are constantly striving to improve on gas-turbine engines performance, sound levels, fuel efficiency, and ease of maintenance, durability and reliability. Many different types of gas-turbine engines have been developed to meet the overall propulsion needs of the civil, transport and military aircraft. An example of an engine which incorporates many of the learned technologies needed to provide excellent fuel consumption, low emissions and a lot of characteristics mentioned earlier, is the GE90, the most power engine in the word.

This engine is a **high bypass turbofan engine** which incorporates a fan with a diameter in excess of 10 ft [3 m] and has a thrust range of 72.000 to 95.000 Ib [32660 to 43090 kg] of takeoff thrust. This engine is designed for the airliner Boeing-777.

The most powerful engine in former USSR is Д-18 – a bypass engine, which was developed by the Ivchenko Zaporizhia PROGRESS design bureau for airliners Ан-124 "Ruslan" and An-225 "Mrij" – the biggest in the world cargo airplanes (Fig. 3.1). Six such engines, installed in the An-225 "Mrij" airliner can create thrust value more than 1300 kN (130.000 kg).

Fig. 3.1. The Д-18 engines power "Ruslan" and "Mrij" aircraft

Today it will continue to see improvements in reliability, efficiency and emissions from gas turbine engines.

The Pratt & Whitney PW4000 model engines (see Fig. 1.16) with advantage deliver the best performance for the A330 family of aircraft. The advantage technology upgrade brings enhanced engine performance, lower fuel consumption and emissions, increased durability and reduced maintenance costs.

Rolls-Royce has concluded a preliminary agreement to supply the next-generation Trent engine for the new A350 XWB twinjet being offered by Airbus. The engine, the sixth variant in the Trent series, would be capable of powering all versions of the aircraft.

Engine design would be based on the three-shaft architecture unique to all Rolls-Royce large engines, used throughout the Trent family as well as the earlier RB211 series (Fig. 3.2–Fig. 3.5).

The 75.000–95.000 lb thrust Trent XWB is the sixth member of the three-shaft Trent engine series and is being developed specifically for the Airbus A350 XWB twinjet. It is available for deliveries from mid-2013 and will be certificated at 95.000 lbs when put into operation.

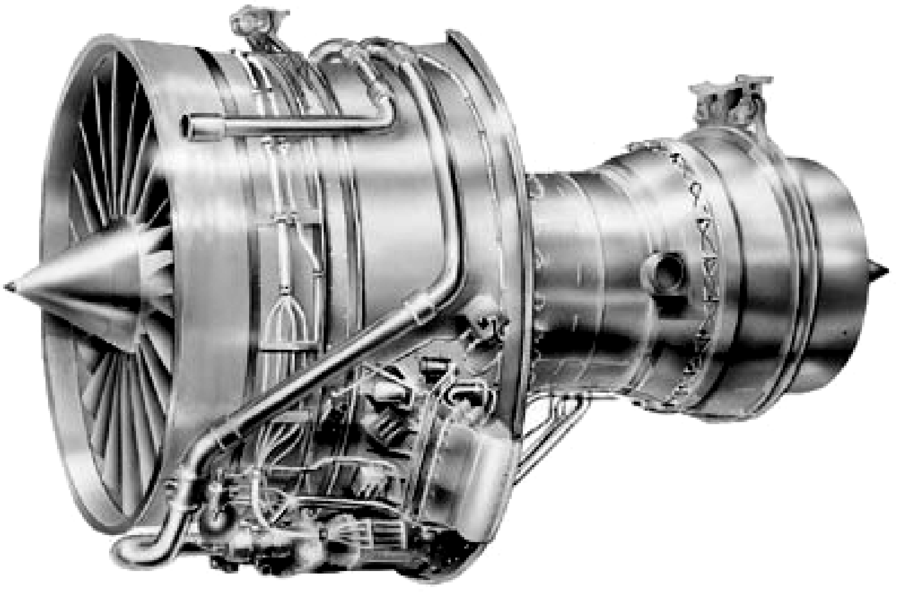


Fig. 3.2. RB211-535E4 engines power the Boeing-757 and Tupolev Tu-204 aircraft

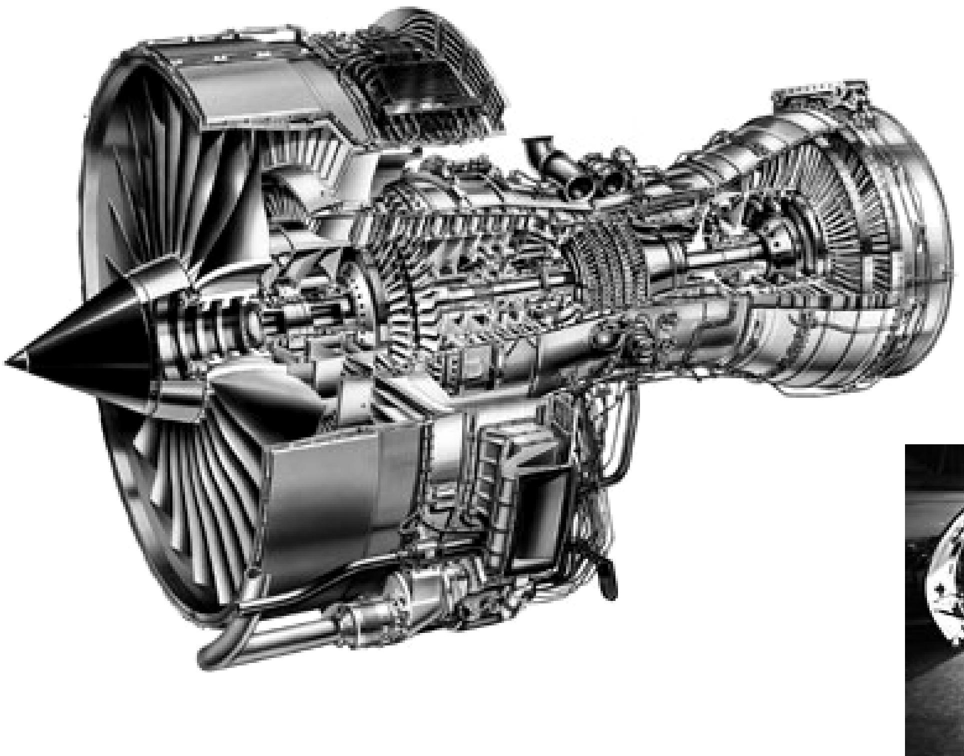


Fig. 3.3. The Trent 700 powers the Airbus A330

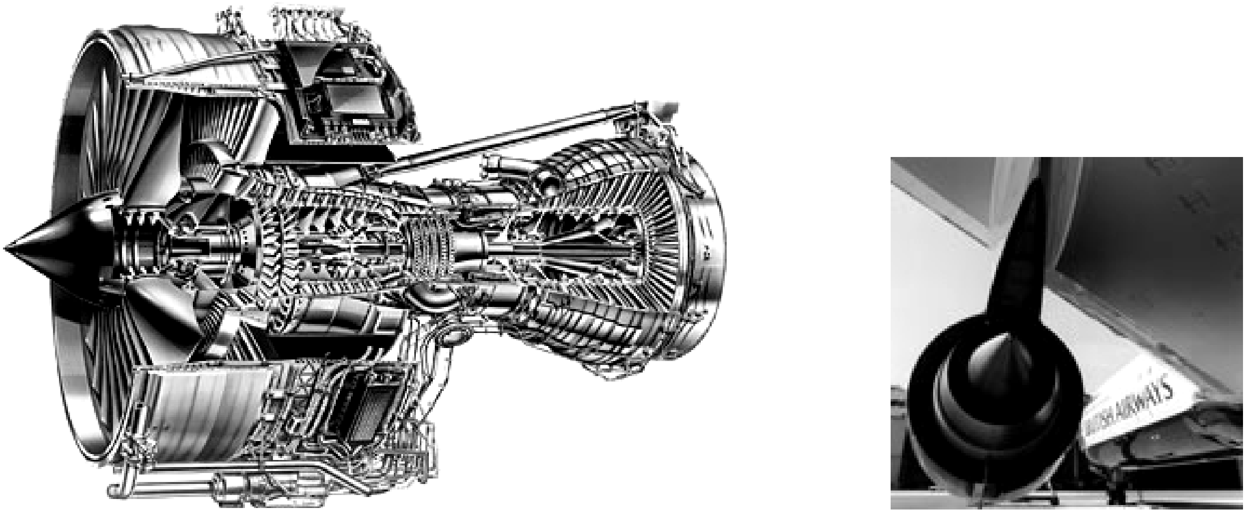


Fig. 3.4. The Trent 800 powers the Boeing-777

Fig. 3.5. The Trent XWB will power the Airbus A350 XWB

3.2. Prop-Fan Concept

The **Prop-Fan** is an advanced propulsion concept designed and developed by NASA in conjunction with various aviation manufacturers. Testing has shown that a Prop-Fan (Fig. 3.6), driven by a turbine engine, can achieve substantial fuel savings and create less airport noise while operating at typical jet speeds and cabin comfort levels.

The Prop-Fan concept combines the power of a jet engine with the efficiency of a propeller in a single new propulsion system. Unlike conventional propellers, which have two to four long, paddle-shaped blades, a Prop-Fan has eight or ten short, wide blades that are delicately curved and set close together in an overlapping pattern. The configuration resembles that of a fan. The larger number of blades in a Prop-Fan permits it to be smaller in diameter than conventional propellers.

Conventional turboprop aircraft have been limited in the maximum speed that they can attain because of the onset of propeller shock waves due primarily to the large diameter of the propeller. The smaller-diameter of Prop-Fan coupled with an improved aerodynamic design has greatly reduced the onset and effects of shock waves, which has greatly increased the speed range of the Prop-Fan.

A Prop-Fan can save significant amounts of fuel compared to a turbofan engine in the range of 450 to 550 mph [724.05 to 884,95 km/h] because it develops a higher level of propulsion efficiency.

The Prop-Fan is particularly suited for 80- to 160-passenger planes that fly short hops in the 300- to 500-mi [482.7 to 804.5 km] range, during which most of the time is spent on climbing and descending.

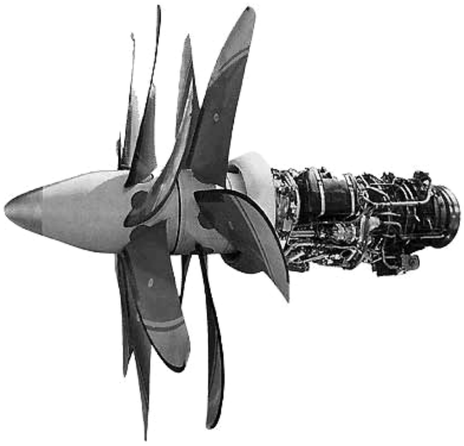
One such type of Prop-Fan engine is the Unducted fan developed by General Electric and illustrated in Fig. 3.6, *b*. This engine incorporates counterrotating turbines that drive two pusher Unducted fans. The counterrotating fans increase the Prop-Fan's ability to convert energy to thrust. The Prop-Fan blades are constructed from composite materials, allowing contours for optimum aerodynamic and acoustic efficiency.

The first in the word flying Prop-Fan engine D-27 was designed by the Ivchenko PROGRESS design bureau at the beginning of nineties. This engine is intended for transport-type airplane An-70 (Fig. 3.6, *c*). Four engines D-27 are installed on the wing of An-70 airplane and develop summer take-off thrust about 500 kN.

a

b

c



d

Fig. 3.6. Prop-Fan — new progressive type of gas turbine engines:
a — TP400-D6 on the engine test bench; *b* — Unducted fan (NASA-General Electric);
c — Antonov-70- the first in the world airplane powered by Prop-Fan engines;
d — Prop-Fan engine Д-27

Basic peculiarity of Д–27 Prop-Fan engine is the presence of two coaxial propellers (Fig. 3.6, *d*). The front propeller has 8 blades, and the rear propeller has 7 blades. This engine is top fuel economy. Its specific fuel consumption in cruise flight conditions is about 0,170 [kg/(N h)]. In addition to it the design of the Д–27 engine ensures the low level of harmful emissions and acceptable noise level.

REVIEW QUESTIONS

1. What are main advantages of prop-fan engine as compared with turboprop and turbofan engines?
2. On what aircraft for the first time in the world were installed Prop-Fan engines?

PART II

TURBOPROP ENGINE ТВ3-117ВМА-СБМ1

Chapter 4. GENERAL INFORMATION ABOUT ENGINE

4.1. Concept

The ТВ3-117ВМА-СБМ1 is an advanced, fuel-efficient and low-noise turboprop engine. It is designed to be used as a cruise engine for short-haul airplanes. The engine is derivative of the ТВ3-117 helicopter engine successfully operated in 40 countries in various climatic areas and granted the type certificate. The engine with the newly developed АВ-140 propeller is installed in the An-140 airplane and is likely to be installed in medium-haul Il-112, Il-l l4 airplanes.

This two-shaft turboprop engine (Fig. 4.1–4.4) is designed to be used in the main powerplant installed in passenger and cargo transport aircraft operated on local airlines.

The ТВ3-117ВМА-СБМ1 is a two-shaft turboprop engine with a remotely located transmission of the propeller drive. The engine consists of a turbine drive and a propeller drive transmission system.

The turbine engine drive is attached to the transmission system by means of two rods *1* (Fig. 4.1 and Fig. 4.2) in the front engine attachment mount and two rods in rear part. The transmission system, in turn, is attached to aircraft by two trunnions *3* in the front attachment mount and one trunnion *7* in the rear attachment mount. The turbine drive consists of inlet section *22* (Fig. 4.4) with an adapter *21,* compressor front bearing housing *20* with the internal drive, accessory drive gearbox *19*, compressor *17*, combustion chamber *16*, compressor turbine *15*, power turbine *13* and exhaust unit *11*.

The propeller drive transmission system consists of rear reduction gear *8*, shafting *3* and front reduction gear *1*. The specific feature of a two-shaft design is that the engine rotor is subdivided into a gas generator rotor, running in three bearings, and power turbine rotor *13*, running in two bearings.

The rotor of the gas generator and that of the power turbine are interconnected by gas-dynamic coupling, which permits to use a low-power starter for starting the engine, as during the starting procedure the starter rotates only the gas generator rotor.

The engine construction provides for assembly of modules (units). The engine is divided into seven main modules (Fig. 4.5), each of them is a self-contained assembly structurally adaptable to streamlined production and can be replaced from the engine without disassembling the adjacent modules under the conditions of an aircraft maintenance base. The module construction of the engine provides for returning the engine to its serviceable state by replacing the parts and assemblies under the operating conditions.

4.2. Brief description of ТВ3-117ВМА-СБМ1 engine unites

Inlet section *22* (Fig. 4.4) with adapter *21* is attached to the front flange of the compressor front bearing housing *20* and is intended to form the gas flow duct of the engine. The inlet section is heated by the flow of oil scavenged from the cavity of compressor *17* rear bearing and delivered to the cavity formed by the inlet section shrouds.

Compressor front bearing housing *20* is located between the inlet section and compressor *17.* The housing is intended for:

- arrangement of the compressor front bearing, the compressor variable inlet guide vane assembly (IGV), internal drive, engine accessory drive gearbox *19*, undriven accessories and the turbine drive-to-transmission system attachment bracket;
- forming the engine gas flow duct in the area between the inlet section and the compressor.

The housing of the compressor front bearing consists of two ring shells interconnected by four struts inside which the engine pipelines are routed. Housing *20* of the compressor front bearing is connected with adapter *21* by its front flange, through the turbine drive attachment bracket, and with the compressor casing by its rear flange. Attached to the lower flange of housing *20* is engine accessory drive gearbox *19.*

The engine accessory drive gearbox *14* is intended to house the following driven accessories:

- air turbine starter *18;*
- oil pump block *15* (Fig. 4.2);
- de-aerator *14* with oil scavenge pumps;
- fuel control unit *11* (Fig. 4.3);
- engine centrifugal fuel pump *9*;
- fuel drain tank *10.*

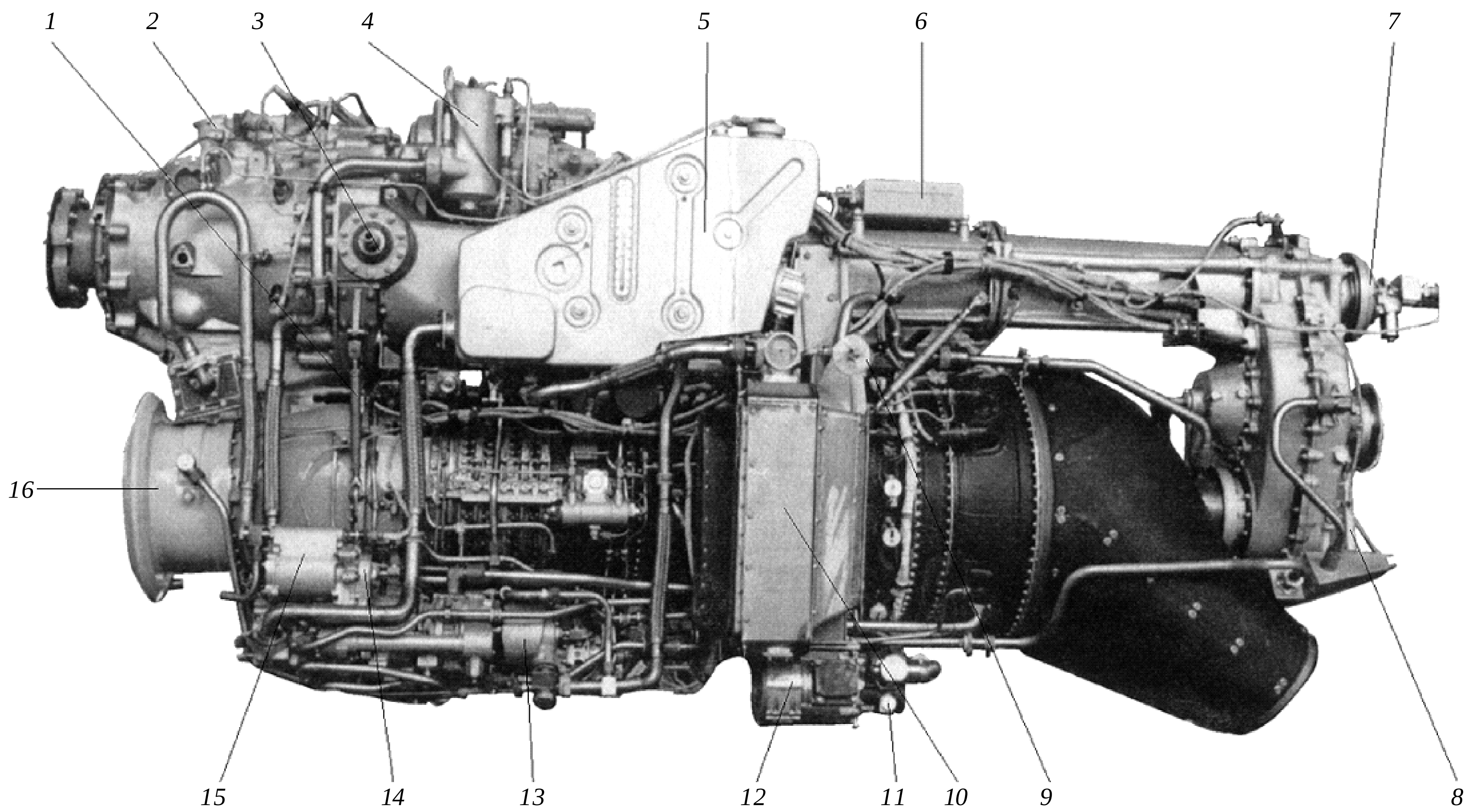


Fig. 4.1. Left-side view of ТВ3-117ВМА-СБМ1 engine:
1 — Gas Generator Drive Attachment Rod; *2* — Torquemeter Oil Pump; *3* — Engine Attachment Trunnion; *4* — Oil Filter; *5* — Oil Tank; *6* — Ignition Unit; *7* — Engine Attachment Trunnion; *8* — Engine Standby Attachment Trunnion; *9* — Air-Oil Cooler Attachment Bracket; *10* — Air-Oil Cooler (Air-Oil Heat Exchanger); *11* — Air-Oil Cooler Ejector Valve; *12* — Fuel Distributor; *13* — Air Turbine Starter; *14* — De-Aerator with Oil Scavenge Pumps; *15* — Oil Pump Block; *16* — Inlet Section

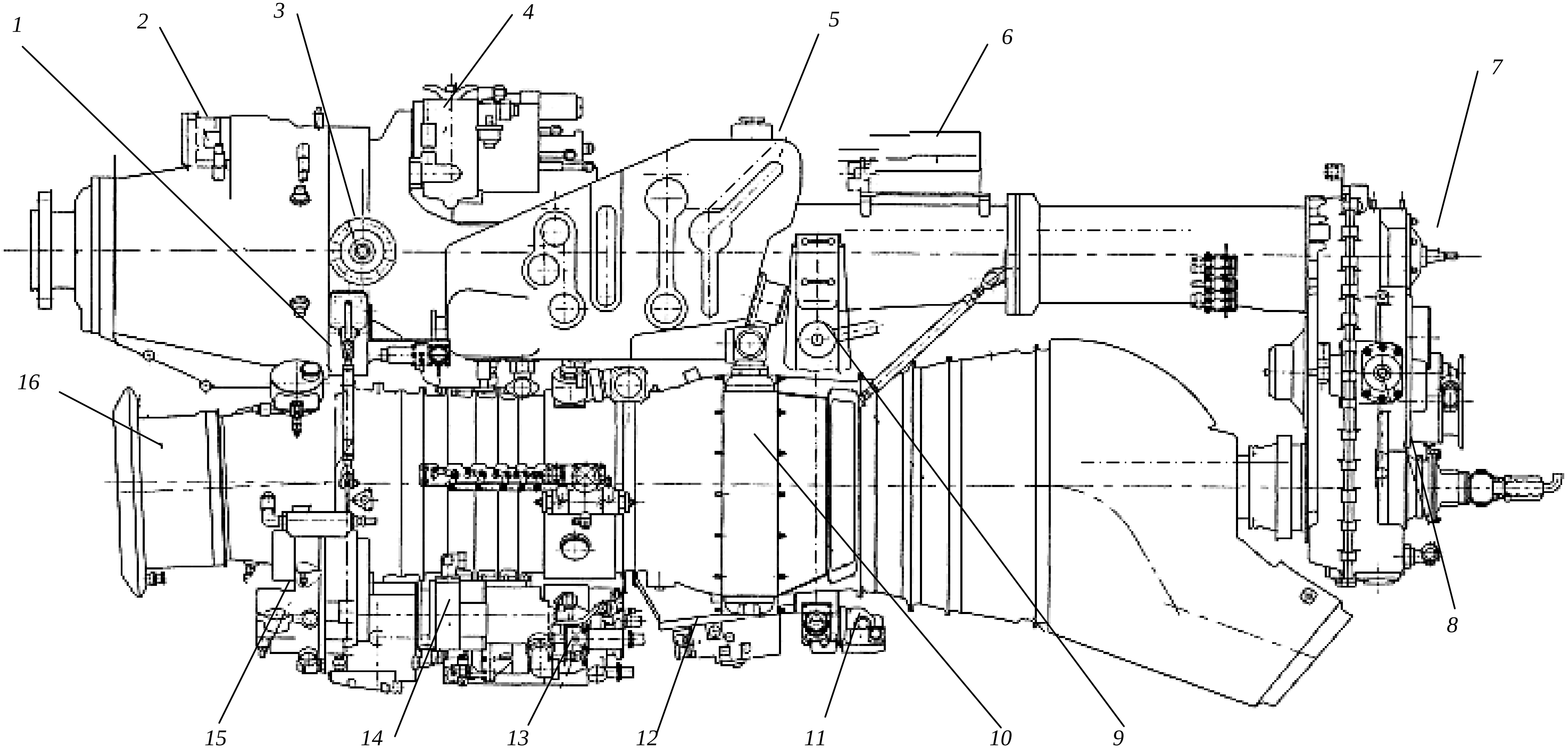


Fig. 4.2. Engine left-side view schematic diagram:
1 — Gas Generator Drive Attachment Rod; *2* — Torquemeter Oil Pump; *3* — Engine Attachment Trunnion; *4* — Oil Filter; *5* — Oil Tank; *6* — Ignition Unit; *7* — Engine Attachment Trunnion; *8* — Engine Standby Attachment Trunnion; *9* — Air-Oil Cooler Attachment Bracket; *10* — Air-Oil Cooler (Air-Oil Heat Exchanger); *11* — Air-Oil Cooler Ejector Valve; *12* — Fuel Distributor; *13* — Air Turbine Starter; *14* — De-Aerator with Oil Scavenge Pumps; *15* — Oil Pump Block; *16* — Inlet Section

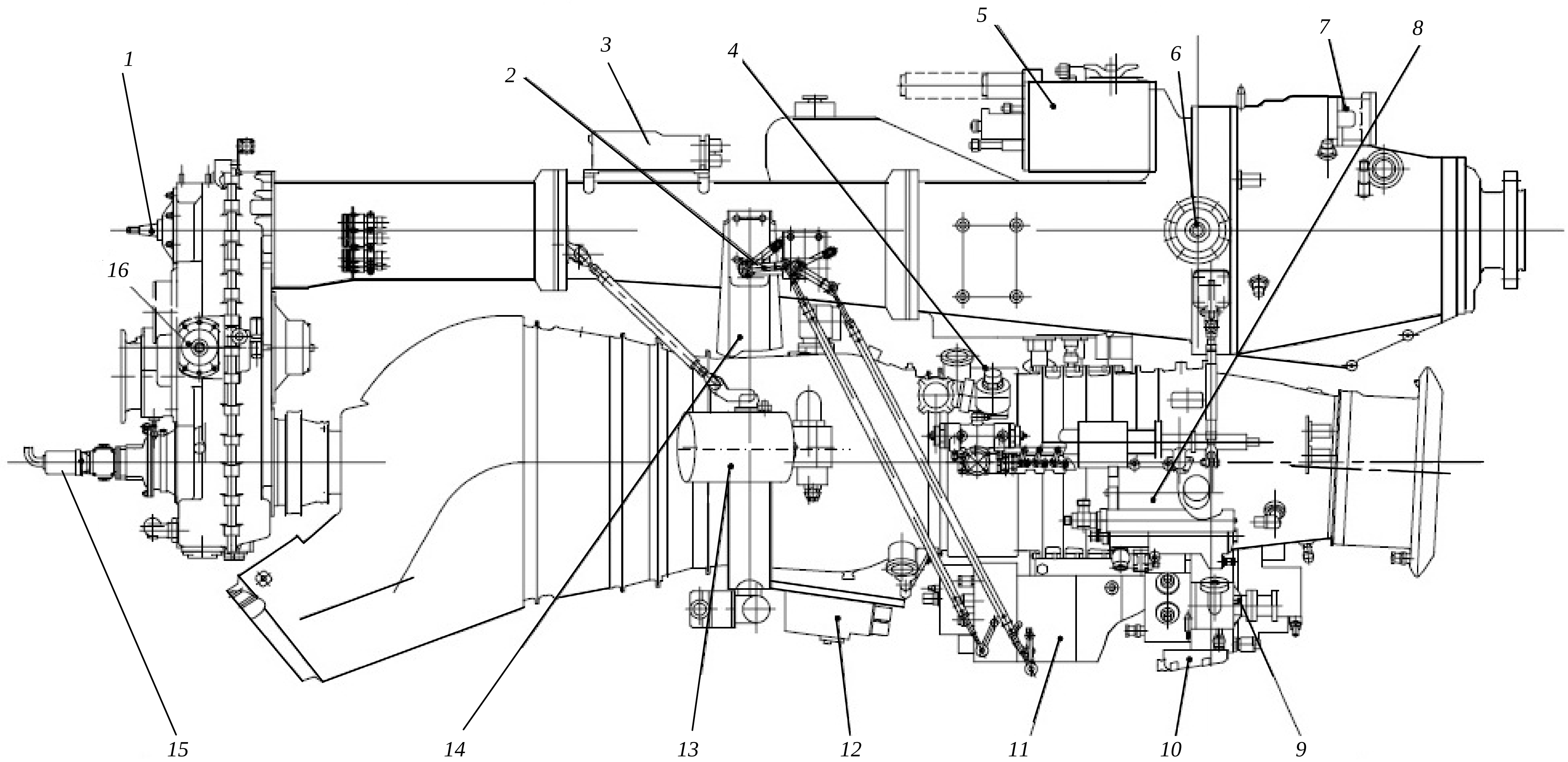


Fig. 4.3. ТВ3-117ВМА-СБМ1 Engine right-side view schematic diagram:
1 — Engine Attachment Trunnion; *2* — Engine Control System (Mechanical); *3* — Ignition Unit; *4* — Air Bleed Valve; *5* — Propeller Speed Governor; *6* — Engine Attachment Trunnion; *7* — Torquemeter Oil Pump; *8* — Fuel-Oil Heat Exchanger; *9* — Engine Fuel Centrifugal Pump; *10* — Fuel Drain Tank; *11* — Fuel Control Unit; *12* — Fuel Distributor; *13* — Feathering Pump; *14* — Air-Oil Cooler Attachment Bracket; *15* — Propeller Brake Drive; *16* — Engine Standby Attachment Trunnion

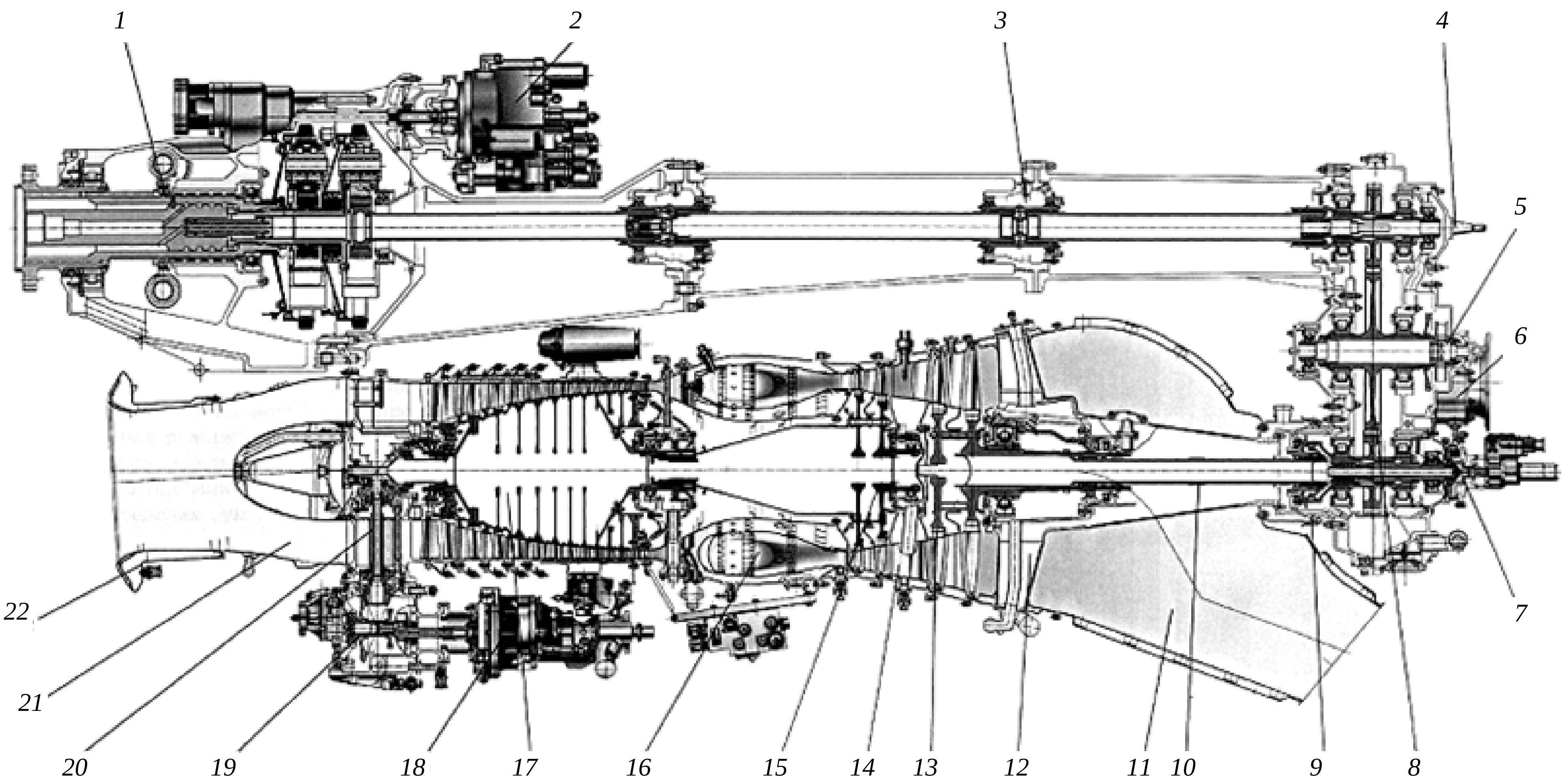


Fig. 4.4. ТВ3-117ВМА-СБМ1 Engine longitudinal section view:
1 — Front Reduction Gear; *2* — Propeller Speed Governor; *3* — Shafting; *4* — Engine Attachment Trunnion; *5* — Centrifugal Breather; *6* — Generator Drive; *7* — Propeller Brake; *8* — Rear Reduction Gear; *9* —Turbine Drive Attachment Fitting; *10* — Torsion Shaft; *11* — Exhaust Unite; *12* — Power Turbine Bearing Housing; *13* — Power Turbine; *14* — Compressor Turbine Bearing Housing; *15* — Compressor Turbine; *16* — Combustion Chamber; *17* — Compressor; *18* — Air Turbine Starter; *19* — Engine Accessory Drive Gearbox; *20* — Front Bearing Housing; *21* — Adapter: *22* — Inlet Section

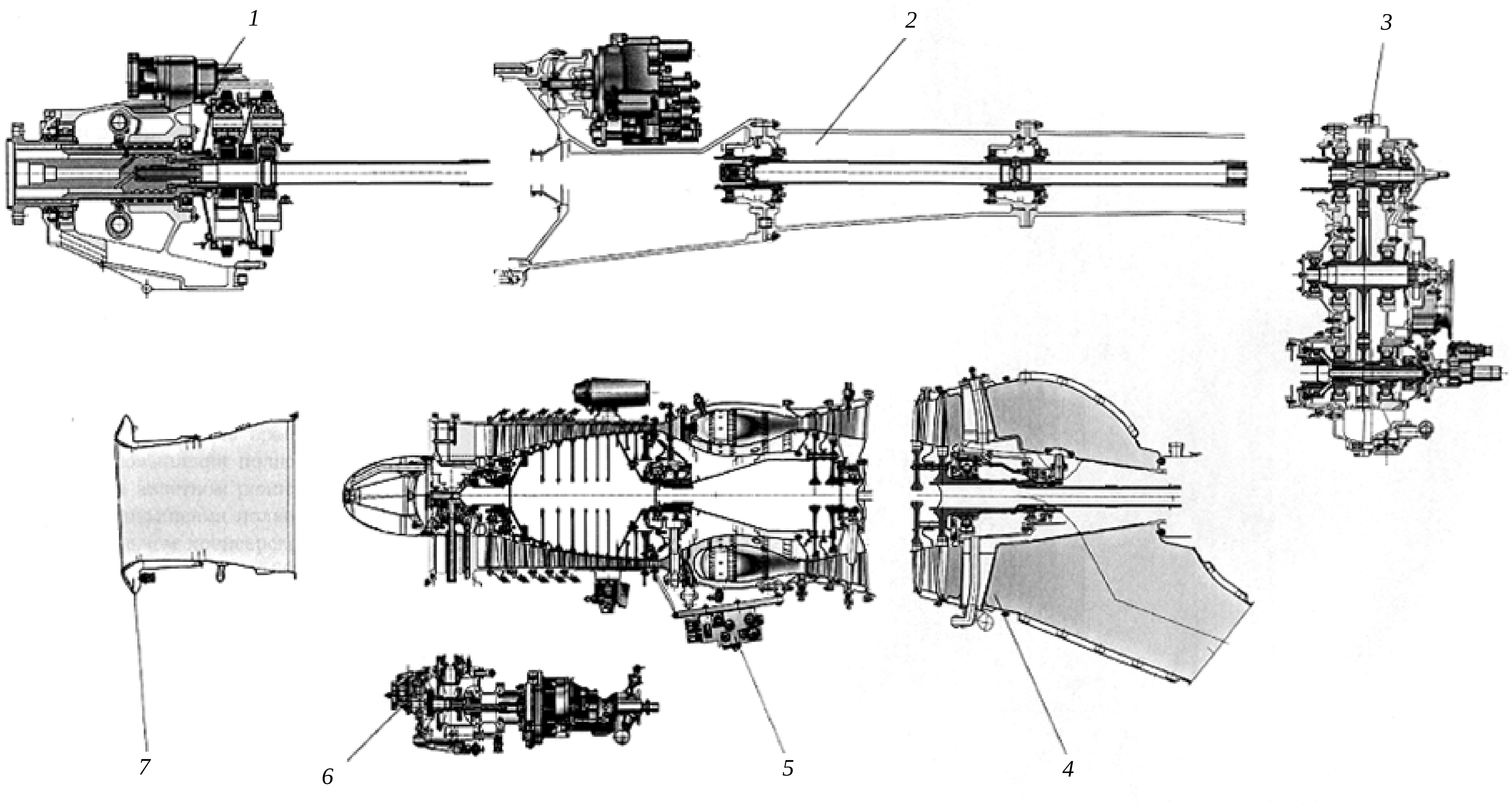


Fig. 4.5. ТВ3-117ВМА-СБМ1 Engine modules:
1 — Module 01-Front Reduction Gear; *2* — Module 02-Shafting; *3* — Module 03-Rear Reduction Gear; *4* — Module 07-Power Turbine;
5 — Module 05-Gas Generator; *6* — Module 06-Engine Accessory Drive Gearbox; *7* — Module 04-Exhaust Unit

The driven accessories are powered from the compressor rotor via the internal drive. The engine accessory drive gearbox *19* (see Fig. 4.4) also mounts two ДТА-10 gas generator rotor rotational speed sensors and the rotor manual cranking unit.

The axial-flow, subsonic, single-spool**, twelve-stage compressor** *17* consists of a stator, inlet guide vanes (IGV), rotor, the front and rear bearings. The compressor stator comprises a casing and inlet guide vanes.

The variable IGV and the variable guide vanes (VGV) of the first, second, third and fourth stages change their positions versus the engine power rating.

Arranged on the compressor casing are:

- two valves for air by-passing aft of the seventh stage;
- the anti-icing system shutter;
- three connections for air bleeding aft of the fifth stage.

The variable IGV assembly consists of separate swivel vanes installed in the housing of the compressor front bearing.

The disc-drum construction compressor rotor consists of discs welded to one another, except the first stage disc which is bolted to the adapter welded to the second stage disc. The rotor blades are secured in the discs by "dovetail" roots. The elastic front bearing of the compressor rotor is fitted with a roller bearing. The rigid rear bearing is fitted with a ball bearing. The torque from the shaft of compressor turbine *15* is transmitted to the compressor rotor through the involute splines made inside the rear shaft.

The annular combustion chamber *16* consists of outer and inner casings, flame tube, twelve dual orifice fuel nozzles and two igniter plugs. Fuel is supplied to the fuel nozzles through the fuel manifold. During the engine starting, the fuel is ignited by two semiconductor igniter plugs.

The turbine unit consists of compressor drive turbine *15*, power turbine *13* and exhaust unit *11*. The axial-flow, jet, two-stage compressor turbine comprises a stator and a rotor. The stator of the compressor turbine consists of two nozzle guide vanes (NGV), two races, front, outer and inner casings and the bearing housing. The first stage NGV is assembled of cast sectors cooled with air. The second stage NGV is a unit-cast assembly with cooled hollow vanes. The rotor of compressor turbine *15* consists of a shaft and two wheels. The rotor blades are attached by "fir-tree" roots. The disc bodies, vane roots and feet are blown with combustion chamber *16* bled air. The front bearing of compressor turbine *15* rotor is the rear end of compressor *17* rotor, on which the rotor centering end rests. Used as the rear bearing is a roller bearing, installed in compressor turbine bearing housing *14*. Arranged on the bearing housing are twelve thermocouples and thermocouple probes.

Power turbine *13* is an axial-flow, reaction, two-stage assembly, which consists of a rotor and a stator. The stator comprises a casing, two NGV and two races. The casing of the power turbine stator is of a welded structure. The first and second stage NGV are unit-cast assemblies with hollow vanes. The power turbine rotor consists of a shaft and two rotor wheels. The rotor blades are secured by "fir-tree" roots. The discs of the first and second stages are cooled with air bled aft of the compressor seventh stage.

The power turbine rotor is of a cantilever-mounted, two-bearing type. Used as the front bearing is a ball bearing, and the rear bearing is a roller bearing. The ball and roller bearings are installed in housing *12* of the power turbine bearings. The rear flange of the power turbine bearing housing is secured to turbine drive-to-transmission attachment fitting *9*. Exhaust unit *11* is used for discharging the exhaust gases overboard.

The torque from power turbine *13* is transmitted via torsion shaft *10* to rear reduction gear *8* of the engine transmission. The propeller drive transmission system is used to transmit the power turbine torque to the propeller shaft and to provide the preset propeller rotational speed. The transmission system comprises rear reduction gear *8*, shafting *3* and front reduction gear *1*.

The rear reduction gear is made as a simple cylindrical gear train and intended for providing the preset distance between the axes of the turbine drive and the propeller. The rear reduction gear consists of a casing and three gears, each being mounted on two roller bearings and one ball bearing.

Attached to the rear reduction gear casing are:

- turbine drive-to-transmission attachment fitting *9* trunnion;
- engine attachment trunnion *4* and two engine standby attachment trunnions *6* (see Fig. 4.3);
- propeller brake *7* (see Fig. 4.4).

Shafting *3* is used to transmit the torque from rear reduction gear *8* to front reduction gear *1*. The shafting comprises three casings, two bearing housings, two torsion shafts and two splined couplings. The torque is transmitted by means of the torsion shafts through the splined couplings, each mounted on two ball bearings.

Attached to the shafting are:

- the brackets of the main connectors of the electric wiring harness;
- attachment brackets of air-oil cooler *9* (see Fig. 4.1), air-oil cooler ejector valve *11* and feathering pump *13* (Fig. 4.3);
- two engine attachment trunnions *1* (see Fig. 4.1);
- ignition unit *6,* oil filter *4,* propeller speed governor *5* (see Fig. 4.3);
- engine control system (mechanical) *2* acting on the levers of fuel control unit *11*.

Front reduction gear *1* (see Fig. 4.4) of a two-stage, differential planetary, closed type provides the main reduction of the power turbine rotational speed and the torque transmission to the propeller. The front reduction gear consists of a casing, the first stage driving gear installed on the driving torsion shaft splines, the first stage planetary gear casing with three planetary gears, the second stage driving gear, two rim gears, two hubs, the second stage gearing train casing with five intermediate gears, propeller shaft, the torque meter assembly, propeller control channel oil by-pass unit, torquemeter oil pump drive gear, two toothed sectors, two cylindrical springs, a rack, bearing casing and three push rods.

The planetary gear casing is mounted on two ball bearings, the planetary and intermediate gears are running in the roller bearings, and the propeller shaft — in a roller and a ball bearings. The torque, from the shafting is transmitted through the driving torsion shaft to the reduction gear, where it increases and is transmitted to the propeller through the propeller shaft. According to the measured oil pressure in the torquemeter assembly, the automatic engine control system calculates the value of the propeller shaft torque.

The torquemeter assembly consists of a collector and six cylinder-piston groups, which sense the reactive torque of the gearing train casing.

Arranged on the reduction gear casing are:

- torquemeter oil pump *2* (see Fig. 4.2);
- the propeller blade angular setting sensor;
- the vibration pickup;
- two propeller rotational speed sensors;
- the propeller rotary phase sensor;
- four transmitters of oil pressure in the propeller control channels and the torque-meter channel.

4.3. Brief description of main engine systems

The engine is provided with a self-contained, circulation **oil system.** The engine oil system provides continuous oil supply under pressure to the frictional surfaces of the rotor bearings, the rotating parts of the reduction gears, internal drive and accessory drive gearbox. In addition to this, the oil system provides oil supply to the propeller control system and the torquemeter system.

The engine oil system comprises the following main units:

- oil tank *5* (see Fig. 4.2);
- oil pump block *15*;
- de-aerator *14* with oil scavenge pumps;
- oil filter *4*;
- torquemeter oil pump *2*;
- air-oil cooler *10*;
- the air-oil cooler ejector, transmitters, warning devices and valves, which ensure the oil system operation and monitoring.

The breathing of the oil-containing spaces of the rotor bearings and the accessory drive gearbox is performed through the high-delivery oil scavenge pumps. The oil-containing space of the power turbine bearings communicates with the transmission oil space, and the oil space of the accessory drive gearbox communicates with the spaces of the transmission and the oil tank.

The engine transmission oil-containing spaces breathing is accomplished through centrifugal breather *5* (see Fig. 4.3) into the exhaust device space. Installed in the breather air discharge pipeline is an ejector, which creates vacuum in the transmission chamber at all engine power ratings.

The engine is provided with an automatic internal power **starting system.** The gas generator rotor rotation is accomplished by air turbine starter *18* installed on engine accessory drive gearbox *19*.

The sources of compressed air, required for the engine starting are:

- the airborne auxiliary power unit (APU);
- the other earlier-started engine;
- a ground pneumatic power source.

During starting the engine operation is controlled by the automatic engine control system (AECS) in accordance with the starting procedure sequence.

The fuel system provides fuel supply to the engine in the quantity determined by the AECS versus the position of the throttle control lever (TCL) and the flight conditions.

The system includes the following main components:

- engine centrifugal pump *9* (see Fig. 4.3);
- fuel-oil heat exchanger *8*;
- fuel control unit (FCU) *11*;
- fuel distributor *12*.

The fuel system units provide:

- automatic fuel flow metering within the entire range of the engine power ratings;
- maintaining of the preset engine power rating with variation in the flight conditions (altitude, airspeed, ambient air temperature).

The engine drainage system provides collection and discharge of drainage fluids leaking through the seals of the fuel and oil system accessories and through the adapters of the combustion chamber and the generator. The drainage fluids from the fuel and oil system accessories flow by gravity through the pipelines into the drain tank, secured on the lower wall of the engine accessory drive gearbox, and discharged into the exhaust unit.

Oil, leaking through the seal of the generator drive adapter, is discharged through the pipelines into the exhaust unit. The automatic engine control, monitoring and fault diagnosis system performs the functions of automatic control of the engine operation parameters, systems and the propeller. The **automatic engine control system (AECS)** consists of the main and standby automatic engine controls.

The main AECS is an electronic-hydromechanical system, the standby AECS is a hydro-mechanical one. Changeover from the main AECS to the standby one is accomplished automatically. The AECS incorporates:

- an electronic engine control (monitor) (EEC);
- a fuel flow control unit (FCU);
- a propeller speed governor (PSG);
- a feathering pump (FP);
- a fuel distributor (FD);
- transmitters and indicators (warning devices) of the operation parameters of the engine, propeller and their systems.

The AECS provides:

- fuel supply and automatic control of the engine operation parameters according to the preset laws at all transient and steady engine power ratings;
- supply of signals, proportional to the engine operation parameters, to the control and monitoring unit and to the switching and starting unit, installed in the aircraft;
- the engine protection from the power turbine rotor overspending and from expedience of the maximum allowable gas temperature.

The engine power rating is set by means of the throttle control lever from the flight compartment. To shut down the engine, use is made of the FUEL ON/OFF (СТОП КРАН) selector switch or FEATHER STOP (ОСТАНОВ ФЛЮГЕР) switch-light installed in the flight compartment.

The throttle control lever acts on the lever of the fuel control unit, the FUEL ON/OFF selector switch, on the solenoid valve which cuts off the main fuel supply of the fuel control unit. The FEATHER STOP switch-light activates the engine shutdown system with feathering the propeller when the main control system is used or with setting the propeller blades to a higher pitch when use is made of the standby control system. In this case the HP fuel shut-off valve lever installed on the fuel control unit is moved to the OFF ("stop") position.

Monitoring of the engine operation is performed by the airborne test facilities. The transmitters and warning devices, installed on the engine, send the signals on the engine operating parameters and state of the assemblies, systems and entire engine to the flight compartment indicators, annunciators and the EEC system.

Visual inspection of the engine gas flow duct is performed with the use of an optical instrument through the special inspection ports. The optical instrument' permits inspecting:

- the compressor turbine rotor blades;
- the power turbine rotor blades;
- the inner surface of the combustion chamber flame tube;
- the nozzle guide vane assembly of the compressor turbine first stage;
- the rotor blades of all compressor stages.

Located on the engine rear reduction gear is electrical generator drive *6* (see Fig. 4.4). Electrical generator is not shown in Fig. 4.4, because it is not delivered by engine plant and is installed during the engine mounting on the aircraft.

4.4. Engine mounting attachments

The engine is attached to the aircraft structure in two attachment mounts to three points: two front trunnions *3* (see Fig. 4.1) for attachment to the shafting and rear turn-on *7* for attachment to the rear reduction gear. Engine standby attachment trunnions *8* hold the engine when mechanical coupling in one of the points of the front mount is disturbed. The engine is attached to the shock absorbers, incorporated, together with the shock mount, into the aircraft attachment fittings set.

REVIEW QUESTIONS

1. What helicopter turboshaft engine was used as a base to design gas generator of ТВ3-117ВМА- СБМ1 airplane turboprop engine?
2. What new structural parts were added to the turboshaft helicopter engine ТВ3-117ВМА to design turboprop engine?
3. What two main structural parts the turboprop engine ТВ3-117ВМА-СБМ1consists of?
4. What function is performed by the turbine drive?
5. What function is performed by the propeller drive transmission system?
6. Name three main structural parts of the propeller drive transmission system.
7. Name main structural modules of the ТВ3-117ВМА-СБМ1 turboprop engine.
8. How the gas generator components are attached to the shafting?
9. How the engine is attached to aircraft structure?
10. How many rotors the ТВ3-117ВМА-СБМ1 turboprop engine includes?
11. How many stages the compressor unit of ТВ3-117ВМА-СБМ1 turboprop engine consists of?
12. Call the main engine systems and point out their functions.

Chapter 5. ENGINE ТВ3-117ВМА-СБМ MAIN DATA AND PERFORMANCE

5.1. Engine Main Data General

— Engine designation ТВ3-117ВМА-СБМ;
— Engine type .... Turboprop;
— Direction of rotor rotation .... Counterclockwise, as viewed from the exhaust unit side;
— Direction of propeller rotation .... Counterclockwise (looking forward);
— Compressor .... Axial, 12 stages;
— Combustion chamber .... Annular with 12 burners and 2 igniter plugs;
— Compressor turbine .... Axial, reaction, 2 stages;
— Power turbine .... Axial, reaction, 2 stages;
— Propeller drive transmission .... Remotely located with rear reduction gear (gear ratio — 1.09), shafting and front reduction gear (gear ratio — 12.11).

Engine overall dimensions:
— length .... 2953 mm;
— height .... 1209 mm;
— width .... 988 mm;
— propeller diameter .... 3.720 mm.

Mass of engine at delivery .... 790.5 kg, maximum.

Operating cycle data

(for takeoff rating at M_{fl} = 0; H = 0; t_{amb} = +15 °C):
Pressure ratio .... approx. 10;
Gas temperature at turbine entrance .... 1235 K;
Mass airflow .... approx. 10 kg/s.

Engine data for power ratings

Maximum emergency power at M_{fl} = 0; P_{amb} = 760 mm Hg ; t_{amb} = +37 °C:
Propeller shaft power .... 2058 kW;
Propeller shaft torque .... 95,3 %;
Propeller rotational speed .... 1203 RPM (100,0 %).

Takeoff power: at M_{fl} = 0; P_{amb} = 760 mm Hg; t_{amb} = +30 °C:
Propeller shaft power .... 1837 kW (2500 h.p.);
Propeller shaft torque .... 85,0 %;
Propeller rotational speed .... 1203 RPM (100,0 %).

Takeoff power at M_{fl} = 0; P_{amb} = 760 mm Hg; t_{amb} = +15 °C:
Specific fuel consumption .... 0,206 kg / e.h.p. · h (0,28 kg / kW · h).

Takeoff power at M_{fl} = 0,5; H = 6000 m; ISA:
Propeller shaft power .... 1456 kW (1980 h.p.);
Propeller shaft torque .... 67,4 %;
Propeller rotational speed .... 1203 RPM (100,0 %).

Maximum continuous power at M_{fl} = 0; P_{amb} = 760 mm Hg; t_{amb} = +35 °C:
Propeller shaft power .... 1543,5 kW (2000 h.p.);
Propeller shaft torque .... 78,1 %.

Maximum continuous power at M_{fl} = 0.5; H = 6000 m; ISA:
Propeller shaft power .... 1359,7 kw (1850 h.p.);
Propeller shaft torque .... 68,8 %;
Propeller rotational speed .... 1100 RPM (91,4 %).

Maximum cruising power at M_{fl} = 0.5; H = 6000 m; ISA:
Propeller shaft power .... 1286 kW (1750 h.p.);
Propeller shaft torque .... 65,1 %;
Specific fuel consumption .... 0,193 kg/e.h.p. · h;
Propeller rotational speed .... 1100 RPM (91,4 %).

Flight idle power (Fl) at M_{fl} = 0.2; H = 0 m; ISA:
Propeller shaft power .... (117,6 ± 18) kW (160 ± 25) h.p.;
Propeller shaft torque .... 6 %;
Propeller rotational speed.. 1100 RPM (91,4 %), maximum.

Ground idle power (G_I): at M_{fl} = 0; H = 0 m; ISA:
Propeller rotational speed.. 1100 RPM (91.4 %), maximum.

Ground idle power — "low-speed taxiing" (Gl-L-ST) at M_{fl} = 0; H = 0 m; ISA:
Propeller rotational speed.. 840 to 910 RPM (70,2 to 76,0 %).

Thrust reversal:
Propeller shaft maximum power — not in excess of maximum continuous power;
Propeller rotational speed.. 1100 RPM (91,4 %).

Propeller rotational speed for thrust reversal power ratings:
Variable within the range of.. 840 to 1203 RPM (70.2 to 100.0 %).

NOTE: The engine main operation parameters are given without taking into account:

- Pressure losses in the inlet section and exhaust unit;
- Loading of the aircraft accessory drives;
- Losses due to engine air bleed for powering the aircraft systems.

Time of engine continuous operation at main power ratings:
Maximum emergency power... 2.5 min, maximum;
In-flight emergency power (one engagement
during overhaul life).. 60 min, maximum.

Takeoff power:
At takeoff .... 5 min, maximum;
In climb to an altitude above 4000 m
in icing conditions .... 10,5 min, maximum;
Ground idle power and "Low-speed taxiing" ground
idle power, total.. 30 min, maximum;
Maximum reverse thrust .... 3 min, maximum;
Other power ratings.. Unlimited within overhaul life.

NOTES. 1. Maximum emergency power is used only in the event of engine failure at aircraft takeoff and go-around. The engine is set at this power automatically in response to the failure signal of one engine.

2. In-flight emergency power is used only for completion of the flight in the event of engine failure.

3. In abnormal flight conditions specified in the aircraft Flight Manual the in-flight emergency power may be used after 2.5-min operation of the engine at maximum emergency power, with a total time at both power ratings not exceeding 60 min.

4. After using the in-flight emergency power the engine is to be removed from the aircraft and restored by the engine manufacturer or repair organization.

5. Takeoff power may be used continuously up to 10.5 min in climb to an altitude above 4000 m in icing conditions, within the engine total time specified below.

6. Maximum continuous power may be used in cruising flight in adverse flight conditions within the engine total time specified below.

Engine total time at power ratings in percent of overhaul life:
In-flight emergency power.. 60 mm, maximum (one engagement);
Takeoff power .... 5.5 %, maximum;
Maximum continuous power.. 31 %, maximum;
Other power ratings.. Unlimited.

NOTES: 1. Upon using the maximum emergency power make an entry in the engine Log Book on the actual operating time for every engagement of this power setting individually.

Calculate engine total time considering the following relationship:

— One engagement of the maximum emergency power is equivalent to 50 flight cycles;

— 1 s of operation at the maximum emergency power is equivalent to 30 s of operation at take-off power and to 20 mm of the engine total time at all power ratings;

2. Upon using the in-flight emergency power make an entry in the engine Log Book and take into account the actual time of operation at this power rating in calculation of the engine total time.

Time of acceleration: on the ground and in flight from flight idle power to 95 % takeoff power by advancing the throttle control lever within 1 s – 5 s, maximum.

Grade of fuel (main and starting):

— Main .... ТС-1 ГОСТ 10227-86;
— Duplicate .... РТ ГОСТ 10227-86;
— Standby .... Т2 ГОСТ 10227-86;
— Foreign grades .... Type JetA-1 ace. to Specification.
DEF STAN 91-91 (DERD 2494, ASTM D 1655) and mixtures thereof in any proportion.

NOTES: 1. To prevent formation of ice crystals in fuel, the following crystal-formation preventive fluids are added to fuel: "И" ГОСТ 8313-88, "H-M" TV 6-10-1458-79, "S-748" DERD 2451 (Great Britain), MIL-I-27686F(USA), AIR 3652B (France) under the conditions and in quantities specified in the aircraft Maintenance Manual.

2. It is allowed to operate the engine on the above fuels with the use of antistatic additives Sigbol (Сигбол) ТУ 38.101741-78, ASA-3 Shell, Stadis 450 Du Pont Co.

Oil grade:

Main — ИПМ-10 ТУ38.1011299-90 with changes 1, 2; Turbonyc oil 210Ato AIR 3514/A and mixtures thereof in any proportion;

Duplicate — Mobil Turbo 319A-2 to MIL-PRF 7808L Gr 3;
Foreign — AeroShell Turbine Oil 390 DEF TA'N 91-94, Shell;
Castrol AERO 325 DEF STAN 91-94, Castrol;
Exxon Turbo Oil 2389 MIL-PRF-7808L Gr. 3, Exxon.

NOTE: Never mix oil of main and duplicate grades, of duplicate and foreign grades and foreign oil of various grades.

Oil consumption .... 0,4 l/h, maximum.
Engine oil system .... self-contained, circulation type.

Engine starting system .... Air, automatic.

Engine attachment .... Engine is attached in the aircraft in three points:

- Two trunnions on the shafting front casing;
- Trunnion on the rear reduction gear casing.

Quantity of bled air and air bleed conditions:

— for air condition system .... 0,25 kg/s;
— for APU heating .... 0,06 kg/s;
— for heating of underfloor space .... 0,0275 kg/s;
— for aircraft anti icing system .... 0,25 kg/s;
— for anti icing system of air-oil cooller air intake .... 0,025 kg/s;
— for aircraft anti icing system .... 0,25 kg/s.

5.2. Engine Performance

General. Variation of the engine parameters versus the flight altitude (H), ambient air temperature (t_{amb}) and Mach number (M_{fl}) characterizing the indicated airspeed (IAS), is stipulated by the fuel supply laws.

The fuel supply laws have been selected proceeding from the conditions of obtaining the engine parameters, which ensure the required aircraft performance and the reliable operation of the engine. The fuel supply laws are ensured by the engine electronic automatic control system. At the steady engine power ratings, the electronic system forms the following control laws:

- maintaining the propeller power (N_{prop}) proportional the throttle control lever (TCL) angular position;
- limitation of the compressor drive turbine exhaust gas temperature EGT ($t_{comp.\ turb}$) versus TCL angular position. At the engine power ratings from flight idle to maximum cruising power, the limit temperature ($t_{comp.\ turb}$) is of the maximum cruising power rating;
- limitation of physical (measured) value of the gas generator rotor rotation speed (RPM)(n_{GG}) versus the throttle control lever angular position. At the engine power ratings from flight idle to maximum cruising power, the limited value of gas generator rotor rotation speed n_{GG} is equal to of maximum cruising power rating;
- limitation of the maximum corrected gas generator rotor speed $n_{GG\,corr} = n_{GG}\sqrt{\frac{288}{T_{in}^*}}$, where T_{in}^* is the absolute air temperature (K) at the engine inlet; 288K is the absolute air temperature for standard atmosphere under ground conditions;

- limitation of the maximum propeller shaft torque (M_{trq});
- maintaining the propeller rotational speed (n_{prop}) appropriate for each engine power rating.

At the engine power ratings from the takeoff power to the maximum emergency (in-flight emergency) power, inclusive, the propeller rotational speed n_{prop} is 1203 RPM. At the power ratings from the ground idle to the maximum continuous, inclusive, the propeller rotational speed $n_{prop}$is 1100 RPM. At the ground idle at "Low-speed taxiing" the propeller rotational speed n_{prop} is 840 to 910 RPM.

Description of engine performances. There are three different dependencies which are used to characterize an engine performance . These three are named as: throttle performance, altitude-speed performance and climatic performance. Graphs, corresponding these three performances are shown in Fig. 5.1.

Throttle Performance. In general case **t**he throttle performance means variation of main engine parameters (engine power, gas generator rotor speed, compressor outlet air pressure gas temperature at the combustion chamber outlet section or gas temperature at the compressor drive turbine unit exhaust,) versus variation of the fuel flow rate (G_f) through the combustion chamber. When engine operates at the ground and when airplane is unmovable (flight speed V_{fl} = 0) the fuel flow rate (G_f) is determined fully by position of power control lever in cockpit. The throttle performance in ground standard atmospheric conditions (H = 0, M_{fi} = 0,P_{amb} = 101,3 kPa (760 mm Hg), t_{amb}= +15 °C) is presented in Fig. 5.1, *a*. As it is visible in graphs propeller horse power N_{prop} increasing from idle lever up to take-off value is achieved due to very considerable increasing G_f(more than four times).The compressor outlet air pressure (P_c) is increased too approximately two times. This process is accompanied by very considerable increasing the compressor drive turbine exhaust gas temperature ($t_{comp.turb}$) from 425 °C up to 680 °C and gas generator rotor speed (n_{GG}) rising from approximately 80 % at idle operation mode up to 100 % at take-off operation mode.

In its turn, the propeller rotational speed varies versus the engine), within the engine power ratings variation from the ground idle operating mode to the maximum emergency power, as dependence of the fuel flow rate (G_f), power rating, as specified subpart 5.1.

Increase in fuel supply to the combustion chamber results in increase in the power fed to the operating medium, which causes increase in the gas temperature, gas generator rotor speed and compressor outlet air pressure. In this case, the engine power increases. Observed within 0,15 to 0,17 maximum continuous power is a slight variation in the throttle performance as within this range of the power ratings the compressor seventh-stage air bleed valves close (open). The above indicated variation is shown in Fig. 5.1, *a*, for clarity as break in the appropriate lines.

Climatic performance(Engine parameters variation versus ambient air temperature) is dependence of the basic engine parameters at the maximum emergency and takeoff power ratings in the static ground conditions (H = 0, P_{amb} = 101,3 kPa(760 mm Hg), M_{fl} = 0) versus the ambient air temperature is presented in Fig. 5.1*, b.* At t_{amb} > 30 °C for takeoff power and t_{amb} > 37 °C for maximum emergency power the engine operates with the constant turbine exhaust gas temperature ($t^*_{comp.turb}$) corresponding to ($t^*_{comp.turblim}$) for each of the above power ratings. In this case, with increase in the engine inlet air temperature the propeller power decreases. At t_{amb} < 30 °C (t_{amb}< 37 °C) maintained in the engine is the propeller power appropriate for each engine power rating, according to the law, described above. In this case, the fuel flow rate, gas generator rotor speed and compressor turbine exhaust gas temperature decrease, while the compressor outlet air pressure increases with decrease in the engine inlet air temperature.

Variation of the propeller power and the specific equivalent fuel flow rate, with the engine operating at the maximum cruising power at altitude (H = 6000 m; M_{fi} = 0,3, 0,4, 0,5; P_{amb}= P_{ambISA}) versus the ambient air temperature, is similar to shown in Fig. 5.1, *b*.

At flight speeds, corresponding to M_{fi} = 0,3 and M_{fi} = 0,4, at an ambient air temperature below minus 30 °C, the engine automatic control system limits the corrected gas generator rotor speed *(*n_{GGcorr}) at a flight speed corresponding to M_{fl} = 0,5, at t_{amb} < –25 °C, maintained is the propeller power (N_{prop}) corresponding to the preset engine power rating. At higher ambient air temperatures and the above indicated flight speeds the engine operates with the compressor turbine exhaust gas temperature ($t^*_{compJuib}$) limited.

Altitude-airspeed performance: The altitude-airspeed performance data of the engine operating at maximum continuous power in standard atmospheric conditions are presented in Fig. 5.1, *c* as the basic engine parameters versus the flight altitude and speed. Up to an altitude of approximately 3500 m, at all presented airspeeds, the engine power is maintained by controlling the propeller power (N_{prop}); at altitudes above 4800 m the engine power is limited by the compressor turbine exhaust gas temperature $t^*_{compturb}$).

At the engine power below or above the maximum continuous, variation of the engine parameters, with change in the altitude-airspeed conditions, is identical to that of the engine operating at the maximum continuous power, except for the engine power ratings and flight conditions at which the maximum physical (measured) or corrected gas generator rotor speed is reached. Reaching the maximum physical or corrected gas generator rotor speed results in the fuel supply procedure which prevents exceeding of this rotor speed.

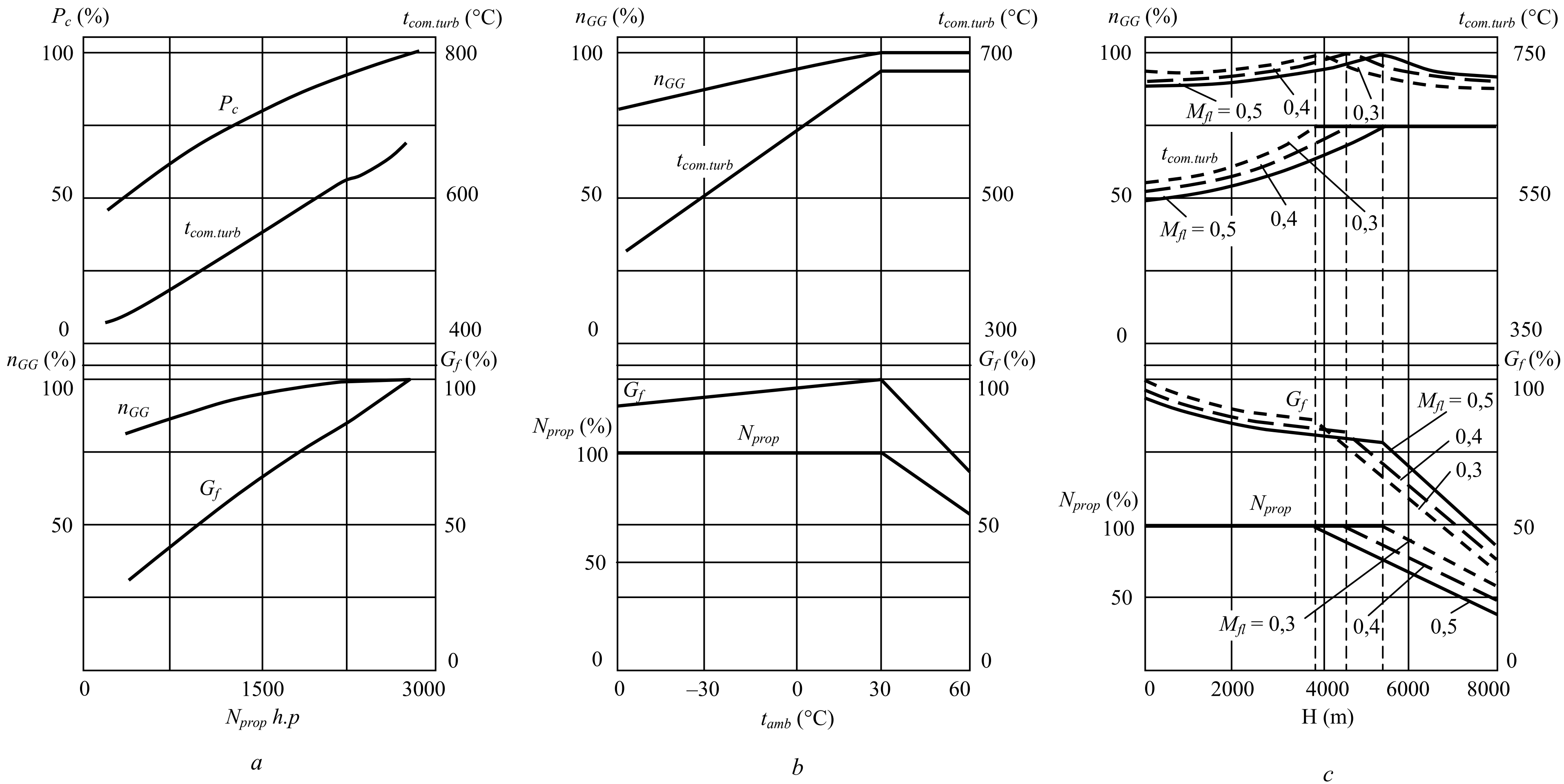


Fig. 5.1. Engine performance:
a — throttle performance at $H = 0$, $M_{flL} = 0$, $t_{amb\ ISA}$, $P_{anb}^{*}{}_{ISA}$; *b* — variation of basic engine parameters at maximum emergency and takeoff power ratings versus variation of ambient air temperature (t_{amb}) at $H = 0$, $M_{fl} = 0$, $t_{amb\ ISA}$, $P_{amb\ ISA}$; *c* — engine altitude-airspeed performance at maximum continuous power versus variation of flight altitude (*H*) and airspeed (M_{fl}) under standard atmospheric conditions

REVIEW QUESTIONS

1. Name main general data of ТВ3-117ВМА-СБМ turboprop engine.
2. What are the main overall dimensions and mass of ТВ3-117ВМА-СБМ turboprop engine?
3. Call the main operating cycle parameters of a gas turbine engine and note the values of these parameters for ТВ3-117ВМА-СБМ turboprop engine when it operates at M_{fl} = 0; H = 0; t_{amb}= +15 °C condition.
4. What main power ratings are usually given for turboprop engines?
5. Note the value of output power of the ТВ3-117ВМА-СБМ turboprop engine at maximum emergency and takeoff operating modes.
6. What grades of fuel and oil are recommended to be used for the ТВ3-117ВМА-СБМ turboprop engine?
7. What working body is used by the ТВ3-117ВМА-СБМ turboprop engine starting system?
8. What are the purposes of the engine bled air used in aircraft?
9. Name a list of engine performances which are mentioned above.
10. How does the turboprop engine output power change when the flight altitude is increased?
11. Why does the turboprop engine output power change when the flight altitude is increased?

Chapter 6. ENGINE ТВ3-117ВМА-СБМ TRANSMISSION

6.1. Introduction

The propeller drive transmission is intended to drive a propeller at a required rotational speed from a power turbine. The transmission provides for a power turbine RPM -to-propeller RPM gear ratio n_{pt} / n_{prop} = 13.2 : 1. The exterior view of engine transmission is shown in Fig. 6.1 and its longitudinal section view is represented in Fig. 6.2. The propeller drive transmission is remotely located over the turbine engine drive and is used for engine mounting to the aircraft structure. The transmission structure consists of rear reduction gear *9* (Fig. 6.1), shafting *7* and front reduction gear *3*.

6.2. Rear reduction gear description and operation

The rear reduction gear is designed to transmit the torque from the power turbine of the gas-turbine drive to the shafting located above the drive, and to reduce the power turbine rotational speed as well. The reduction gear is made as an ordinary cylindrical gearing having a gear ratio 1.09 : 1.

The rear reduction gear consists of casing *23* (Fig. 6.2), which has a lateral joint in which three spur pinion gears *29*, *26*, *25* are running in ball bearings. Driving gear *29* is connected with splined coupling *30*, the latter connecting the rear reduction gear with the power turbine drive shaft. Intermediate gear *26* engages driving gear *29* and driven gear *25* and, via drive gear *27*, and via accessory drive gearbox *28* gets in mesh with generator accessory drive gearbox *28* gear. Driven gear *25*, via the splined coupling is connected with the drive shaft of the shafting.

The lower section of casing *23* of the rear reduction gear incorporates a turbine drive attached with the use of ball support *19* (Fig. 6.1), the upper section of the casing is provided with a flange for attaching shafting casing *19* (Fig. 6.2).

6.3. Shafting description

The shafting is intended for transmitting the torque from the rear reduction gear to the front one. The shafting consists of casing *19* (Fig. 6.2), which has two lateral joints to house intermediate bearings *17* and *21* of splined couplings *16* and *20* with drive shafts *18* and *22* installed between them.

6.4. Front reduction gear description and operation

The front reduction gear is intended for reducing the shafting rotational speed down to a value required for propeller rotation. The front reduction gear is a two-stage, co-axial differential planetary reduction gear with the gear ratio equal to 12.11 : 1.

The first stage of the reduction gear has driving gear *12* that is connected by means of drive shaft *14* with shafting coupling *16* and is engaged simultaneously with three planetary gears *11* installed in planetary gear casing *31*, which, in their turn are, engaged with rim gear *10*. Planetary gear casing *31* is connected by means of splines with propeller shaft *3*, while rim gear *10* is connected, also by means of splines, with driving gear *32* of the second stage. Driving gear *32* of the second stage is engaged simultaneously with four planetary gears *8* installed in second stage planetary gear casing *33*, which, in their turn, are engaged with second stage rim gear *7* installed on propeller shaft *3* with the help of splined hub *35*.

Planetary gear casing *33* is connected by splines with torquemeter rim *9*. Propeller shaft *3* is installed in the front reduction gear casing *4* on two bearings (front — roller bearing type, and rear — ball bearing. The shaft flange has openings for propeller attachment.

The propeller attachment sealing is provided by rubber sealing ring *1* placed on the centering belt of propeller shaft *3*. The oil cavity of the reduction gear is sealed by annular seal placed in cap *40* and by pressurized air flowing from the compressor five stage via connection on the reduction gear casing.

Installed in the upper section of casing *4* is driven gear 6 of the drive of torquemeter oil pump *5* and propeller speed governor *13*. The driven gear is engaged with driving gear *34* installed on hub *35*.

The oil for controlling the propeller is supplied from propeller speed governor *13* through the inner high-pitch, low-pitch, and pitch-lock passages in reduction gear casing *4* and through oil by-pass assembly *36* into oil by-pass bushing *2* mounted in propeller shaft *3*. Oil by-pass bushing *2* has three co-axial cylindrical cavities arranged in sequence; the cavities are connected with the respective passages of the propeller when mounting it on shaft *3*. Over the circumference of shaft *5* are positioned three pushing rods *39* connected, at the side of propeller, with the propeller pitch control mechanism. The pushing rods, located in the reduction gear, are connected with propeller blade angle sensor via bearing *38* and rack gear train *37* transforming the translational motion of the rods into the rotary motion. The angle sensor measures the angle proportional to the angle of propeller blade turning.

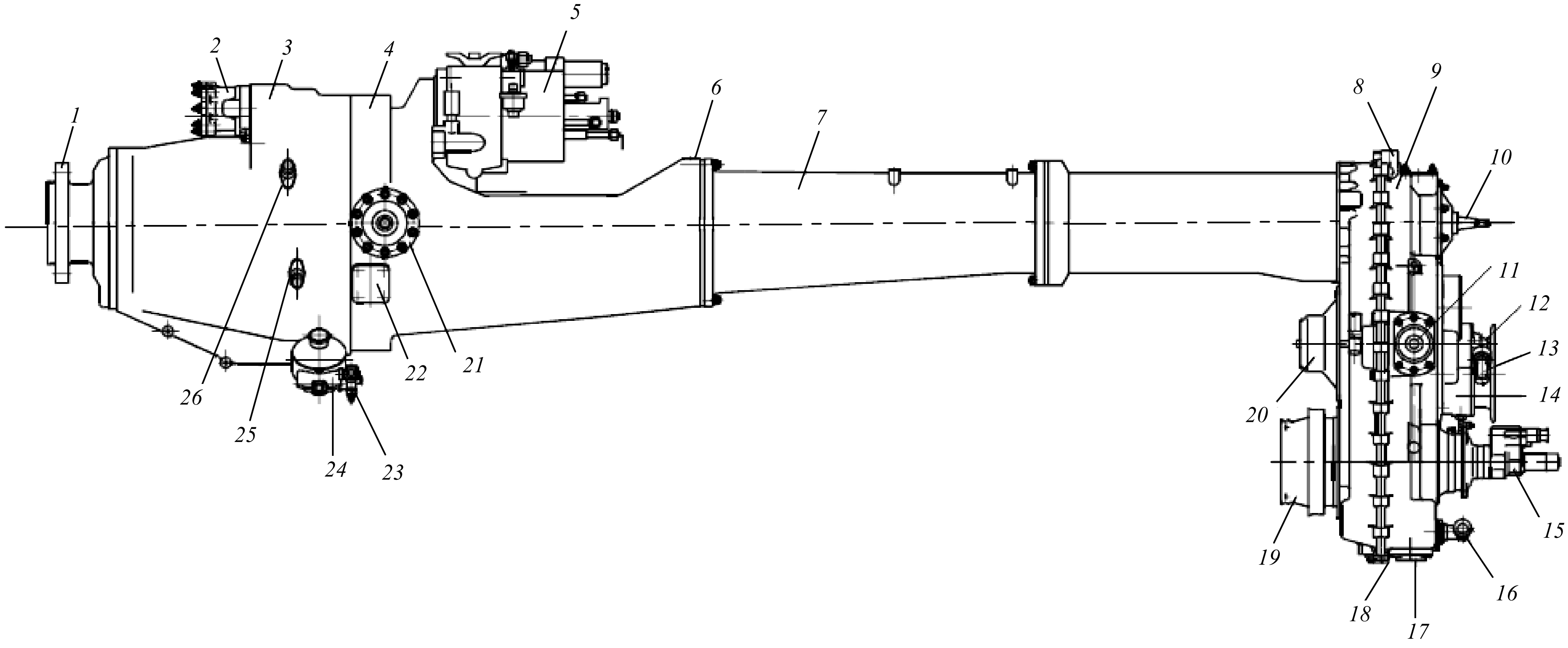


Fig. 6.1. Transmission — left side view:
1 — propeller shaft flange; *2* — torquemeter oil pump; *3* — front reduction gear; *4* — vibration pickup and mounting eye-bolts installation point; *5* — propeller speed governor; *6* — mounting eye-bolts installation point; *7* — shafting; *8* — mounting eye-bolts installation point; *9* — rear reduction gear; *10* — engine attachment rear trunnion; *11* — engine attachment spare trunnion; *12* — generator attachment adapter; *13* — centrimgal breather connection; *14* — accessory drive gearbox; *15* — propeller brake; *16* — rear reduction gear oil scavenge connection; *17* — rear reduction gear chip detector; *18* — reduction gear oil drain plug; *19* — turbine drive ball support; *20* — support cap; *21* — engine attachment front trunnion (both at left and right sides); *22* — turbine drive suspension brackets (both at left and right sides); *23* — front reduction gear oil drain valve; *24* — front reduction gear chip detector pan; *25* — propeller rotational speed sensors installation points; *26* — propeller rotary speed phase sensor installation point

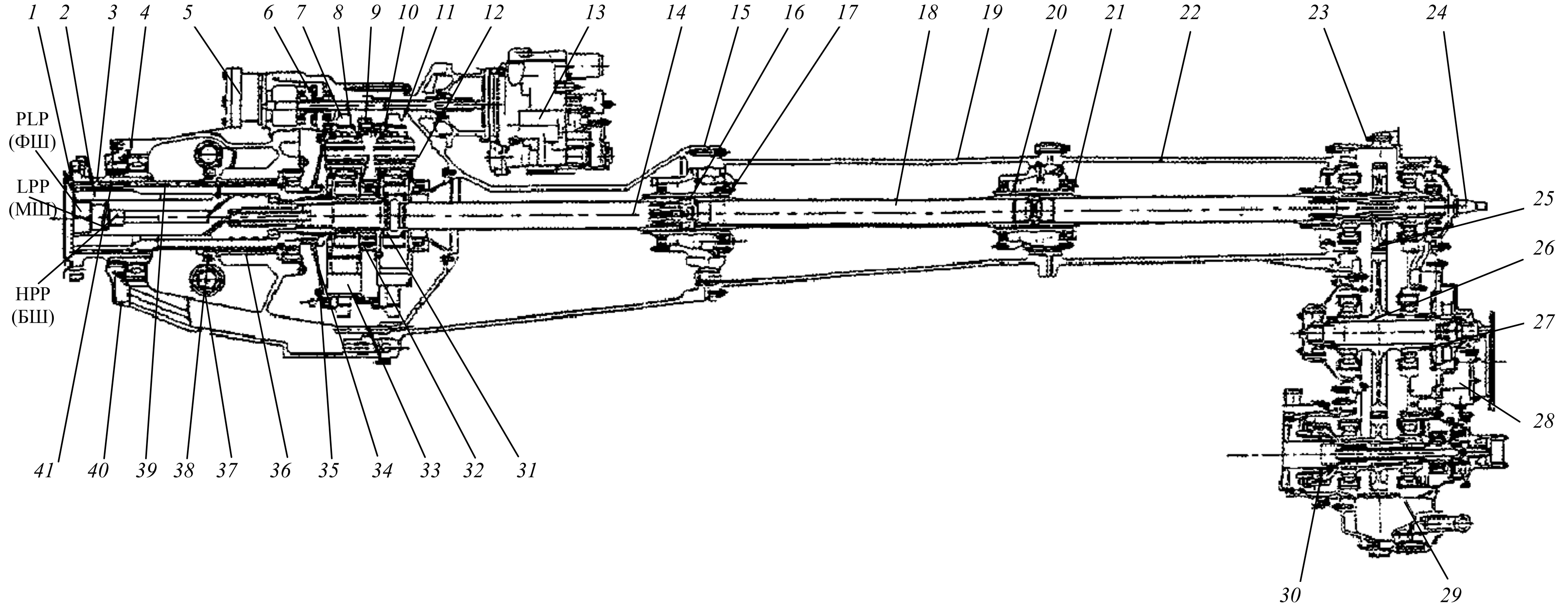


Fig. 6.2. Transmission-longitudinal section view:
1 — propeller sealing ring; *2* — oil by-pass bushing; *3* — propeller shaft; *4* — front reduction gear casing; *5* — torquemeter (TRQMT) oil pump drive; *6* — driven gear; *7* — second stage rim gear; *8* — second stage planetary gear; *9* — torquemeter rim; *10* — first stage rim gear; *11* — first stage planetary gear; *12* — first stage driving gear; *13* — propeller speed governor; *14* — drive shaft; *15* — mounting eye-bolts installation point; *16* — shafting coupling; *17* — shafting intermediate bearing; *18* — drive shaft; *19* — shafting casing; *20* — shafting coupling; *21* — shafting intermediate bearing; *22* — drive shaft; *23* — rear reduction gear casing; *24* — engine attachment rear trunnion; *25* — driven gear; *26* — intermediate gear; *27* — drive gear; *28* — accessory drive gearbox; *29* — rear reduction gear driving gear; *30* — splined coupling; *31* — first stage planetary gear casing; *32* — second stage driving gear; *33* — second stage planetary gear casing; *34* — driving gear; *35* — hub; *36* — oil by-pass assembly; *37* — rack gear train; *38* — bearing; *39* — pushing rod; *40* — seal cap; *41* — propeller anti-icier slip ring bracket; PLP (ФШ) — Propeller pitch lock passage; LPP (МШ) — Low-pitch passage; HPP (БШ) — High-pitch passage

6.5. Torquemeter description

The front reduction gear has a torquemeter. Torquemeter rim *7* (Fig. 6.3) is connected by its eyelets with six hydraulic actuators *6* which, by means of pistons *5* arranged in them, are connected with reduction gear casing *4*. The oil from torquemeter oil pump *1* is forced through the inner passages of reduction gear casing *4* via manifold *3*. The pressure in the torquemeter is measured by oil pressure transmitter *2* installed on front reduction gearbox casing.

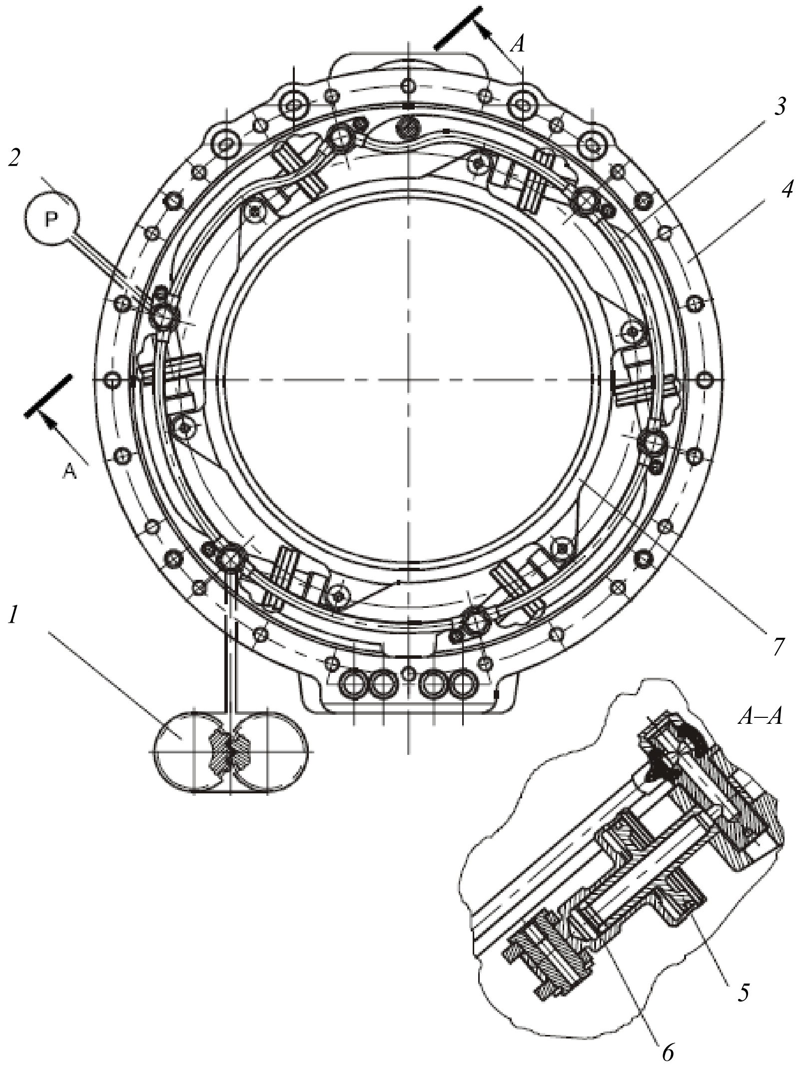


Fig. 6.3. Torquemeter:
1 — torquemeter oil pump; *2* — oil pressure transmitter; *3* — manifold; *4* — reduction gear casing; *5* — piston; *6* — hydraulic actuator; *7* — torquemeter rim

During operation of the reduction gear the torque arising on second stage planetary gear casing *33* (see Fig. 6.2) is transmitted by torquemeter rim *7* (Fig. 6.3) to casing *4* of the reduction gear via hydraulic actuators *7*, where oil pressure is built-up in proportion to the torque being transmitted. Oil pressure transmitter *2* forms an electrical signal which is used by the electronic engine control (EEC) monitor for computing the torque value. The torque value is indicated on the indicator in the flight compartment.

REVIEW QUESTIONS

1. What are the main purposes of the ТВ3-117ВМА-СБМ engine transmission?
2. What main components does the ТВ3-117ВМА-СБМ engine transmission comprise?
3. What is the ratio of power turbine rotor rotational speed to the propeller rotational speed of the ТВ3-117ВМА-СБМ engine transmission?
4. What purpose is the torquemeter used for?
5. Where is the torquemeter mounted?
6. Where is the torquemeter oil pressure transmitter installed?

Chapter 7. ENGINE AIR INLET SECTION

7.1. Concept

The air entrance or flight inlet duct is normally considered to be part of the airframe, not part of the engine. Nevertheless, it is usually identified as engine station number one. Understanding the function of the inlet and its importance to engine performance makes it a necessary part of any discussion on gas turbine design and construction.

Although the inlet duct is made by aircraft manufacturer, during flight operation becomes very important to the overall jet engine performance and will greatly influence jet engine thrust output.

The faster the airplane goes, the more critical the duct design becomes. Engine thrust can be high only if the inlet duct supplies the engine with the required airflow at the highest possible pressure. The nacelle/duct must also allow the engine to operate with minimum stall tendencies and permit wide variation in angle of attack and yaw of aircraft. For subsonic aircraft, the nacelle/duct should not produce strong shockwaves or flow separations and should be of minimum weight for both subsonic and supersonic designs.

Inlet ducts add to the parasitic drag, or aerodynamic resistance drag. Parasitic drag can be broken down into skin friction due the viscosity of the air, form drag due to the shape of the duct, and interference drag that comes from the junctions of aircraft's components.

Inlet duct must operate from static ground run up to high aircraft flight speed with a high duct efficiency at all altitudes, attitudes, and flight speeds.

Inlet duct should be as straight and smooth as possible and should be designed in such a way that the boundary layer air (layer of still, dead air lying next to the surface) will be held to minimum. The length, shape and placement of the duct is determined to a great extend by the location of the engine in the aircraft.

The inlet ducting of a gas turbine engine provide a relatively distortion-free, high-energy supply of air, in the required quantity, to the face of compressor.

A uniform and steady airflow is necessary to avoid compressor stall and excessive internal engine temperatures at the turbine. The high energy enables the engine to produce an optimum amount of thrust or power. Normally the air-inlet duct is considered an airframe part, and not a part of the engine. However, the duct is so important to engine performance that it must be in any discussion of the complete engine, as inefficiencies of the inlet duct result in successively magnified losses through other components of the engine.

The inlet duct used on multi-engine subsonic aircraft such as we find in the commercial jet aircraft fleet is a fixed geometry duct whose diameter progressively increases from the front to back. A diverging duct is sometimes called an inlet diffuser because of the effect it has on the pressure of the air entering the engine. As air enters the inlet at ambient pressure it begins to diffuse, or spread out, and by the time it arrives at the inlet to the compressor its pressure is slightly higher then the ambient pressure. The inlet for a turbofan is similar in design to that for a turbojet except that it discharges only a portion of its air into the engine, the remainder passes only through the fan.

The inlet duct has two engine functions and one aircraft function. First, it must be able to recover as much of total pressure of the free air stream as possible and deliver this pressure to the front of engine with a minimum loss of pressure or differential. This is known as "total pressure recovery factor" or, sometimes, as "ram recovery factor". Secondly, the duct must uniformly deliver air to the compressor inlet with as little turbulence and pressure variation as possible. As far as the aircraft is concerned, the duct must hold to a minimum the drag effect, which it creates.

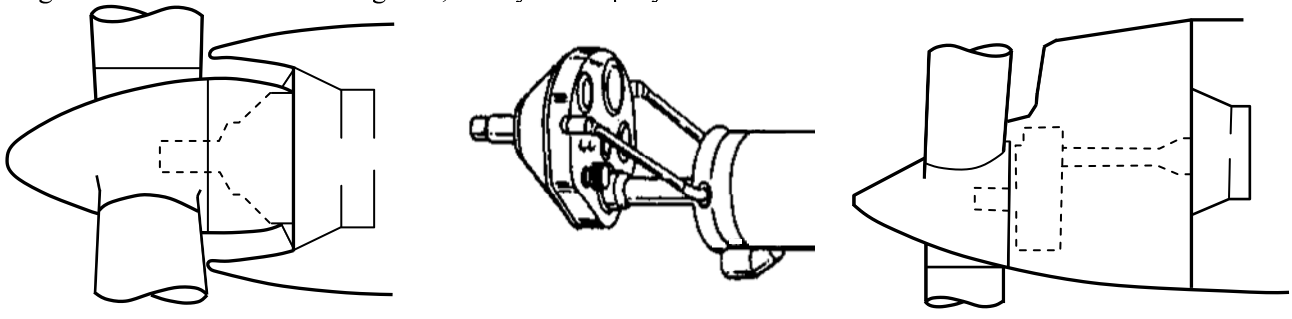
Pressure drop or differential is caused by the friction of the air along sides of the duct and by the bends in the duct system. The choice of configuration of the entrance to the duct is dictated by the location of the engine within the aircraft and the airspeed, altitude, and attitude at which the aircraft is designed to operate.

In general case an engine air inlet unit consists of tree main parts: the outer cowling, the inner intake fairing and the engine air inlet frame. Often the compressor front bearing case or reduction gear case serves as the engine air inlet frame.

The air inlet on a turboprop engine is more of a problem than that on a turbojet because the propeller drive shaft and reduction gear, the hub, and the spinner must be considered in addition to the usual other inlet design factors.

There are some different types of the design of turboprop engine inlet ducts (Fig. 7.1).

The ducted spinner arrangement (Fig. 7.1, *a*) is generally considered the best inlet design of the large turboprop engine as far as airflow and aerodynamic characteristics are concerned. However, the ducted spinner is heavier, and is more difficult to maintain and anti-ice than the conventional streamline spinner which is frequently used. A conical spinner, which is a modified version of the streamline spinner, is sometimes employed. In either event, the arrangement of the spinner and inlet duct is similar to that shown in Fig. 7.1, *b*. When the nose section of the turboprop engine is offset from the main axis of the engine, an arrangement similar to that in Fig. 7.1, *c* may be employed.



a *b* *c*

Fig. 7.1. Types of engine inlet sections for turboprop engines

7.2. Air inlet section of theTB3-117BMA-CBM engine description

The ТВ3-117ВМА-СБМ engine air inlet section with spacer (Fig. 7.2) is a main part of the compressor and is intended for feeding the compressor with the air having the parameters required for a stable compressor operation at all power ratings. Spacer 7 is attached to engine air inlet section *2* with the aid of studs *13* screwed into flange *10* of the engine air inlet section and self-locking nuts *12*.

The engine air inlet section consists of fairing *5*, shells *6*, *7*, *8*, *9*, *23*, pocket *22* and flange *10* welded up together and forming cavity *F* for supplying and cavity *E* for tapping hot oil in order to protect the engine air inlet section against icing. Welded up to shell 7 are adapter *21* (section *C–C)* for installation of oil supply swivel connection *20* and adapter *4* for installation of oil drain valve *5*. The oil drain valve serves to drain oil from the engine air inlet section, if it is necessary to remove the engine air inlet section.

Welded up to shell *23* (section *B–B*) for installation of oil outlet swivel connection *24*. Spacer *1* consists of shell *15* with, welded up to it, manifold *14*, flanges *11* and *16*, connection *17* of cleansing liquid drainage. Welded up to manifold *14* is connection *18* of cleansing liquid delivery. Welded up to flange *11* of the spacer is flange *10* of the engine air inlet section, while flange *16* serves to attach the engine air inlet section and spacer to the first bearing casing.

To flush the compressor airflow duct, spacer 1 is provided with twelve openings *K* in shell 75 which serve for cleansing liquid outlet to the airflow duct. Cleansing liquid is supplied to openings *K* through connection *18* and cavity *J* of manifold *14.* Manifold *14* is welded up to shell *15* in such a way that openings *K* enter cavity *J.*

Connection *17* serves to drain the cleansing liquid residue after flushing the compressor airflow duct, to drain the liquid as water and atmospheric precipitation. When the aircraft is parked, the liquid is drained overboard via connection *17* and opening in the engine cowl panel.

7.3. Air inlet section operation

The kinetic energy of the ram airflow in the engine air inlet section with spacer is transformed into potential pressure energy, with the airflow at the outlet of the spacer having all parameters required for a compressor steady operation. To prevent its icing, the engine air inlet section is heated by hot oil supplied from the lubrication system. During the engine operation hot oil from the lubrication system is supplied through swivel connection *20* and fills up cavity *F*, of pocket *22*. From cavity *F* the oil is delivered to cavity *L* formed by fairing *5* and shell *6*. From cavity *L* the oil flows into cavity *M* through opening *H* in shell *6,* as well as into a slot formed by the wall of pocket *22* and rib of shell *6*. From cavity *M* the oil flows into cavity *E*, formed by shells *9* and *23*, and through opening *P* in shell *9*. Through connection *24* the oil is tapped into the engine lubrication system.

REVIEW QUESTIONS

1. What are main purposes of an engine inlet duct?
2. What types of engine inlet ducts design are used in turboprop engines?
3. Name main quantitative index that is used to characterize the efficiency of an inlet duct operating.
4. What main components the ТВ3-117ВМА-СБМ engine air inlet section consists of?
5. For what purpose hot oil from the engine the lubrication system is supplied to some inlet duct components?
6. What components of the ТВ3-117ВМА-СБМ engine inlet section are heated?
7. Describe the path along which hot oil flows inside of the ТВ3-117ВМА-СБМ engine inlet section components.

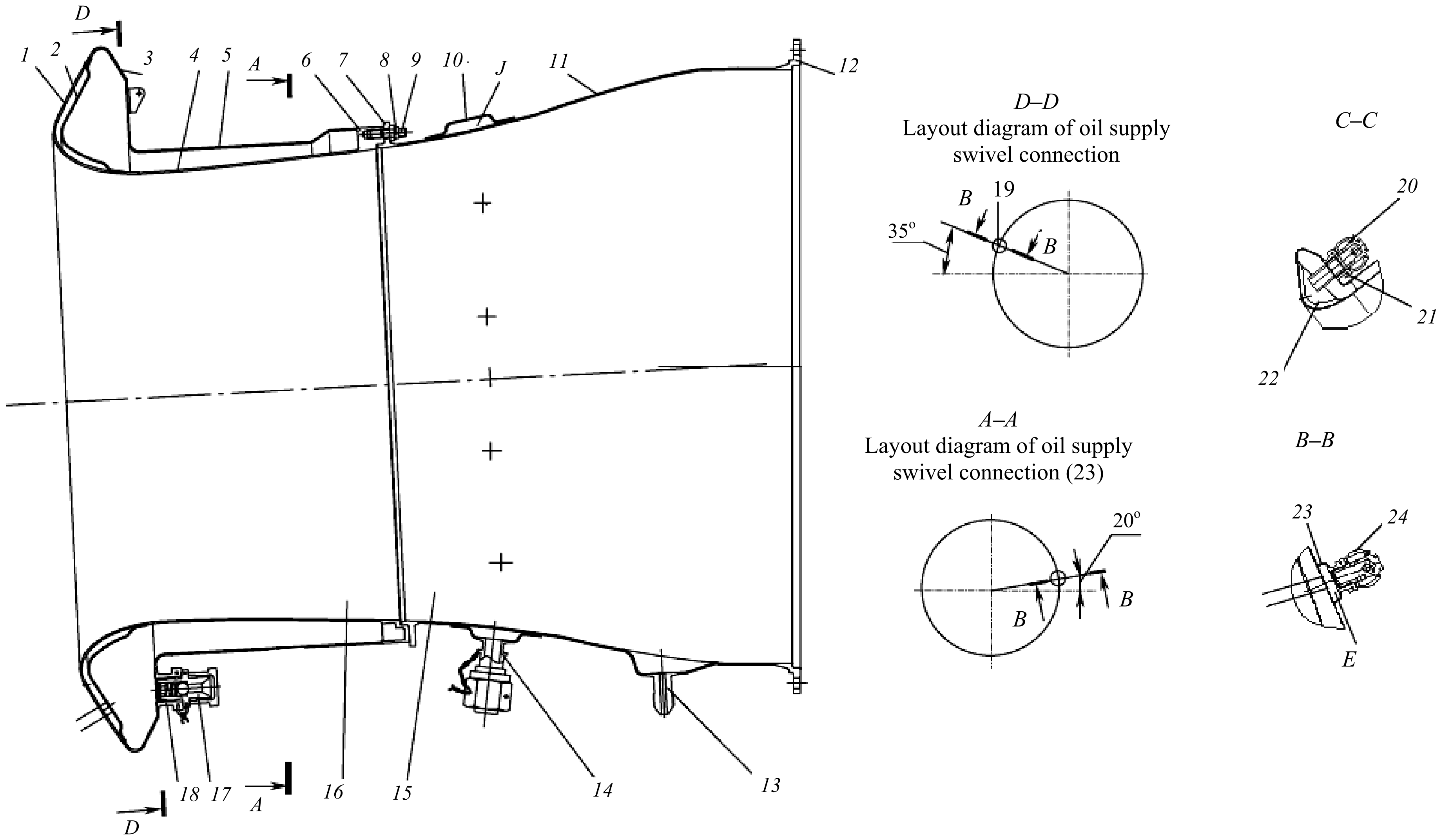


Fig. 7.2. Engine air inlet section with spacer:
1 — fairing; *2* — shell; *3* — shell; *4* — shell; *5* — shell; *6* — engine air inlet section flange; *7* — spacer flange; *8* — self-locking nu; *9* — stud; *10* — manifold; *11* — shell; *12* — flange; *13* — cleansing liquid drainage connection; *14* — cleansing liquid delivery connection; *15* — spacer; *16* — engine air inlet section; *17* — oil drain valve; *18* — adapter; *19* — oil supply swivel connection; *20* — adapter ; *21* — pocket; *22* — shell; *23* — oil outlet swivel connection; *24* — adapter; *E* — oil

Chapter 8. COMPRESSOR UNIT

8.1. Concept

In the gas turbine engine, compression of the air before expansion through the turbine is realized by compressor. As it was shown above, this compression of working body is need to increase thermal efficiency of the engine working cycle.

The compressor section of a turbine engine comprises the compressor rotor and the stator vanes and it supplies air in sufficient quantity to satisfy the needs of the combustor. The primary purpose of the compressor is to increase the pressure of the mass of air entering the engine inlet and then to discharge it into the diffuser and the combustors at the correct velocity, pressure, and temperature. The problems associated with these requirements are great because the compressor must move air at a velocity of around 120 to 150 meters per second and increase its pressure by perhaps 15 to 30 times in a space of only a few feet.

The secondary purpose of the compressor is to supply engine bleed air to cool the internal hot section, and supply heated air for inlet anti-icing. Air is also extracted for such aircraft uses as cabin pressurization, air conditioning, fuel system deicing heat, pneumatic engine starting, and various other functions that require compressed air.

Compression of the air before expansion through the turbine is effected, in the gas turbine engine, by one of two basic types of compressor, one giving centrifugal flow and the other axial flow. Both types are driven by the engine turbine and are usually coupled direct to the turbine shaft.

8.2. Centrifugal compressors

The centrifugal compressor, sometimes referred to as a radial outflow compressor is the oldest design and it still in use today. Many of the smaller flight engines as well as the majority of gas turbine auxiliary power units use this design. A centrifugal compressor performs its duties (Fig. 8.1) by receiving the air at its center with relatively low velocity (around 80–90 meters per second) and accelerating it outward by centrifugal force up to more higher velocity (around 430–450 meters per second). During the air flows along impeller passes the mechanical energy is transferred from impeller to the air flow. As a result of this, process is increasing in both velocity and pressure of air. After out flowing from impeller air enters a divergent duct called a diffuser, and as it spreads out, it slows down and its static pressure increases according to Bernoulli's principle.

Centrifugal compressors (Fig. 8.2) consist basically of an impeller rotor, a diffuser, and a manifold. The impeller is usually forged from aluminum alloy, and can be either single-or double-sided. The diffuser acts as a divergent duct in which the air spreads out, flows down and increases in static pressure. The compressor manifold distributes the air in a smooth flow to the compressor section. A single-stage, dual-side impeller allows a high mass airflow from a small diameter engine, and it has been used in a number of aircraft engines in the past for those reasons. This design does not, however, receive the full benefit from ram effect because incoming air must turn as it enters and leaves the compressor.

A single-sided impeller does benefit from ram intake and its less turbulent air entry makes it well suited for aircraft installation. Compression ratios attainable are about the same for both of the single-stage types of centrifugal impellers. More than one stage of compression can be used but the use of more than two stages of single-entry compressors is considered impractical. The energy lost in the airflow as it slows down to make the turns from one impeller to the next, the added weight, and the amount of power needed to drive the compressor all seem to offset the benefits of additional compression by using more than two stages.

The most generally used centrifugal compressor is the single-sided type with one stage. Recent developments in such centrifugal compressors have produced compression ratios as high as 5:1. Centrifugal compressors are shorter than axial flow compressors.

Tip speed of a centrifugal impeller may reach speeds as high as Mach 1.3, but the pressure within the compressor casing prevents airflow separation and provides a high transfer of energy into the airflow.

A centrifugal compressor may be used in combination with an axial flow compressor, and this is done in some of the smaller flight engines, but all of the larger engines today use axial flow compressors.

Advantages of centrifugal compressors are:

- high pressure rise per stage;
- good efficiency over a wide rotational speed range, idle to approximately Mach 1.3 tip speed;
- simplicity of manufacture and relatively low cost;
- low weight;
- low starting power.

Disadvantages of a centrifugal compressors are:

- large frontal area for a given airflow;
- more than two stages are not practical because of the energy losses between the stages.

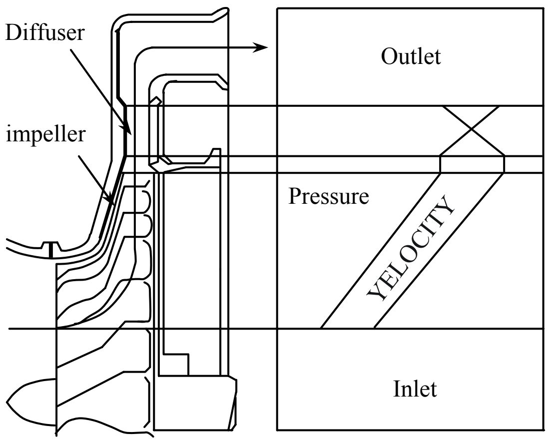


Fig. 8.1. Centrifugal compressor stage: pressure and velocity change

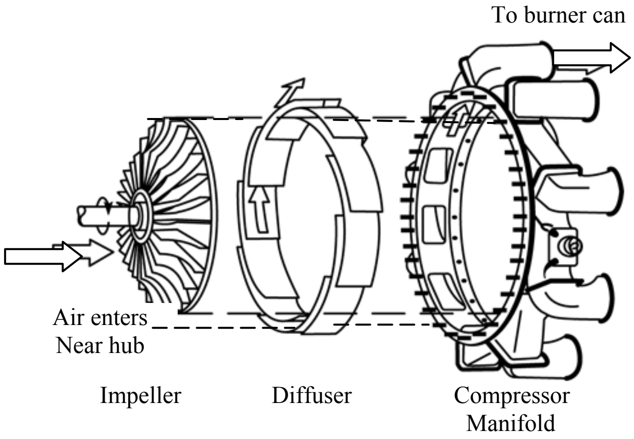


Fig. 8.2. Centrifugal compressor components

8.3. Axial flow compressors

There are three types of axial flow compressors: single-spool, dual-spool, and triple-spool. Single- and dual-spool compressors are used in turbojet and turboshaft engines while dual — and triple-spool compressors are commonly used in turbofan engines. A single-spool compressor has only one rotating mass. The compressor, shaft, and turbine all rotate together as a single unit. Fig. 8.3 shows how in a multi-spool engine, the turbine shafts attach to their respective compressors by fitting coaxially, one within the other. The front compressor is referred to as the low pressure, low speed, or N_1 compressor. Its turbine is referred to in the same manner. The rear compressor is called the high pressure, high speed, or N_2 compressor.

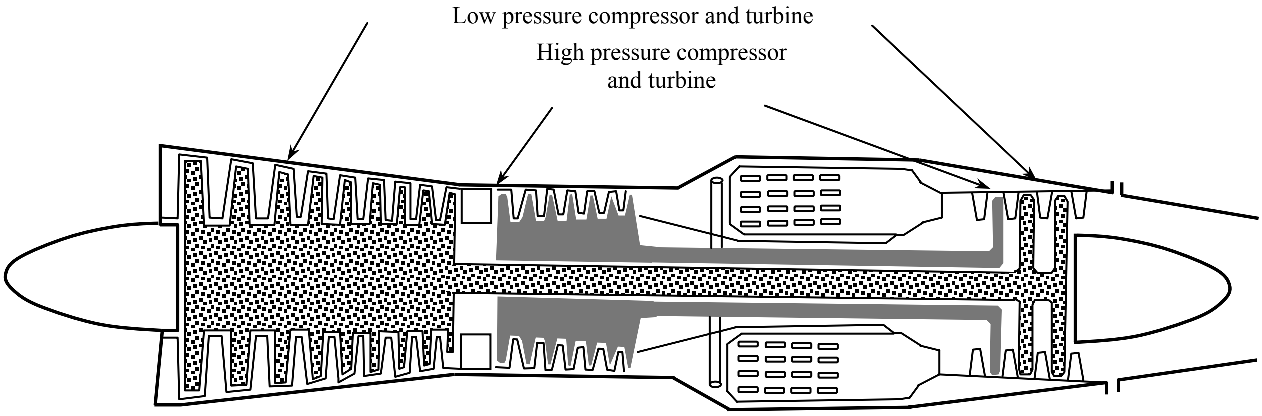


Fig. 8.3. A dual-spool axial flow compressor

Dual and triple-spool compressors were developed for the operational flexibility they afford the engine in the form of high compression ratios, quick acceleration, and better control of the stall characteristics.

There are several advantages of the axial flow compressors. There are:

- high peak efficiencies from ram, created by its straight-through design.
- high peak pressures attainable by addition of compression stages.
- small frontal area and resulting low drag.

The disadvantages of the axial flow compressor are: *r*-difficulty of manufacture and high cost:

- relatively high weight.
- high starting power requirements.
- low pressure rise per stage, approximately 1.27 : 1.

To take advantages of the good points of both the centrifugal and the axial flow compressors and eliminate some of their disadvantages, the combination axial-centrifugal compressor was designed (Fig. 8.4). This application is currently being used in many small turbine engines installed in business jet airplanes and helicopters. The combination compressor is specially well suited to engines using reverse-flow annular combustors. The engine diameter is wider to accommodate this type of combustor so there is no disadvantage in using the centrifugal compressor which, by the nature of its design, is much wider than a comparable axial flow compressor.

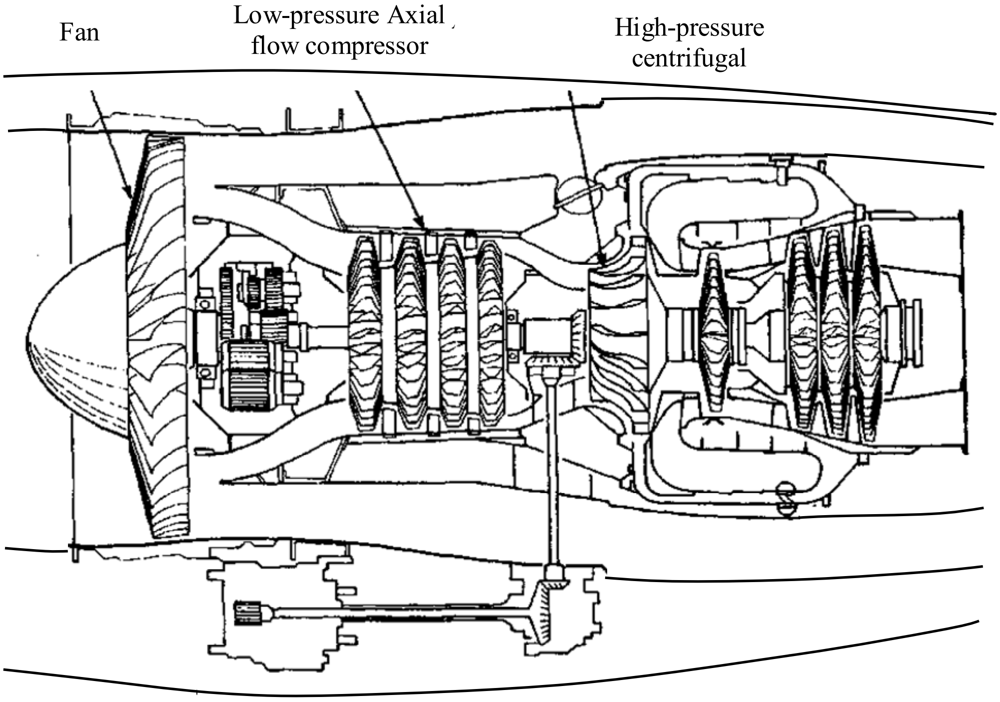


Fig. 8.4. The combination of axial flow and centrifugal compressors

8.4. Operating principle of axial flow compressor stage

An axial flow compressor has two main elements, the rotor and the stator. Such type of compressors usually consists of 8–13 stages. A compressor stage is a rotor blade set followed by a stator vane set. First stage (Fig. 8.5) beside it may include additional row of vanes which is named inlet guide vanes (IGV).

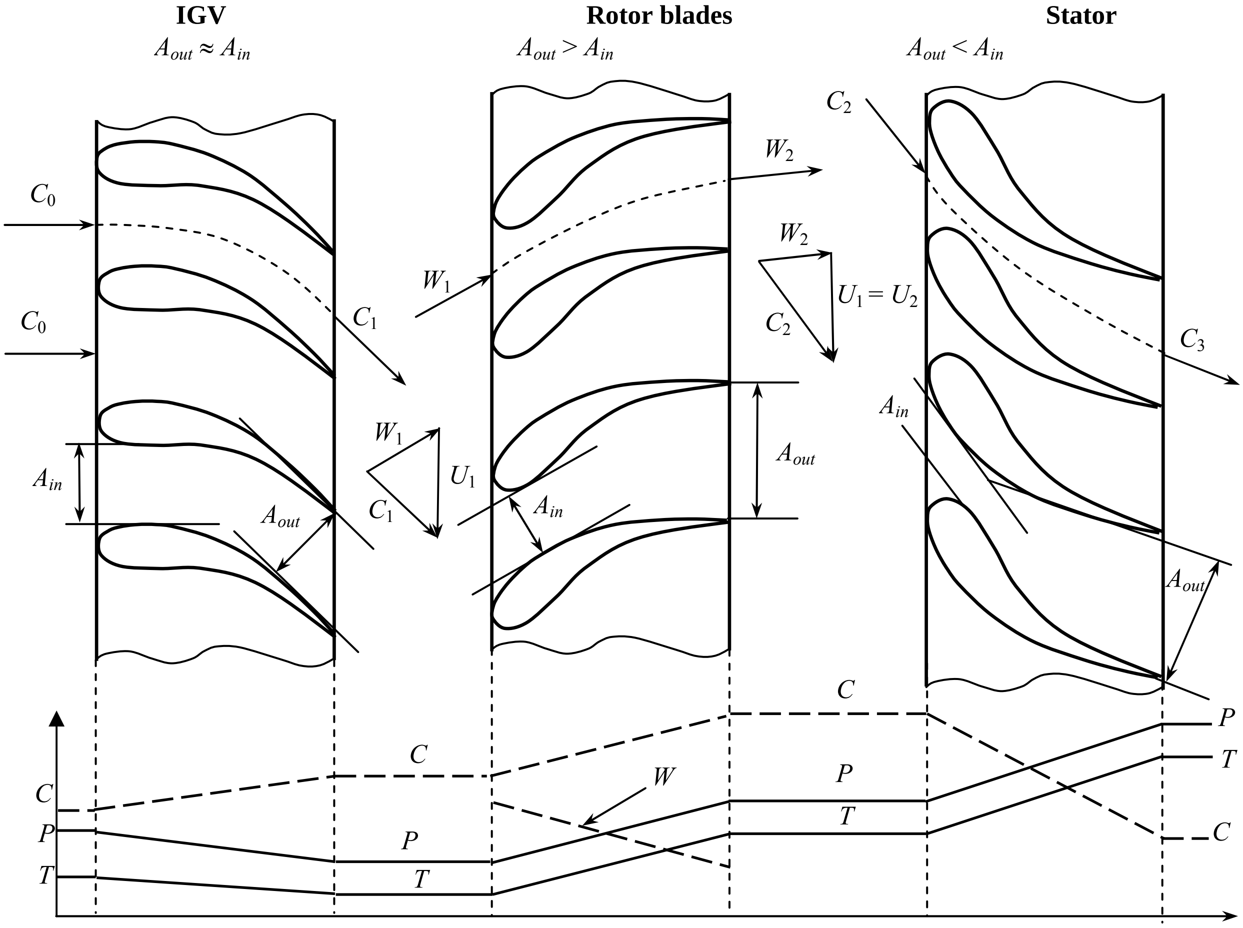


Fig. 8.5. Vector diagrams of interstage compressor airflow and pressure, velocity and temperature change through an axial flow compressor stage

Inlet guide vanes is a row of stator vanes which form curvilinear intervene channels for air passing. Cross-section of IGV, rotor blades and stator vanes (guide vanes) are shown in the upper part of the drawing (Fig. 8.5). In the same place, the velocity vector diagrams are shown. Compressor airfoils experience an infinite variety of angles of attack and air densities, and controlling the angle of attack is a design function of the inlet duct, the compressor and the fuel control sensors. The rotating compressor blades speed up the air in the inlet duct and the air passes through the inlet guide vanes, which changes its angle of flow, but does not change either its velocity ($C_1 = C_0$) or its pressure, as the outlet area A_{out} is about equal to inlet area A_{in}. Sometimes the velocity of airflow slightly increase when air flows in IGV intervene channels.

When the air passes through the rotor blades row, every its particle is involved in complex motion: rotation with speed U and translational movement with velocity *W*. Resultant velocity of these two motions C is equal vector sum of vectors W and U. As shown by the vector diagram in Fig. 8.5, the rotor blades increase the air velocity *C*. Due to this effect of rotation the power is input to airflow and total energy of airflow is increasing. When the air velocity increases ($C_2 > C_1$), the ram pressure of the air, passing through a rotor wheel, also increases. This increase in velocity and pressure is somewhat, but not entirely, nullified by the effects of diffusion ($A_{out} > A_{in}$). When the air is forced past the rear section of the rotor blades passage, the static pressure also increases, because the large area at the rear of the blades (due to their airfoil shape) acts as a diffuser. As the outlet area A_{out} of interblade channels is more than that of inlet area A_{in} the relative velocity W is decreasing, when the air passes through the rotor blades (according to Bernoulli's principle).

As the air leaves the trailing edge of the compressor rotor blades, it flows through a row of stator vanes which also form diverging ducts ($A_{out} > A_{in}$). As its intervenes channels are diffuser, the air velocity decreases ($C_3 < C_2$). In the stators the air velocity decreases, while the static pressure increases. As the air velocity decreases in the stators, the pressure due to velocity or ram pressure, which has just been gained in passing through the preceding rotor stage, decreases somewhat, although the total pressure remains the same.

The action of the compressor rotor blades and the stator vanes continues through all of the stages of compression, and when the air leaves the compressor it has approximately the same velocity it had when it started, but it has a much a higher static pressure (Fig. 8.6). As the pressure is built up by successive sets of rotors and stators, less volume is required. Thus the duct within the compressor is gradually decreased, as shown in Fig. 8.6.

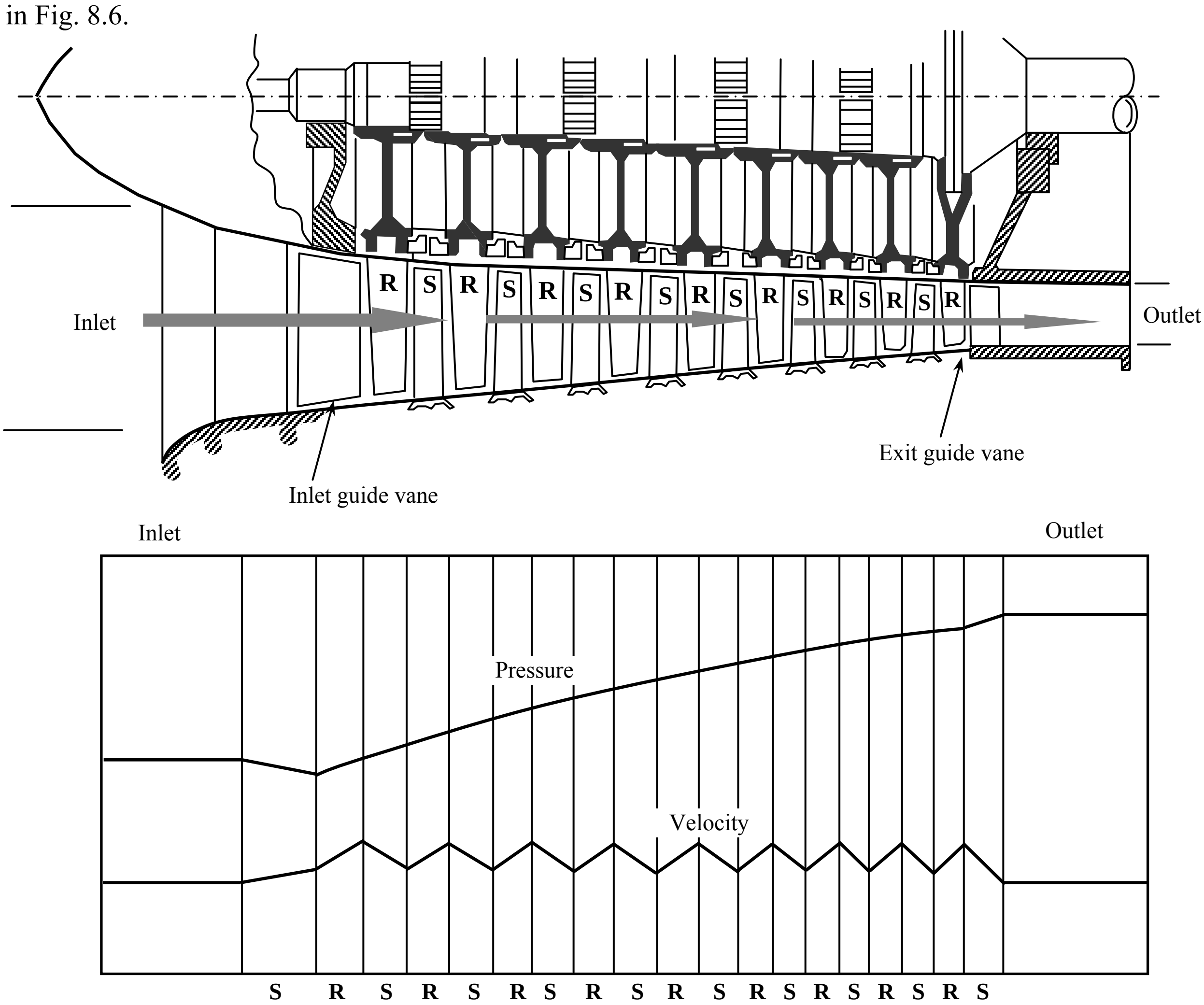


Fig. 8.6. Pressure and velocity changes as air passes through each stage of an axial flow compressor

Blades and vanes. The rotor blades force air rearward through each stage which consists of one set of rotor blades and the following set of stator vanes. The speed of the rotor determines the air velocity in each stage. As the velocity increases, kinetic energy is added to the air. The stator vanes are placed to the rear of the rotor blades to receive the high velocity air and act as diffusers, changing the kinetic energy of velocity into potential energy of pressure. The stators also serve a secondary function of directing the airflow into the next stage of compression. Compressor blades are constructed with a varying angle of incidence, or twist. This twist compensates for the blade velocity change caused by its radius. The further from the axis of rotation, the faster the blade section travels. The blades also decrease in size from the first stage to the last to accommodate the converging, or tapering, shape of the space in which they rotate.

The length, chord, thickness, and aspect ratio (ratio of the length to the width) of the compressor blade are designed to suit the performance factors required for a particular engine and aircraft combination.

Axial flow compressors normally have from 10 to 18 stages of compression with the fan considered to be the first stage of compression. Some long fan blades have a mid-span shroud fitted to each blade to form a circular ring which helps support the blades against the bending forces from the airstream. The shrouds, however, block some of the airflow, and the aerodynamic drag they produce reduces the efficiency of the fan. The section of the fan blade from the mid-span shrouds to the root, is the compressor blade section for the core engine. The roots of the compressor blades are often loosely fitted into the compressor disk for ease of blade assembly and for the vibration damping it provides. As the compressor rotates, centrifugal force keeps the blade in its correct position, and the airstream over the airfoil provides a shock absorbing or cushioning effect. These blades are attached to the disk with a dovetail and are secured with a pin and a lock-tab or lock-wire. Some blades are cut off square at the tip and these are referred to as flat machine tips. Other blades have a reduced thickness at the tips and these are called profile, or squealer, tips. All rotating machinery has a tendency to vibrate, and profiling a compressor blade increases the natural frequency of the blade. By raising the natural frequency of the blade beyond the frequency of rotation, the vibration tendency is reduced.

Profiling changes the aerodynamics at the tip to produce a smooth axial airflow even if the tip is rotating at speeds beyond the speed of sound and flow separation has started to occur. On some of the newer engines, the compressor rotor tips are designed to have a tight running clearance and rotate within a shroud tip strip of abrogable material. This strip will wear away rather than cause blade damage if contact takes place and the strip is replaced when the engine is overhauled. A high pitch noise can be heard on coast down if the compressor blade touches the shroud tip strip, and this is the reason profile tips are called squealer tips.

Variable vanes. Stator vanes may either be stationary or may have their angle variable. The inlet guide vanes which are the vanes immediately in front of the first stage rotor blades may also be stationary or variable. The function of the inlet guide vanes is to direct the airflow into the compressor at the most desirable angle. Exit guide vanes are placed at the compressor discharge to remove the rotational moment imparted to the air by the compressor. The air can flow rearward against the ever increasing pressure only because energy is transferred from the turbine as it drives the compressor. When a compressor blade or a stator vane has a positive angle of attack, the pressure on the bottom of its airfoil shape is higher than the pressure on the top. The high and low pressure areas formed on the airfoils allow air passing through one stage to be influenced by the next stage. This is called the cascade effect.

Angle of attack and compressor stall. The angle of attack of the compressor blade is determined by the inlet air velocity and the compressor RPM. These two forces combine to form a vector force which gives us the angle of attack of the airfoil. A compressor stall is a condition of airflow when the angle of attack becomes excessive. A compressor stall causes the airflow to slow down, stop, or even reverse its direction, depending upon the severity of the stall. Stalls can range from a slight air vibration, or fluttering sound, to a louder pulsating sound, or even to a violent backfire or explosion. Quite often the gages in the cockpit do not show a mild or transient stall condition, and these stalls are usually not harmful to an engine. They often correct themselves after one or two pulsations. But severe stalls, called hung stalls, can significantly impair engine performance, cause loss of power and can even damage the engine. The pilot can identify a stall condition by its audible noise, by fluctuations of the RPM, by an increase in the exhaust gas temperature, or by a combination of these symptoms. Compressor stalls may be caused by: turbulent or disrupted airflow to the engine inlet, which reduces the velocity vector; excessive fuel flow caused by abrupt engine acceleration — this increases the back pressure of the combustor and reduces the velocity vector; contaminated or damaged compressor blades or stator vanes; damaged turbine components which cause a loss of shaft horsepower delivered to the compressor — this decreases the compressor speed and reduces the velocity vector. The remedy for an acceleration stall is to reduce the power and allow the inlet air velocity and the engine RPM to get back into their proper relationship. Every compressor has a best operating condition for a given compression ratio, speed, and mass airflow. This is commonly called the design point.

8.5. The TB3-117BMA-CBM1 engine compressor description and operation

Description. The compressor of the TB3-117BMA-CBM1 engine (Fig. 8.7, 8.8) is an axial-flow, subsonic, single-rotor, twelve-stage unit. The engine compressor consists of the following main assemblies:

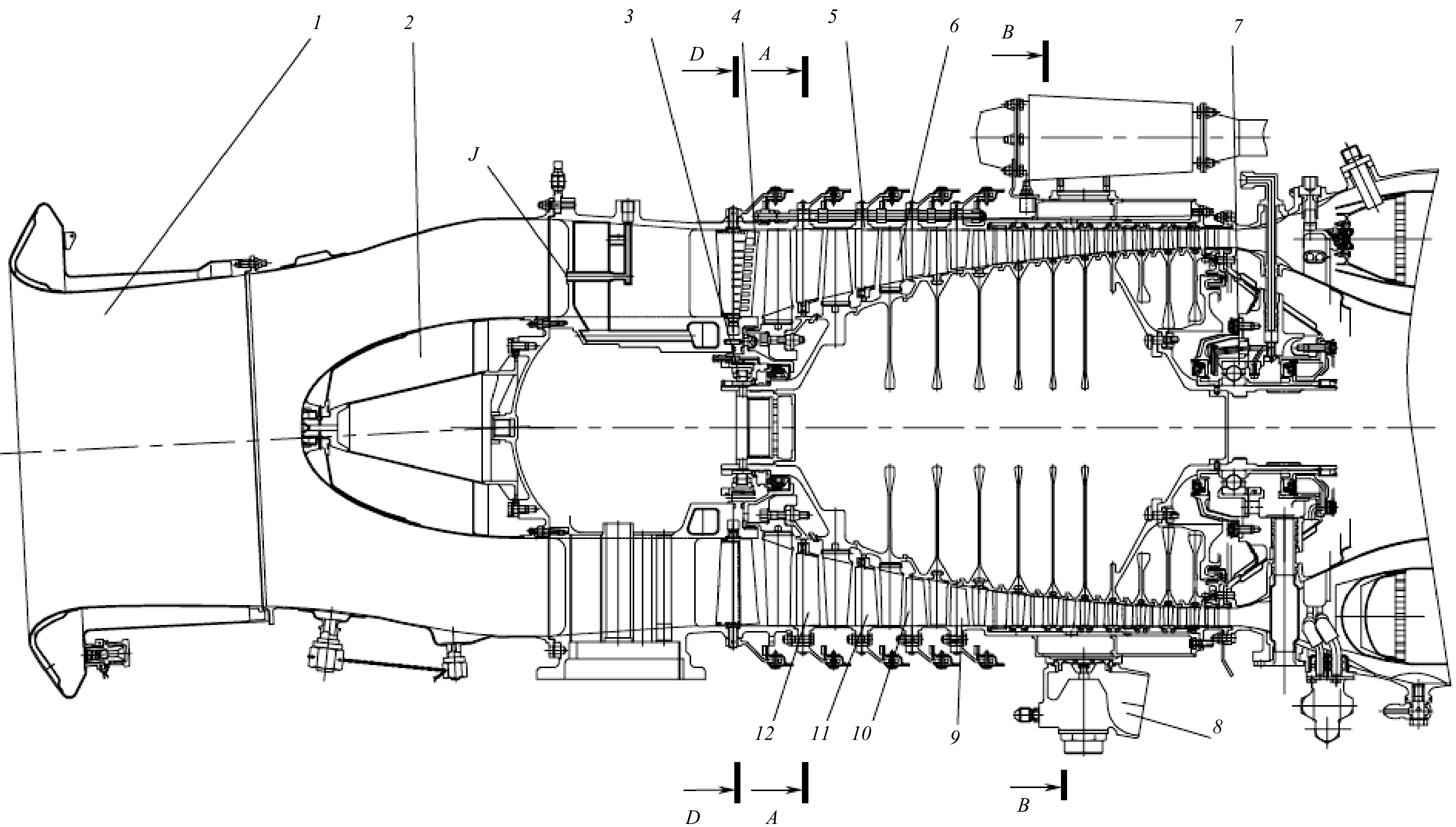


Fig. 8.7. The engine TB3-117TMA-CBM1 compressor unit:
1 — engine air inlet section with spacer; *2* — inlet fairing; *3* — compressor front (first) bearing; *4* — variable inlet guide vane assembly (IGV); *5* — stator with IGV and GV control mechanism; *6* — rotor; *7* — compressor rear (second) bearing; *8* — air bleed valve; *9* — 4th stage GV; *10* — 3rd stage GV; *11* — 2nd stage GV; *12* — 1st stage GV

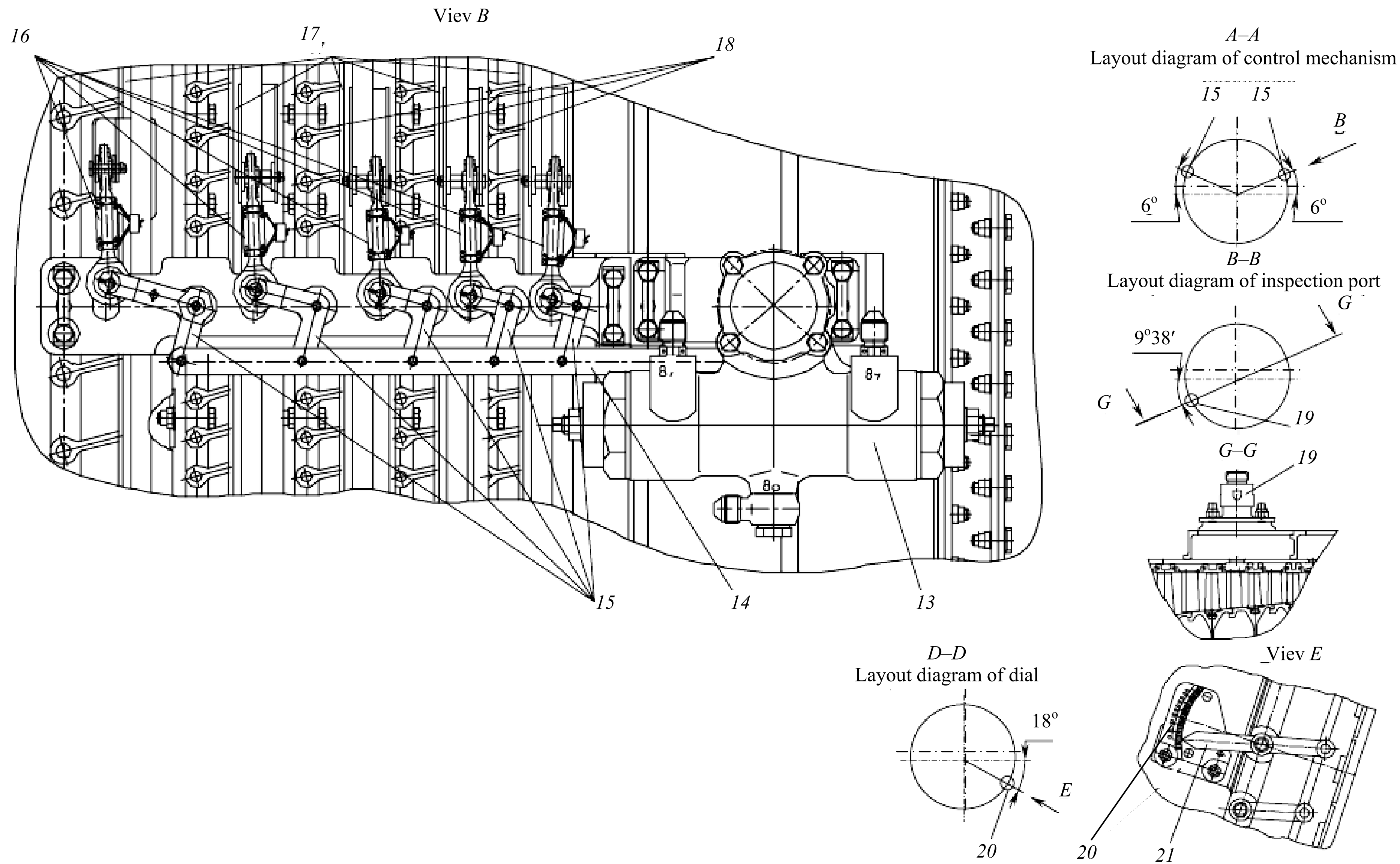


Fig. 8.8. Additional views to the engine TB3-117TMA-CBM1 compressor unit (cross sections and numeration are continued from Fig. 8.7):
13 — Hydraulic Actuator; *14* — Driving Element; *15* — Levers; *16* — Rods; *17* — Turning Rings; *18* — Vane Levers;
19 — Inspection Port; *20* — Dial; *21* — Angle Indicator; *J* — Opening to supply air to pressure transmitter at engine inlet

– engine air inlet section with spacer *1*;
– inlet fairing *2*;
– compressor front (first) bearing *3* with variable inlet guide vane assembly (IGV) *4*;
– stator *5* with control mechanism of IGV and variable guide vane assemblies (GV) of the first through fourth stages;
– rotor *6*;
– compressor rear (second) bearing *7*.

Operation. The axial force acting on the compressor rotor is taken up by a ball bearing mounted in the compressor rear bearing. A roller bearing of the compressor front bearing is mounted on an elastic damper. Rotor *6* of the compressor is driven to rotate from a two-stage compressor turbine. To provide for a stable operation of the compressor, the latter has IGV *4* and GV of the 1st through 4th stages *9*, *10*, *11*, *12*. The IGV and GV of the first through fourth stages, with the control mechanism are designed so as to allow automatic adjustment of the angle of guide vane setting versus the engine power ratings. In the process of the engine operation the value of guide vane angle position is controlled by a respective sensor (ДВСКТ-650-lШ duplicate resolver) which supplies an electric signal, proportional to guide vane setting angle, to the electronic engine control (EEC).

The duplicate resolver is mounted on the compressor front bearing casing and is cinematically connected with one of the IGV. Two hydraulic actuators *13* (Fig. 8.8, view *B*) located from the left and from the right, on the compressor stator, following a command from the automatic engine control system, move synchronously driving elements *14*, which, via levers *15* and rods *16*, via turning rings *17* and vane levers *18* turn the vanes of the IGV and GV of the first through fourth stages.

For determination of the inlet guide vane setting angle during the engine maintenance operations, the front bearing casing is provided with dial *20*, with angle indicator *21* travelling along the dial, which is fixed on one of the variable IGV. The vertical strut of compressor front bearing 3 has opening *J* o deliver air to the air pressure transmitter at the engine inlet.

To provide the engine gas dynamic stability during starting and at compressor rotor low speed, the compressor is equipped with two air bleed valves *8* (see Fig. 8.7) aft of the seventh stage of the compressor. The compressor has inspection port *19* (Fig. 8.8, Section *G–G*) protected by quick-removable plug, which permits, if necessary, to inspect the compressor seventh and eighth stages rotor blades with the help of an optical instrument. To prevent possible formation of ice on the surfaces of the compressor intake flow section during engine operation under low temperatures, a provision is made to heat the surfaces with hot air and oil.

Hot air bled from the compressor is used to heat the surfaces of the inlet fairing, front edges of both horizontal struts and a vertical upper strut of the compressor front bearing, compressor IGV.

Hot oil scavenged from the compressor front bearing is used to heat the vertical lower strut of the compressor front bearing. Hot oil is also used to heat the engine air inlet section.

8.6. The ТВ3-117ВМА-СВМ1 engine accessory drive system description and operation

The engine accessory drive system is intended for transmitting rotation from the engine rotor to the engine-rotor-driven accessories. In addition, accessory drive system serves for accelerating the gas generator rotor by means of the pneumatic (air) turbine starter when starting up the engine and for cranking of the gas generator rotor when performing the maintenance of the engine.

The accessory drive system consists of accessory drives from the gas generator rotor and of accessory drives from the power turbine.

The drives of accessories from the gas generator rotor (Fig. 8.9) consist of internal drive assembly *1* and accessory drive gearbox *4*.

The internal drive and the accessory drive gearbox are manufactured as individual assemblies mounted in and on housing *2* of the first bearing support and cinematically coupled together. The internal drive is attached to front flange *B* of the first bearing support housing. The accessory drive gearbox is attached to lower flange *A* of the first bearing support housing.

The drives of the accessories driven by the power turbine are arranged in the transmission (Chapter 6). Oil is supplied into the internal drive from the accessory drive gearbox via oil bypass pipe *3*. The accessory drive gearbox mounts the following units and assemblies: engine-mounted centrifugal fuel pump; oil pump block; fuel control unit; de-aerator with the oil scavenge pumps; pneumatic turbine starter; gas generator rotational speed sensors.

Mounted on the transmission are the following driven accessories and aggregates: propeller brake; electrical generator; propeller speed governor; torquemeter oil pump.

The kinematic diagram of the engine is shown in Fig. 8.10.

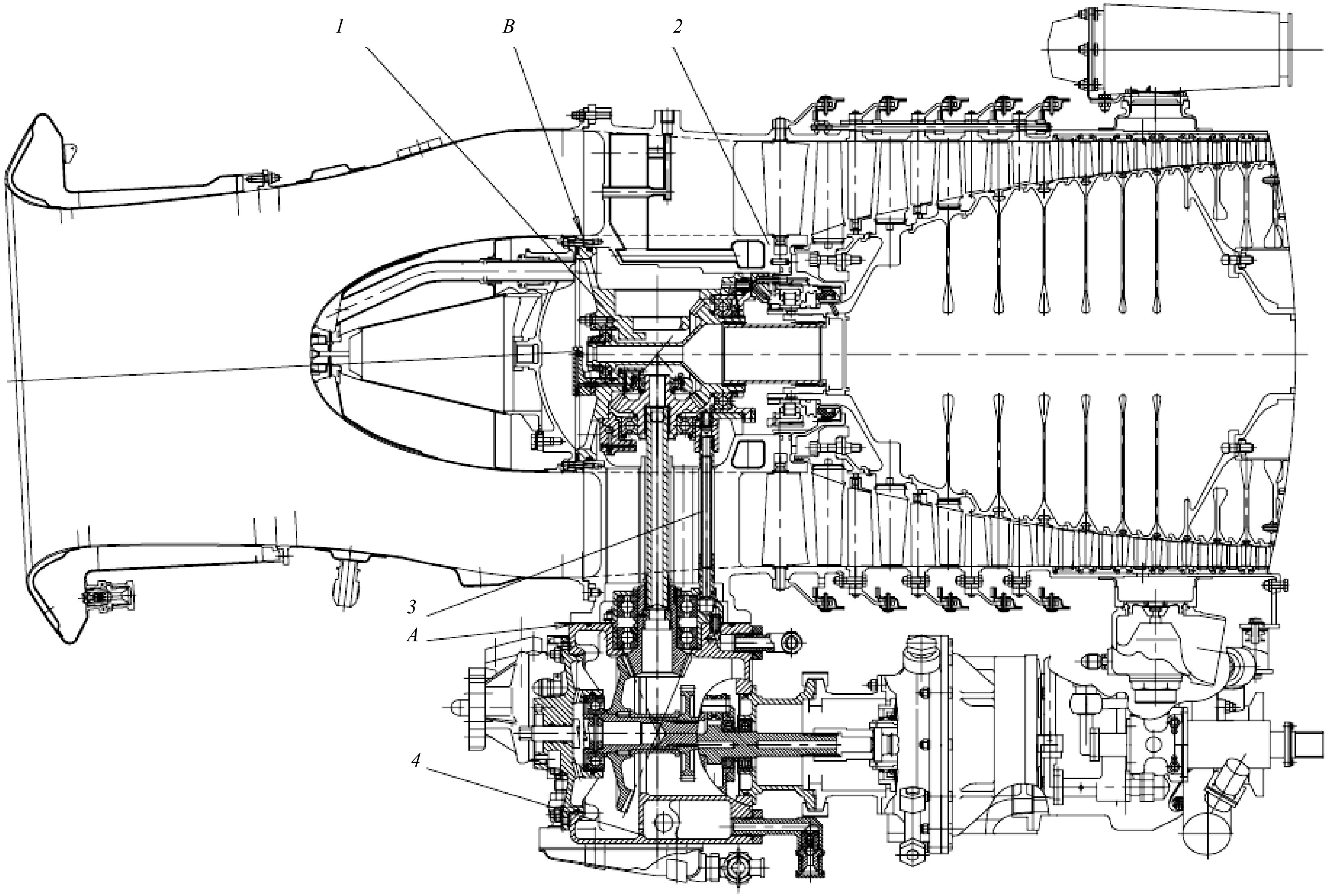


Fig. 8.9. The engine TB3-117TMA-CBM1 accessory drives from gas generator rotor:
1 — internal drive assembly; *2* — first bearing support housing; *3* — oil bypass pipe; *4* — accessory drive gearbox;
A — lower flange of first bearing support housing; *B* — front flange of first bearing support housing

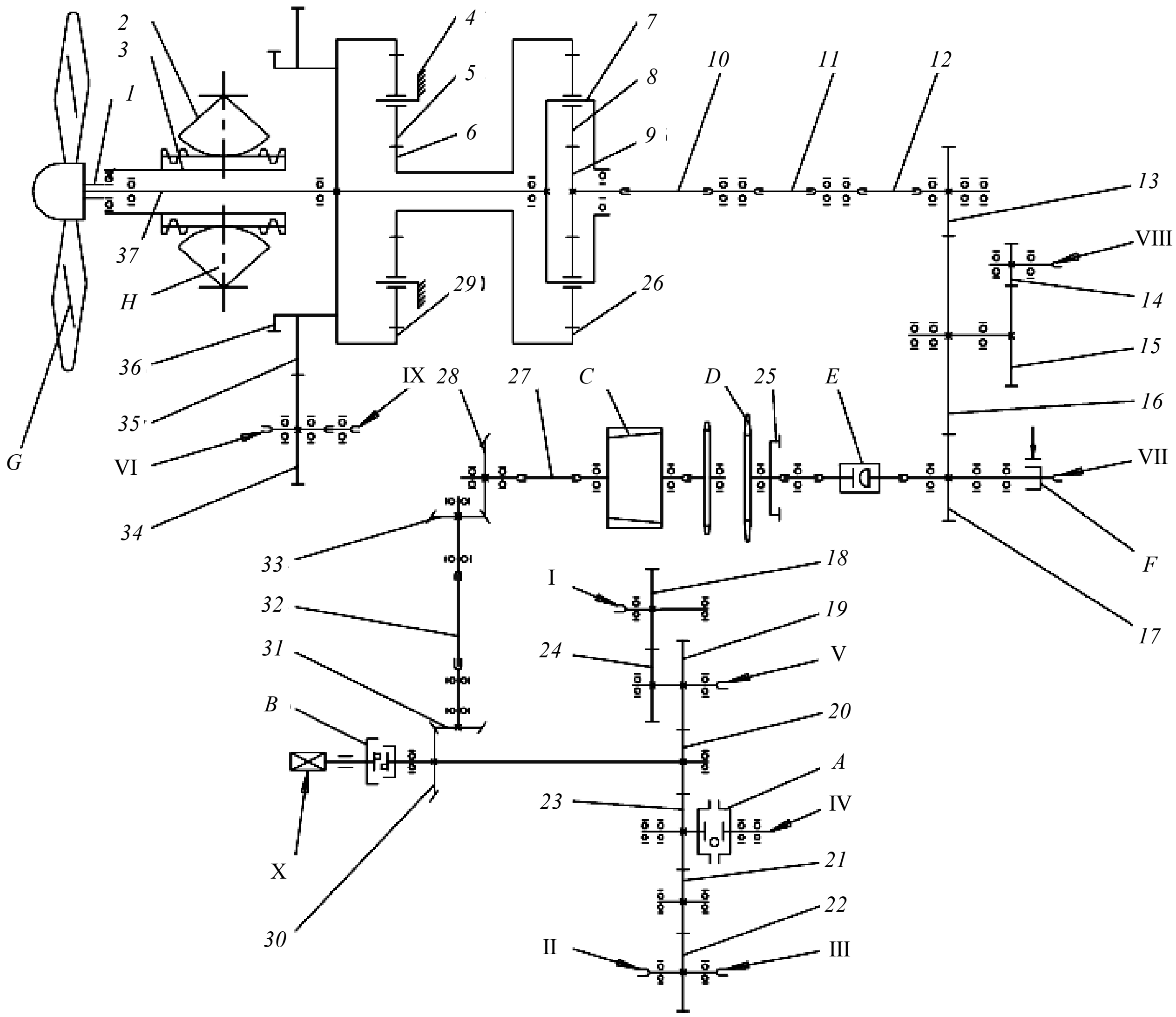


Fig. 8.10. The TB3-117TMA-CEM engine kinematic diagram:
1 — propeller blades turning mechanism; *2* — toothed segment; *3* — toothed rack;
4 — second stage planetary gears housing; *5*, *6* — gears; *7* — first stage parasitic gears housing;
8, *9* — gears; *10*, *11*, *12* — drive shafts; *13*, *14*, *15*, *16*, *17*, *18*, *19*, *20*, *21*, *22*, *23*, *24* — gears;
25 — inductor; *26* — gear; *27* — drive shaft; *28*, *29*, *30*, *31* — gears; *32* — drive shaft;
33, *34*, *35* — gears; *36* — inductor; *37* — propeller shaft

***Inductors*:** *18* — gas generator rotational speed inductor; *25* — power turbine rotational speed inductor; *35* — propeller rotational speed inductor; *36* — propeller rotation phase inductor.

Structural elements of accessory drive system: *A* — pneumatic turbine starter drive clutch; *B* — ratchet coupling; *C* — gas generator; *D* — power turbine; *E* — compensating coupling; *F* — propeller brake; *G* — propeller; *H* — propeller blades setting angle transmitter drive.

Driven accessories drives: I — Centrifugal fuel pump drive; II — Oil pump block; III — De-aerator drive with oil scavenge; IV — Pneumatic turbine starter drive; V — Fuel control unit; VI — Torquemeter oil pump drive; VII — Propeller brake drive; VIII — Electrical generator drive;
IX — Propeller speed governor drive; X — Gas generator rotor cranking drive.

Rotation from the gas generator rotor is transmitted to horizontal drive shaft *27* of the internal drive, from which, through bevel gears *28* and *33* and through vertical drive shaft *32* of the internal drive, the rotation is transmitted to bevel gears *31* and *30* of the accessory drive gearbox.

By means of its internal splines spur gear *20* is coupled with bevel gear *30* and is engaged with gear *19* of the fuel control unit drive and with gear *23* of the pneumatic turbine starter drive. Gear *24* engaged with gear *18* of the engine-mounted centrifugal fuel pump drive is arranged on one and the same shaft with gear *19*. Gear *18* serves as an inductor for the sensors measuring the rotational speed of the gas generator rotor. Gear *23* is engaged with intermediate gear *21* that, in its turn, is engaged with gear *22* of the drives of the oil pump block and de-aerator with the oil scavenge pumps. The cranking of the gas generator rotor is performed through ratchet coupling B connecting the driving ratchet of cranking with the driven ratchet mounted in gear *30*.

When starting the engine, the rotation from the pneumatic turbine starter is transmitted to the gas generator rotor through clutch *A* of the pneumatic turbine starter and via gears *23*, *20*, *30*, *31*, *33*, *28* and drive shafts *32* and *27*.

As soon as the gas generator rotor rotational speed reaches the preset value, the turbine starter is cut out, and the pneumatic turbine starter drive clutch automatically breaks the kinematic coupling between the turbine starter with the gas generator rotor.

Through the drive shaft the power turbine rotates gear *17* of the rear reduction gear that through intermediate gear *16* imparts rotation to gear *13*. Gear *13* of the rear reduction gear is engaged with gear *9* of the first stage of the front reduction gear via three drive shafts *12*, *11* and *10* of the shaft line. Gear *9* of the front reduction gear is engaged simultaneously with three first-stage planetary gears *8* that, in their turn are engaged with first-stage rim gear *26*. Gear *26* is engaged with second-stage gear *6*, while the first-stage planetary gear housing *7* is engaged with propeller shaft *37*.

Second-stage gear *6* is engaged simultaneously with five second-stage planetary gears *5* that, in their turn, are engaged with second-stage rim gear *29*. Second-stage rim gear *29* is engaged with shaft *37*, and second-stage planetary gears housing *4* is coupled via the torquemeter with the reduction gear housing.

On propeller shaft *37* there is mounted gear *35*, that when engaged with gear *34*, imparts rotation to the torquemeter oil pump and to the propeller speed governor.

Gear *35* serves also as an inductor for propeller rotational speed sensors. In addition, gear *35* mounts inductor *36* of the propeller rotation phase sensor. Mounted on intermediate gear *16* is gear *15* that, when being engaged with gear *14*, imparts rotation to the generator.

On propeller shaft *37* there is mounted bearing *1* affected by propeller blades turning mechanism *3*. As soon as the propeller blades setting angle is changed, bearing *1* together with toothed racks *3* moves along shaft *37*, thus imparting rotation to toothed segments *2* of the propeller blade setting angle transmitter shaft drives.

REVIEW QUESTIONS

1. What is the purpose of compression process within of the GTE operating cycle?
2. What is the main task of any gas turbine engine compressor?
3. What are purposes of the compressed air bled from compressor in an aircraft?
4. Name the basic types of gas turbine engine compressors.
5. What main components does the centrifugal compressor unit consist of?
6. Define the term “axial flow compressor stage”.
7. Draw air velocity vector diagram for axial flow compressor stage.
8. Draw a graph of air pressure, velocity and temperature variation in an axial flow compressor stage.
9. What is the specific feature of the first stage of axial flow compressor?
10. In what engines is a centrifugal compressor used now?
11. What is the principle outline of air path in the centrifugal compressor?
12. What are the main components of the centrifugal compressor?
13. How does the centrifugal compressor diffuser work?
14. What is the centrifugal compressor air manifold needed for?
15. What is the main difference between the single-enter and double-enter impeller?
16. What pressure ratio is produced in the centrifugal compressor stage?
17. What is the range of centrifugal impeller tip circumference velocity?
18. What are advantages and disadvantages of the centrifugal compressor in comparison with the axial flow compressor?
19. What are main types of the axial flow compressors?
20. Where are dual- and triple-spool compressors used?
21. What is the particular feature of single-spool high pressure ratio axial flow compressor?
22. What are advantages and disadvantages of the axial flow compressor in comparison with the centrifugal compressor?
23. What does the abbreviation IGV mean?
24. Why doesn’t the airflow velocity and pressure change while air is flowing through the IGV?
25. How are air flow parameters (velocity, pressure and temperature) varied when air flows through the rotor blade row?

26. Where does the air flow to after leaving the trailing edge of the compressor rotor blade row?
27. How are air flow parameters (velocity, pressure and temperature) changed when air flows from the first to the last stage of the axial flow compressor?
28. What purposes are the stator vanes of the axial flow compressor used for?
29. Why are the compressor blades constructed with a twist?
30. What main numerical factor is used to estimate the rate of air pressure increase in a compressor unit? How can it be calculated?
31. Why does an axial flow compressor comprise several stages?
32. In what cases does an axial flow compressor consist of more than one spool?
33. What is the compressor stall? What are the reasons of compressor stall?
34. Name the main assemblies of the ТВ3-117ВМА-СБМ engine compressor unit.
35. How is the setting angle of the inlet guide vanes measured?
36. How is the ТВ3-117ВМА-СБМ engine compressor gas dynamic stability provided?
37. How many spools and how many stages does the ТВ3-117-ВМА-СБМ engine compressor unit consist of?
38. What are the engine accessory drive system intended for?
39. Where are inlet drive unit and accessory gear boxes located?

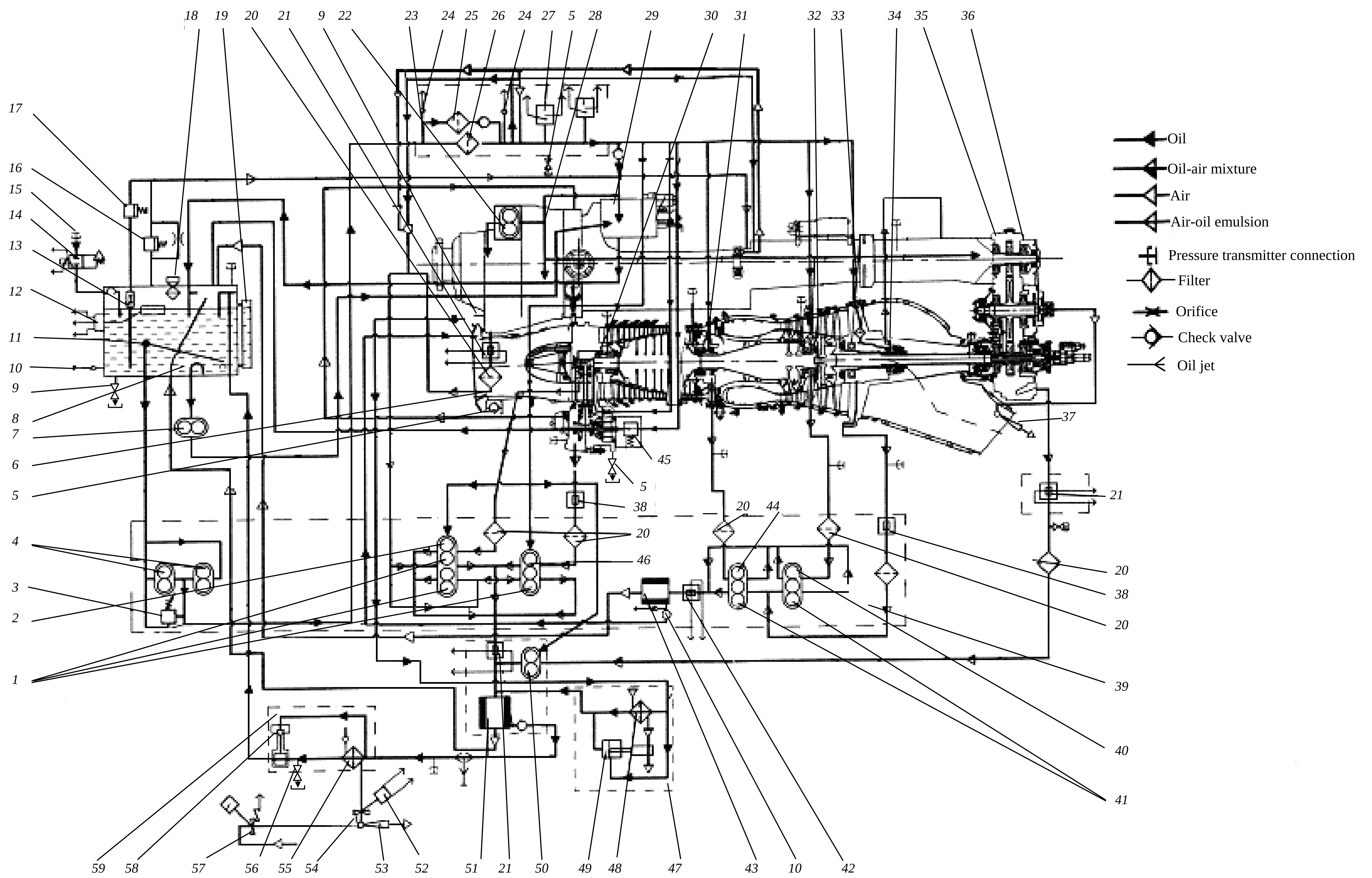


Fig. 12.1. Diagram of TB3-117BMA-CBM1 engine oil system:

1 — section of pumps for scavenging oil from front reduction gear; *2* — section of pump for scavenging oil from 1-st bearing support; *3* — pressure-reducing valve; *4* — oil pressure pump sections; *5* — oil drain valve; *6* — engine air inlet section; *7* — feathering pump; *8* — oil tank; *9* — drain valve; *10* — oil temperature probe; *11* — pendulum (gravity valve); *12* — oil quantity transmitter; *13* — oil tank gravity filling valve; *14* — solenoid valve; *15* — pressure-filling connection; *16* — safety valve; *17* — breathing valve; *18* — oil filler neck; *19* — sight gage; *20* — protective filter; *21* — chip detector; *22* — torquemeter oil pump; *23* — oil filter block; *24* — pressure transmitter; *25* — protective filter; *26* — filtering element; *27* — minimum oil pressure switch; *28* — front reduction gear; *29* — propeller speed governor; *30* — roller bearing of 1-st bearing support; *31* — ball bearing of 2-nd bearing support; *32* — roller bearing of 3-rd bearing support; *33* — ball bearing of 4-th bearing support; *34* — roller bearing of 5-th bearing support; *35* — rear reduction gear; *36* — centrifugal breather; *37* — drain plug; *38* — magnetic plug; *39* — oil pump block; *40* — section of pump for scavenging oil from 3-rd bearing support; *41* — sections of pump for scavenging oil from 4-th and 5-th bearing supports; *42* — chip detector; *43* — de-aerator; *44* — section of pump for scavenging oil from 2-nd bearing support; *45* — oil pump regulator; *46* — section of pump for scavenging oil from accessory drive gearbox; *47* — fuel-oil heat exchanger; *48* — heat exchanger; *49* — thermal valve; *50* — pump for scavenging oil from rear reduction gear; *51* — de-aerator with scavenge pumps; *52* — electric drive of ejector shutter; *53* — ejector; *54* — ejector shutter; *55* — heat exchanger; *56* — drain valve; *57* — electric driven air gate valve; *58* — thermal valve; *59* — air-oil heat exchanger unit

Chapter 9. COMBUSTION CHAMBER

9.1. Concept

The combustion chamber (Fig. 9.1) has the difficult task of burning large quantities of fuel, supplied through the fuel spray nozzles, with extensive volumes of air, supplied by the compressor, and releasing the heat in such a manner that the air is expanded and accelerated to give a smooth stream of uniformly heated gas at all conditions required by the turbine. This task must be accomplished with the minimum loss in pressure and with the maximum heat release for the limited space available.

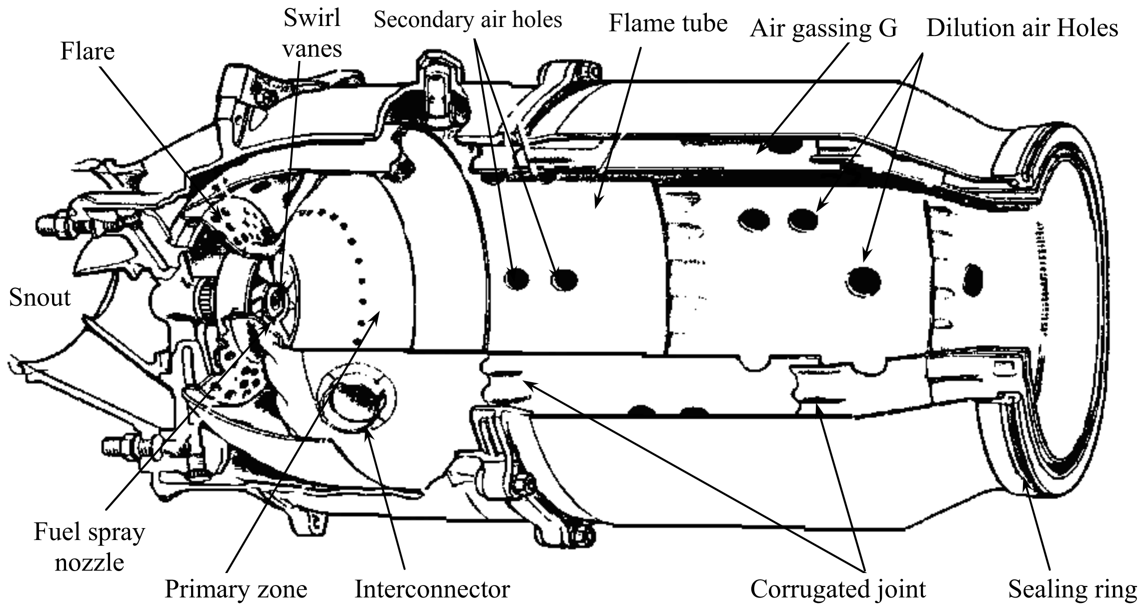


Fig. 9.1. An early combustion chamber

The amount of fuel added to the air will depend upon the temperature rise required. However, the maximum temperature is limited to within the range of 850 to 1700 °C by the materials from which the turbine blades and nozzles are made.

The air has already been heated to between 200 and 550 °C by the work done during compression, giving a temperature rise requirement of 650 to 1150 °C from the combustion process. Since the gas temperature required at the turbine varies with engine thrust, and in the case of the turbo-propeller engine upon the power required, the combustion chamber must also be capable of maintaining stable and efficient combustion over a wide range of engine operating conditions. Efficient combustion has become increasingly important because of the rapid rise in commercial aircraft traffic and the consequent increase in atmospheric pollution, which is seen by the general public as exhaust smoke.

9.2. Combustion process

Air from the engine compressor enters the combustion chamber at a velocity up to 130...150 meters per second, but because at this velocity the air speed is far too high for combustion, the first thing that the chamber must do is to diffuse it, i.e. decelerate it and raise its static pressure. Since the speed of burning kerosine at normal mixture ratios is only a few meters per second, any fuel lit even in the diffused air stream, which now has a velocity of about 30 meters per second, would be blown away. A region of low axial velocity has therefore to be created in the chamber, so that the flame will remain alight throughout the range of engine operating conditions.

In normal operation, the overall air/fuel ratio of a combustion chamber can vary between 45:1 and 130:1. However, kerosene will only burn efficiently at, or close to, a ratio of 15:1, so the fuel must be burned with only part of the air entering the chamber, in what is called a primary combustion zone. This is achieved by means of a flame tube (combustion liner) that has various devices for metering the airflow distribution along the chamber.

Approximately 20 per cent of the air mass flow is taken in by the snout or entry section (Fig. 9.1, 9.2).

Immediately downstream of the snout are swirl vanes and a perforated flare, through which air passes into the primary combustion zone. The swirling air induces a flow upstream of the centre of the flame tube and promotes the desired recirculation. The air not picked up by the snout flows into the annular space between the flame tube and the air casing.

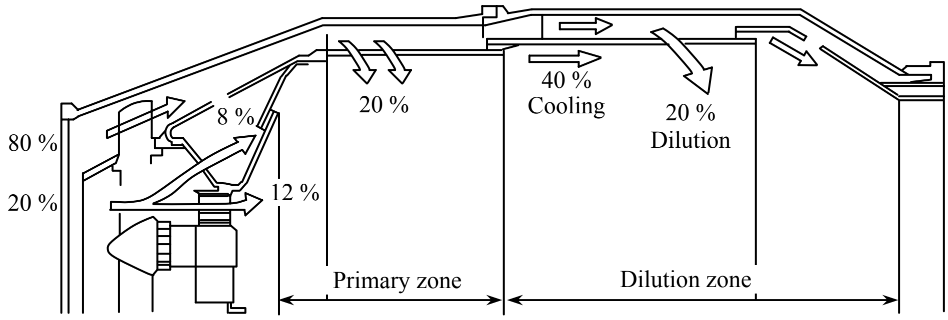


Fig. 9.2. Apportioning the airflow

Through the wall of the flame tube body, adjacent to the combustion zone, are a selected number of secondary holes through which a further 20 per cent of the main flow of air passes into the primary zone. The air from the swirl vanes and that from the secondary air holes interacts and creates a region of low velocity recirculation. This takes the form of a toroidal vortex, similar to a smoke ring, which has the effect of stabilizing and anchoring the flame.

The recirculating gases hasten the burning of freshly injected fuel droplets by rapidly bringing them to ignition temperature. It is arranged that the conical fuel spray from the nozzle intersects the recirculation vortex at its centre. This action, together with the general turbulence in the primary zone, greatly assists in breaking up the fuel and mixing it with the incoming air.

The temperature of the gases released by combustion is about 1800...2000 °C, which is far too hot for entry to the nozzle guide vanes of the turbine. The air not used for combustion, which amounts to about 60 per cent of the total airflow, is therefore introduced progressively into the flame tube. Approximately a third of this is used to lower the gas temperature in the dilution zone before it enters the turbine and the remainder is used for cooling the walls of the flame tube.

This is achieved by a film of cooling air flowing along the inside surface of the flame tube wall, insulating it from the hot combustion gases. A recent development allows cooling air to enter a network of passages within the flame tube wall before exiting to form an insulating film of air, this can reduce the required wall cooling airflow by up to 50 per cent.

Combustion should be completed before the dilution air enters the flame tube, otherwise the incoming air will cool the flame and incomplete combustion will result.

An electric spark from an igniter plug initiates combustion and the flame is then self-sustained.

The design of a combustion chamber and the method of adding the fuel may vary considerably, but the airflow distribution used to effect and maintain combustion is always very similar to that described.

9.3. Fuel supply

Fuel is supplied to the airstream by one of two distinct methods. The most common is the injection of a fine atomized spray into the recirculating airstream through spray nozzles. The second method is based on the pre-vaporization of the fuel before it enters the combustion zone.

In the vaporizing method (Fig. 9.3) the fuel is sprayed from feed tubes into vaporizing tubes which are positioned inside the flame tube.

These tubes turn the fuel through 180 degrees and, as they are heated by combustion, the fuel vaporizes before passing into the flame tube. The primary airflow passes down the vaporizing tubes with the fuel and also through holes in the flame tube entry section which provide Tans' of air to sweep the flame rearwards. Cooling and dilution air is metered into the flame tube in a manner similar to the atomizer flame tube.

9.4. Types of combustion chamber

There are three main types of combustion chamber in use for gas turbine engines. These are the multiple chamber, the tubo-annular (cannular) chamber and the annular chamber.

Multiple combustion chamber. This type of combustion chamber is used on centrifugal compressor engines and the earlier types of axial flow compressor engines.

The chambers are disposed around the engine (Fig. 9.4) and compressor delivery air is directed by ducts to pass into the individual chambers. Each chamber has an inner flame tube around which there is an air casing. The air passes through the flame tube snout and also between the tube and the outer casing.

The separate flame tubes are all interconnected. This allows each tube to operate at the same pressure and also allows combustion to propagate around the flame tubes during engine starting.

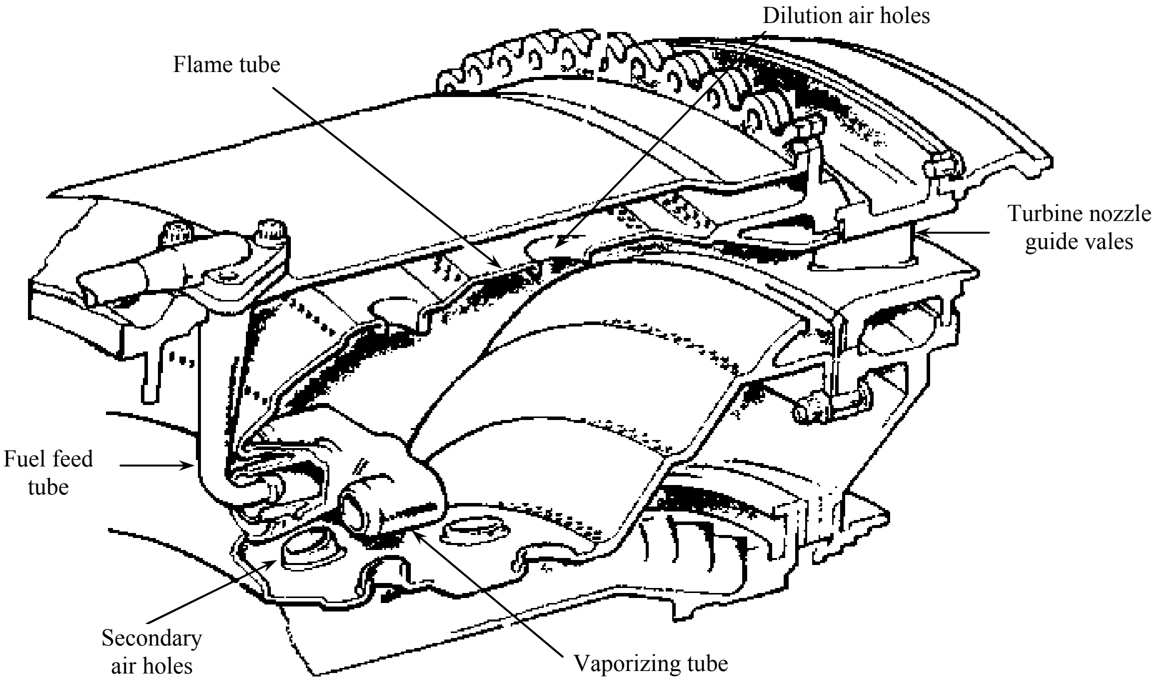


Fig. 9.3. A vaporizer combustion chamber

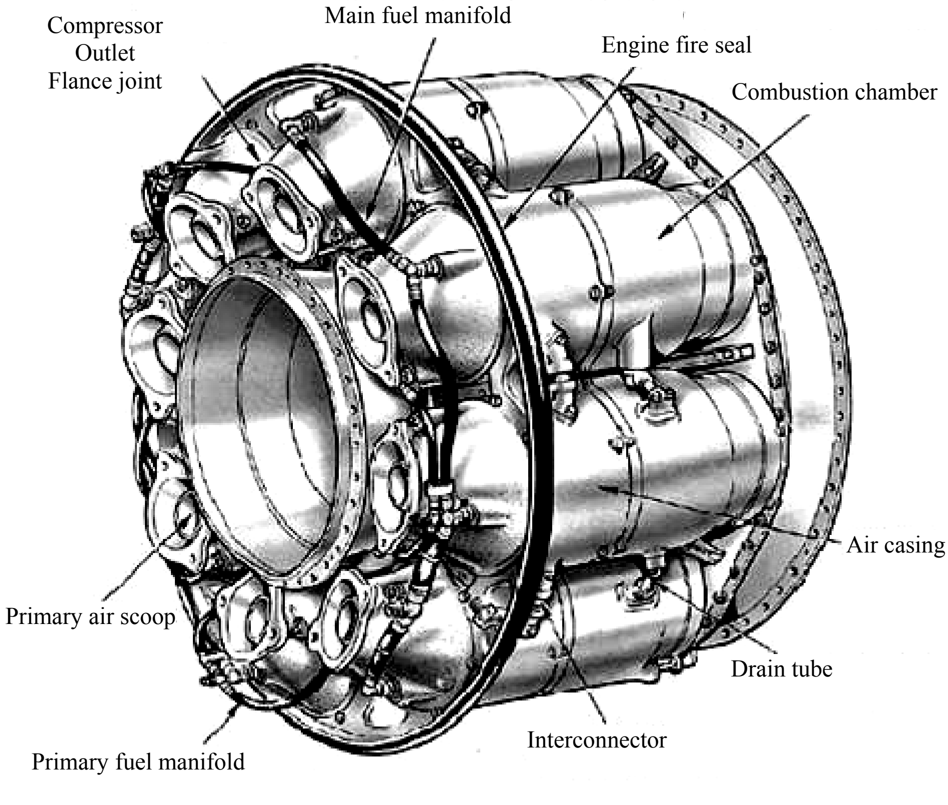


Fig. 9.4. Multiple combustion chambers

Tubo-annular combustion chamber. The tube-annular combustion chamber bridges the evolutionary gap between the multiple and annular types. A number of flame tubes are fitted inside a common air casing (Fig. 9.5). The airflow is similar to that already described. This arrangement combines the ease of overhaul and testing of the multiple system with the compactness of the annular system.

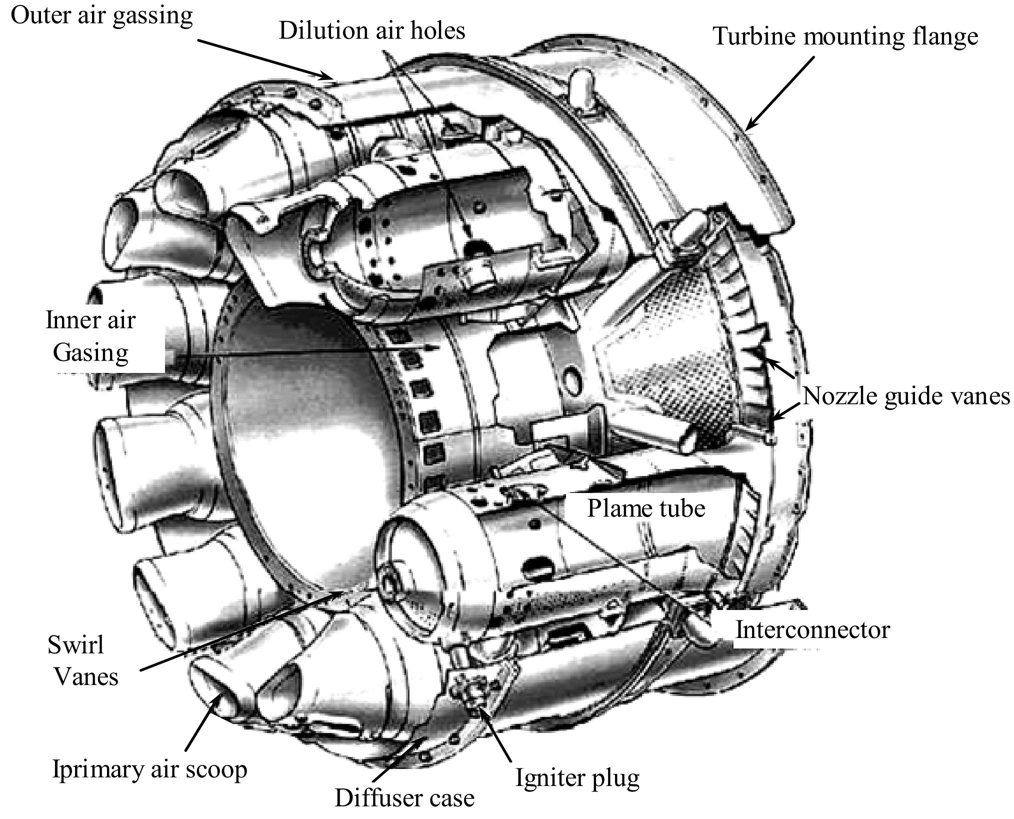


Fig. 9.5. Tubo-annular (cannular) combustion chamber

Annular combustion chamber. This type of combustion chamber consists of a single flame tube, completely annular in form, which is contained in an inner and outer casing (Fig. 9.6). The airflow through the flame tube is similar to that already described, the chamber being open at the front to the compressor and at the rear to the turbine nozzles.

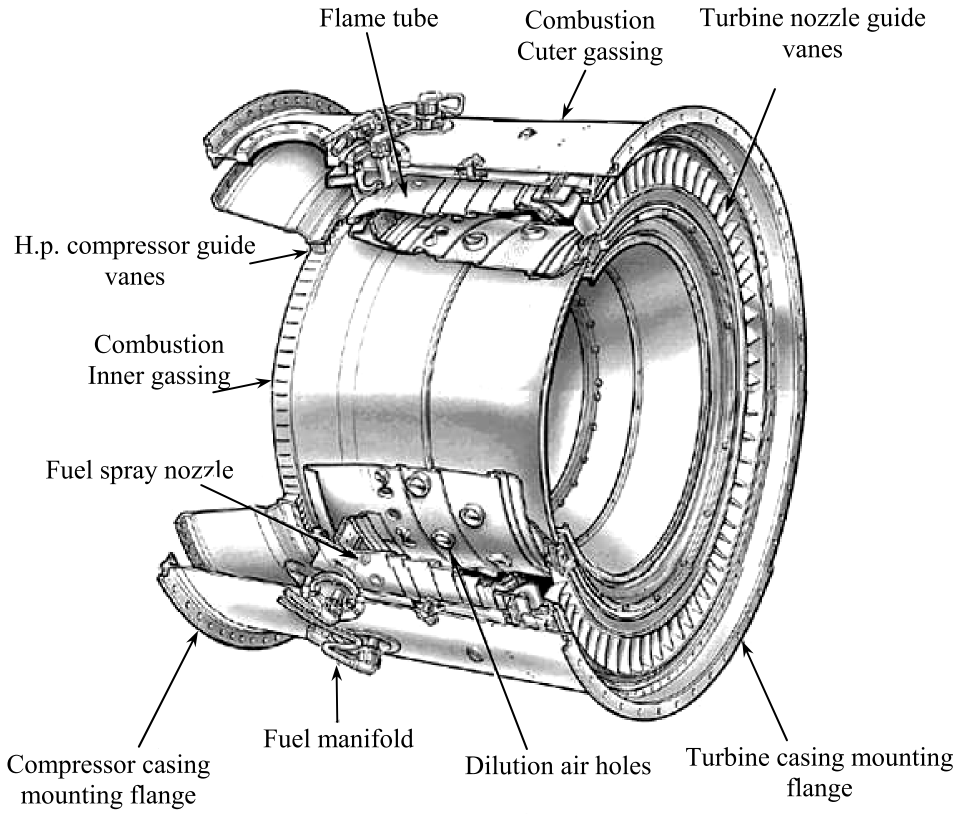


Fig. 9.6. Annular combustion chamber

The main advantage of the annular chamber is that, for the same power output, the length of the chamber is only 75 per cent of that of a tube-annular system of the same diameter, resulting in considerable saving of weight and production cost. Another advantage is the elimination of combustion propagation problems from chamber to chamber.

In comparison with a tube-annular combustion system, the wall area of a comparable annular chamber is much less; consequently the amount of cooling air required to prevent the burning of the flame tube wall is less, by approximately 15 per cent. This reduction in cooling air raises the combustion efficiency to virtually eliminate unburned fuel, and oxidizes the carbon monoxide to non-toxic carbon dioxide, thus reducing air pollution

The introduction of the air spray type fuel spray nozzle to this type of combustion chamber also greatly improves the preparation of fuel for combustion by aerating the over-rich pockets of fuel vapors' close to the spray nozzle; this results in a large reduction in initial carbon formation.

9.5. Combustion chamber performance

A combustion chamber must be capable of allowing fuel to burn efficiently over a wide range of operating conditions without incurring a large pressure loss. Combustion process in a combustion chamber may be characterized by means of several value among which main are combustion intensity, combustion efficiency and combustion stability. In addition, if flame extinction occurs, then it must be possible to relight. In performing these functions, the flame tube and spray nozzle atomizer components must be mechanically reliable.

The gas turbine engine operates on a constant pressure cycle, therefore any loss of pressure during the process of combustion must be kept to a minimum. In providing adequate turbulence and mixing, a total pressure loss varying from about 3 to 8 per cent of the air pressure at entry to the chamber is incurred.

Combustion intensity. The heat released by a combustion chamber or any other heat generating unit is dependent on the volume of the combustion area. Thus, to obtain the required high power output, a comparatively small and compact gas turbine combustion chamber must release heat at exceptionally high rates. For example, at take-off conditions a Rolls-Royce RB211-524 engine will consume 9.000 kg of fuel per hour. The fuel has a calorific value of approximately 37.000 kJ per kg, therefore the combustion chamber releases nearly 100.000 kJ per second. Expressed in another way, this is an expenditure of potential heat at a rate equivalent to approximately 150.000 horse-power.

Combustion efficiency is a ratio of heat output of combustion chamber to heat input to combustion chamber with fuel (heat input equals heat output plus heat wasted). The combustion efficiency of most gas turbine engines at sea-level takeoff conditions is almost 100 per cent, reducing to 98 per cent at altitude cruise conditions.

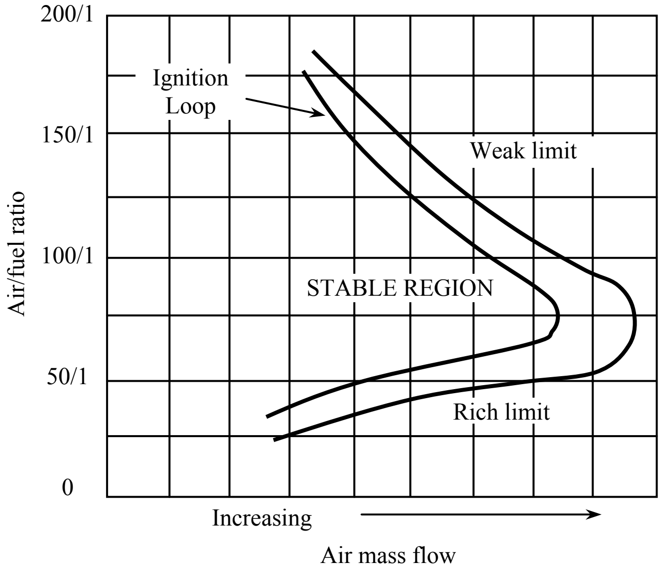


Fig. 9.7. Combustion stability limits

Combustion stability means smooth burning and the ability of the flame to remain alight over a wide operating range. For any particular type of combustion chamber there is both a rich and weak limit to the air/fuel ratio, beyond which the flame is extinguished. An extinction is most likely to occur in flight during a glide or dive with the engine idling, when there is a high airflow and only a small fuel flow, i.e. a very weak mixture strength. The range of air/fuel ratio between the rich and weak limits is reduced with an increase of air velocity, and if the air mass flow is increased beyond a certain value, flame extinction occurs.

A typical stability loop is illustrated in Fig. 9.7. The operating range defined by the stability loop must obviously cover the air/fuel ratios and mass-flow of the combustion chamber. The ignition process has weak and rich limits similar to those shown for stability in Fig. 9.7.

The ignition loop, however, lies within the stability loop since it is more difficult to establish combustion under 'cold' conditions than to maintain normal burning.

Emissions. The unwanted pollutants which are found in the exhaust gases are created within the combustion chamber. There are four main pollutants which are legislatively controlled: unburnt hydrocarbons (unburnt fuel), smoke (carbon particles), carbon monoxide and oxides of nitrogen. The principal conditions which affect the formation of pollutants are pressure, temperature and time.

In the fuel rich regions of the primary zone, the hydrocarbons are converted into carbon monoxide and smoke. Fresh dilution air can be used to oxidize the carbon monoxide and smoke into non-toxic carbon dioxide within the dilution zone.

Unborn hydrocarbons can also be reduced in this zone by continuing the combustion process to ensure complete combustion. Oxides of nitrogen are formed under the same conditions as those required for the suppression of the other pollutants. Therefore it is desirable to cool the flame as quickly as possible and to reduce the time available for combustion. This conflict of conditions requires a compromise to be made, but continuing improvements in combustor design and performance has led to a substantially 'cleaner' combustion process.

9.6. Materials

The containing walls and internal parts of the combustion chamber must be capable of resisting the very high gas temperature in the primary zone. In practice, this is achieved by using the best heat-resisting materials available, the use of high heat resistant coatings and by cooling the inner wall of the flame tube as an insulation from the flame.

The combustion chamber must also withstand corrosion due to the products of the combustion, creep failure due to temperature gradients and fatigue due to vibrational stresses.

9.7. The TB3-117BMA-CBM1 engine combustion chamber description and operation

The combustion chamber is intended to provide an effective fuel combustion in a compressed air flow delivered from a compressor, as well as to obtain a preset gas temperature upstream of the turbine. The combustion chamber (Fig. 9.8) is of annular straight flow type, positioned between the compressor and compressor turbine and consists of a combustion chamber, diffuser inner casing, flame tube and fuel manifold provided with twelve duplex fuel nozzles.

During engine start, the fuel is ignited in the flame tube by means of two igniter plugs mounted on the combustion chamber casing.

The flame tube outer fairing has nine flanges *3* (Fig. 9.9) for installation of the flame tube suspensions which serve to attach the flame tube to the combustion chamber casing.

Fuel is supplied to the manifold by two metering flows through adapter *25 (*see Fig. 9.8). Fuel is delivered to the primary fuel manifold of each fuel nozzle through inlet pipe. Then the fuel flows through filter to cone swirled. The fuel is forced through two inclined slots into the chamber of orifice-swirled from which it is discharged into the flame tube in the form of atomized sprays, out of fuel is delivered to the main manifold of each fuel nozzle through separate inlet pipe.

Then the fuel flows through the spiral passages and ejected into the flame tube in the form of atomized sprays. To prevent carbon deposition on orifice 5 and on the butt ends of nozzle housings, they are blown with compressed air delivered through the openings in nozzle housing.

Fuel is supplied to the primary fuel manifold of fuel nozzles both during engine starting and at all power ratings. To facilitate ignition in the combustion chamber during engine starting, compressed air is delivered to the main duct manifold of fuel nozzles from the air starter, which provides the effective atomization of fuel supplied via the primary fuel manifold of fuel nozzles.

Operation. Compressed air flows from the compressor into the front cavity *A* of the diffuser (see Fig. 9.8) wherein it gets braked and separated into two flows. One of the air flows enters combustion zone *B* of the flame tube, the other one is delivered for cooling the flame tube and the turbine.

The fuel atomized by the fuel nozzles burns in a swirled air flow, and hot gases get into mixing zone C of the flame tube. In the mixing zone the hot gases are mixed with the air delivered from the annular passages *D* and *E* through the openings made in the inner and outer mixer sections.

Effective cooling of the flame tube walls is ensured by the air passing inside the flame tube through inner and outer slots *F*. During false and unsuccessful starts the drained fuel gets accumulated in the combustion chamber casing bottom point and, from drain connection *23*, is drained via the fuel distributor into the drain tank. The fuel seeped through the adapter sealing rings is delivered directly to the drain tank.

The fuel manifold with fuel nozzles is installed in the combustion chamber and is attached to the combustion chamber casing with the help of three suspensions screwed in fuel nozzle casings and locked by washers. The fuel manifold is of an annular type and is equipped with twelve duplex fuel nozzles. The fuel manifold is protected by heat shield to prevent fuel from heating during the engine operation at high power ratings. The fuel nozzles are duplex, of a centrifugal type.

Each fuel nozzle is fitted with spraying and filtering elements secured by housing welded up to it.

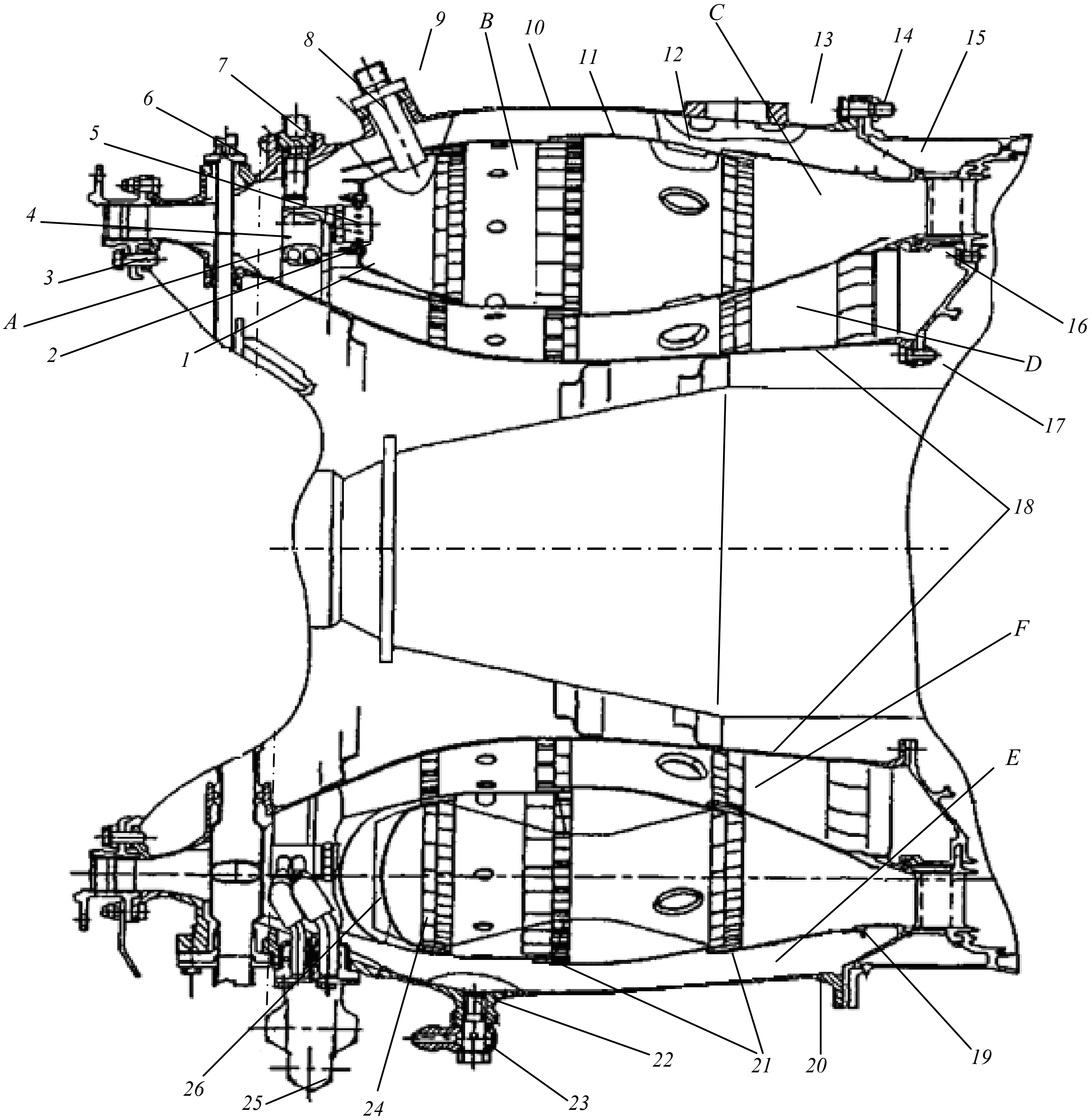


Fig. 9.8. Combustion chamber longitudinal section:
1 — swifter; *2* — thrust bushing; *3* — front inner flange; *4* — manifold; *5* — fuel nozzle;
6 — oil supply pipeline attachment flange; *7* — flame tube suspension attachment flange;
8 — igniter plug attachment flange; *9* — igniter plug; *10* — combustion chamber casing;
11 — flame tube; *12* — mixer outer section; *13* — flange of air bleed to AIS; *14* — rear outer flange;
15 — NGV front casing; *16* — NGV inner casing; *17* — rear inner flange; *18* — diffuser inner casing;
19 — outer back-up ring; *20* — diffuser outer section; *21* — outer corrugated ring; *22* — drainage pot;
23 — drainage connecting pipe; *24* — adapter; *25* — outer fairing;
26 — oil scavenge pipeline attachment flange;
A — diffuser front cavity; *B* — flame tube combustion zone;
C — flame tube mixing zone; *D* — annular inner passage;
E — annular outer passage; *F* — cooling slots

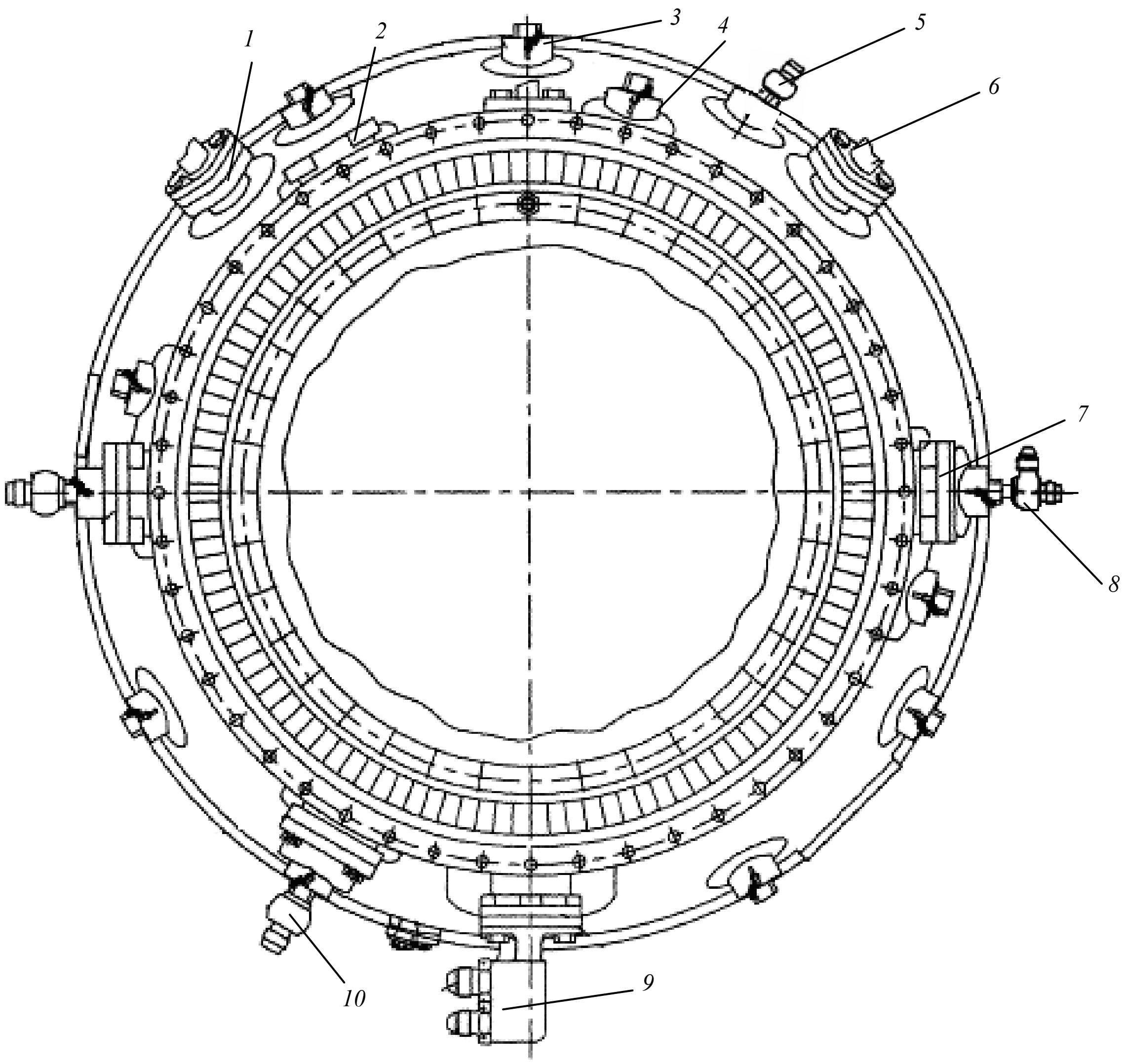


Fig. 9.9. Combustion chamber front view:
1 — igniter plug attachment flange; *2* — attachment flange of air bleed pipelines to ACS; *3* — flame tube suspension mounting flange; *4* — fuel manifold suspension mounting flange; *5* — air bleed-to-FCU connection; *6* — air bleed-to-FCS pipeline attachment flange; *7* — compressor second bearing air cavity breathing pipelines attachment flange; *8* — connection of air bleed to drainage system ejector; *9* — adapter; *10* — spare connection

REVIEW QUESTIONS

1. What is the main purpose of combustion chamber?
2. List the tree types of combustion chamber commonly used on gas turbine engines.
3. What is the temperature range of the air entering combustion chamber?
4. What is the range of the maximum gas temperature at the exit of combustion chamber?
5. Why is the air flow entering combustion chamber divided into primary and secondary airflows?
6. What is the primary air flow used for in the combustion chamber?
7. What is the secondary air flow used for in the combustion chamber? What part of air flowing into the combustion chamber is considered as secondary air flow?
8. What range of gas temperature is released by combustion of proper air/fuel mixture in the primary zone?
9. What is a range of axial air velocity entering the primary zone?
10. What amount of air is needed for full burning of one kilogram of kerosene in primary zone of combustion chamber?
11. Name and compare different methods of flame tube cooling.
12. What is combustion efficiency?
13. What methods of combustion chamber fuel supply are used in modern gas turbine engines?
14. What type of combustion chamber is used in the ТВ3-117-ВМА-СБМ engine?

Chapter 10. TURBINE UNIT

10.1. Concept

The turbine has the task of providing the power to drive the compressor and accessories and, in the case of engines which do not make use solely of a jet for propulsion, of providing shaft power for a propeller or rotor. The turbine transforms a portion of the kinetic energy and heat energy in the exhaust gases into mechanical work, so it can drive the compressor and the accessories. It does this by extracting energy from the hot gases released from the combustion system and expanding them to a lower pressure and temperature. High stresses are involved in this process, and for efficient operation, the turbine blade tips may rotate at speeds over 1.500 feet per second (450 meters per second). The continuous flow of gas to which the turbine is exposed may have an entry temperature between 850 and 1.500 degrees C. and may reach a velocity of over 2.500 feet per second (750 meters per second) in parts of the turbine.

There are two principle types of gas turbines: axial-flow turbines and radial-inflow turbines.

The type of turbine design used in almost all flight engines is the axial flow type, in which the products of combustion pass through the turbine vanes and blades, changing their angle momentarily, then returning to an axial direction.

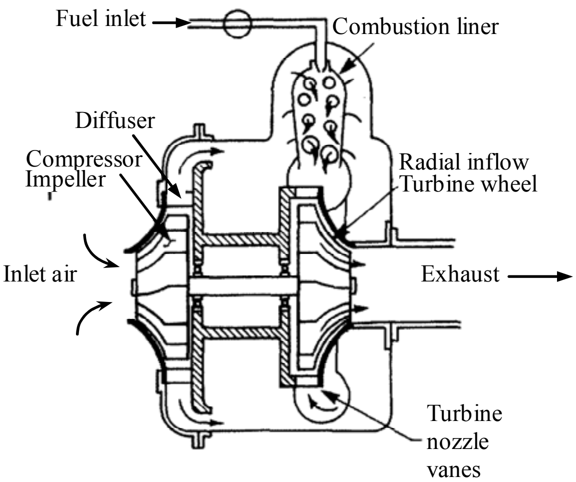


Fig.10.1. Operation principle of auxiliary power unit with radial-inflow turbine

Another type of turbine design is the radial-inflow turbine Operation principle if this turbine is shown in Fig. 10.1 and main components structure of radial-inflow turbine are shown in Fig 10.2. Like the centrifugal compressor, it has advantages of low cost and simplicity of design. Its primary application is in auxiliary gas turbine engines. It derives its name from the fact that gases flow through a stator vane assembly located at the radius of the turbine. The gas then flows inward from the tip area and finally exits at the center. This design is used because it extracts up to 100 percent of the kinetic energy from the flowing gases.

The radial inflow turbine has high single stage turbine efficiency but has poor multistage efficiency. It also has the disadvantage of low axial velocity discharge and short service life under high temperature loads due primarily to high centrifugal loads on the disc. These problems have not been solved for use of radial-inflow turbine in flight engines.

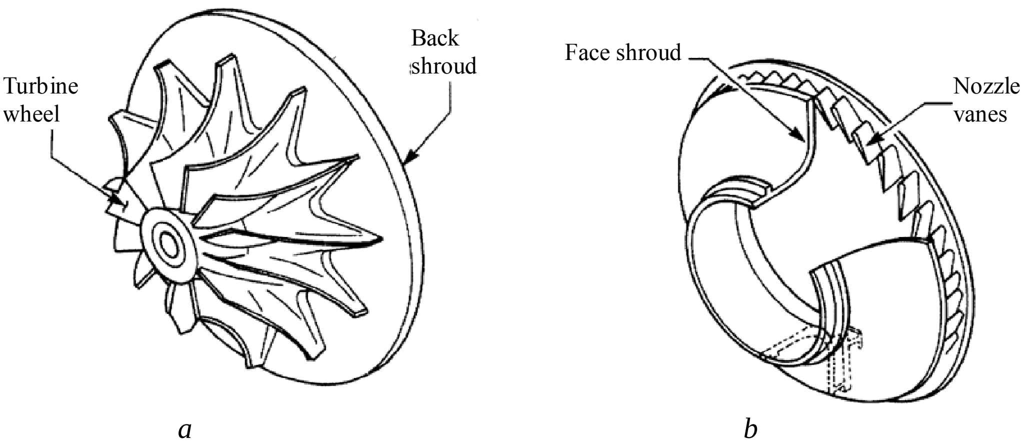


Fig. 10.2. Main components of the radial-inflow turbine:
a — turbine rotating wheel; *b* — stationary nozzle diaphragm and face shroud

Each axial turbine stage (Fig. 10.3 and Fig. 10.4), like a compressor stage, consists of one row of stationary vanes (stator vanes) and one row of moving blades (rotor blades), but, unlike a compressor stage, its stator vanes are located in front of the rotor blades. The term stator may be a bit misleading when discussing turbine assemblies, but it uses the same principle as a stator in a compressor, guiding the air onto the next stage of compression at the right angle.

It is also designed as a convergent duct accelerating the air onto the turbine disk. It is because of the latter purpose that it is known as a nozzle guide vane. The turbine nozzle assembly is also known as the turbine nozzle, nozzle diaphragm, or the turbine nozzle guide vane assembly.

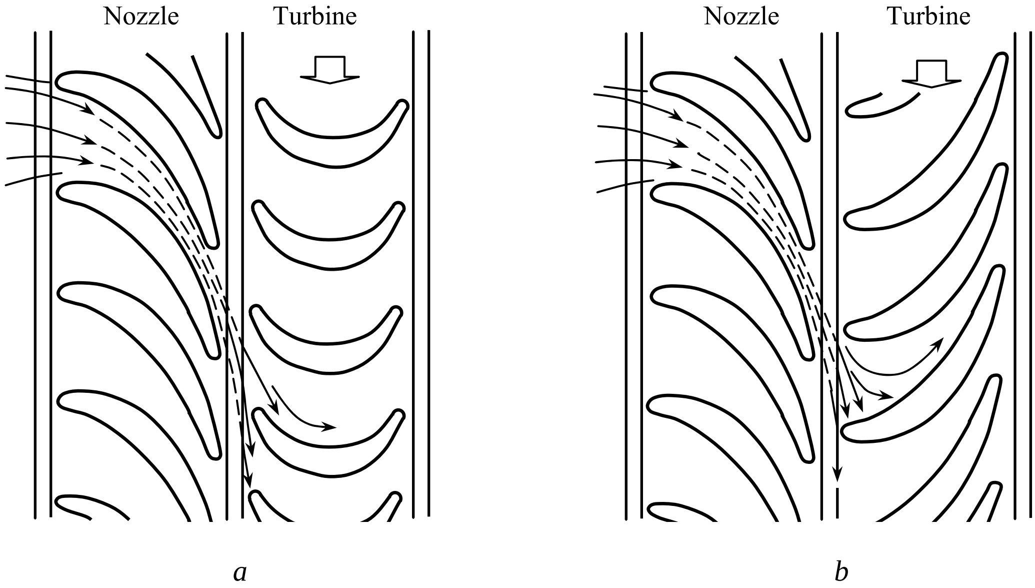


Fig. 10.3. Comparison between a pure impulse turbine and an impulse-reaction turbine:
a — impulse turbine stage; *b* — impulse-reaction turbine stage

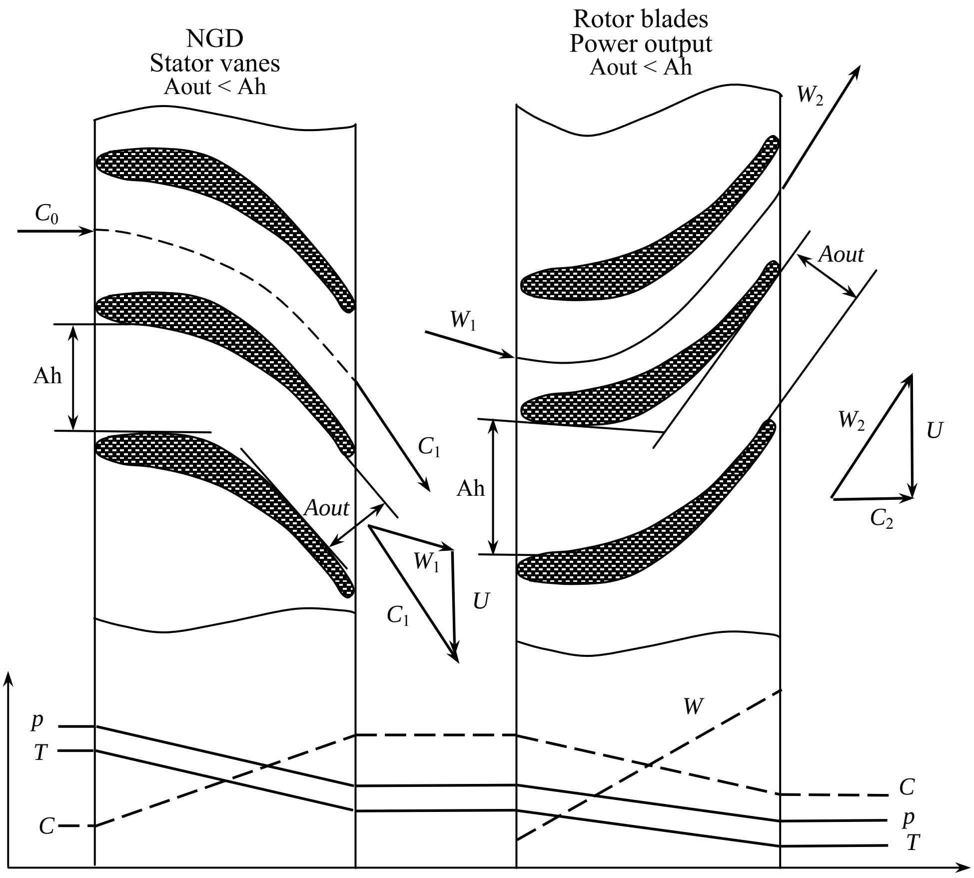


Fig. 10.4. To explanation of operating principles of gas turbine stage

The turbine nozzle guide diaphragm (NGD) is a series of airfoil-shaped vanes arranged in ring at the rear of the combustion section of a gas-turbine engine. Its function is to control the speed, direction, and pressure of the hot gases as they enter the turbine. NGD although stationary play a vital part in the operation of the turbine assembly. If the nozzle area is too large, the turbine will not operate at its optimum efficiency: if the area is too small, the nozzle will have a tendency to 'choke under maximum thrust operation.

To produce the driving torque, the turbine may consist of several stages. The number of stages depends upon the relationship between the power required from the gas flow, the rotational speed at which it must be produced and the diameter of turbine permitted.

The number of shafts, and therefore turbines, varies with the type of engine. High compression ratio engines usually have two shafts, driving high and low pressure compressors. On high by-pass ratio fan engines that feature an intermediate pressure system, another turbine may be interposed between the high and low pressure turbines, thus forming a triple-spool system. On some engines, driving torque is derived from a free-power turbine. This method allows the turbine to run at its optimum speed because it is mechanically independent of other turbine and compressor shafts.

The mean blade speed of a turbine has considerable effect on the maximum efficiency possible for a given stage output. For a given output the gas velocities, deflections, and hence losses, are reduced in proportion to the square of higher mean blade speeds. Stress in the turbine disc increases as the square of the speed, therefore to maintain the same stress level at higher speed the sectional thickness, hence the weight, must be increased disproportionately. For this reason, the final design is a compromise between efficiency and weight. Engines operating at higher turbine inlet temperatures are thermally more efficient and have an improved power to weight ratio. By-pass engines have a better propulsive efficiency and thus can have a smaller turbine for a given thrust. The design of the nozzle guide vane and turbine blade passages is based broadly on aerodynamic considerations, and to obtain optimum efficiency, compatible with compressor and combustion design, the nozzle guide vanes and turbine blades are of a basic aerofoil shape.

There are three types of turbine; impulse, reaction and a combination of the two known as impulse-reaction. In the impulse type (see Fig. 10.3, *a*) the total pressure on the vanes which, because of their convergent shape, increase the gas velocity whilst reducing the pressure.

The gas is directed onto the turbine blades which experience an impulse force caused by the impact of the gas on the blades. In the reaction type the fixed nozzle guide across each stage occurs in the fixed nozzle guide vanes are designed to alter the gas flow direction without changing the pressure. The converging blade passages experience a reaction force resulting from the expansion and acceleration of the gas. Normally gas turbine engines do not use pure impulse or pure reaction turbine blades but the impulse-reaction combination (see Fig. 10.3, *b*). This type of turbine is driven by the impulse of the gas flow and its subsequent reaction as it accelerates through the converging blade passages. The proportion of each principle incorporated in the design of a turbine is largely dependent on the type of engine in which the turbine is to operate, but in general it is about 50 percent impulse and 50 percent reaction. Impulse-type turbines are used for cartridge and air starters.

10.2. Energy transfer from gas flow to turbine

From the description contained above, it will be seen that the turbine depends for its operation on the transfer of energy between the combustion gases and the turbine. This transfer is never 100 percent because of thermodynamic and mechanical losses.

When the gas is expanded by the combustion process (Fig. 10.4), it forces its way into the discharge nozzles of the turbine with speed C_0 which is increased up to about 1000 feet per second (about 300 meters per second). When the gas stream flows through the nozzle guide vanes passages it is accelerated (because of shape of these passages is convergent and outlet area A_{out} is considerable less in comparison with inlet area A_{in}), so that its absolute velocity C_i is increased up to the speed of sound which, at the gas temperature, is about 2.500 feet per second (about 750 meters per second). At the same time the gas flow is given a 'spin' or 'whirl' in the direction of rotation of the turbine blades by the nozzle guide vanes. After NGD gas flows to turbine wheel, which is a row of rotor blades attached to a disc. The turbine blades are of an aerofoil shape, designed to provide passages between adjacent blades that give a steady acceleration of the flow up to the 'throat', where the area is smallest ($A_{out} < A_{in}$) and the velocity reaches that required at exit to produce the required degree of reaction. On impact with the blades and during the subsequent reaction through the blades, energy is absorbed, causing the turbine to rotate at high speed and so provide the power output for driving the turbine shaft and compressor. Relative gas velocity W is decreasing, energy of gas flow is output, and its total pressure p and temperature T are decreasing. The torque or turning power applied to the turbine is governed by the rate of gas flow and the energy change of the gas between the inlet and the outlet of the turbine blades. The design of the turbine is such that the whirl will be removed from the gas stream so that the flow at exit from the turbine will be substantially straightened out and its absolute velocity in this section C_2 is about axial directed. This is designed to give an axial flow into the exhaust system. Excessive residual whirl reduces the efficiency of the exhaust system and also tends to produce jet pipe vibration which has a detrimental effect on the exhaust cone supports and struts. It will be seen that the nozzle guide vanes and blades of the turbine are 'twisted', the blades having a stagger angle that is greater at the tip than at the root (Fig. 10.5).

The reason for the twist is to make the gas flow from the combustion system do equal work at all positions along the length of the blade and to ensure that the flow enters the exhaust system with a uniform axial velocity.

This results in certain changes in velocity, pressure and temperature occurring through the turbine.

The 'degree of reaction' varies from root to tip, being least at the root and highest at the tip, with the mean section having the chosen value of about 50 percent.

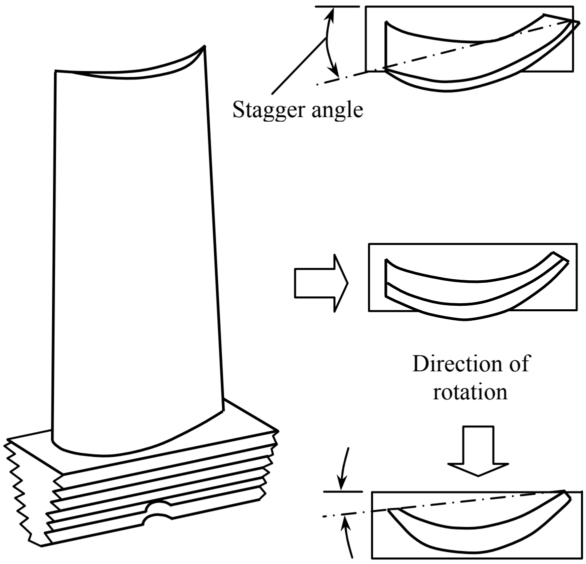


Fig. 10.5. A typical turbine blade showing twisted contour

10.3. Turbine components construction

The basic components of the turbine (Fig. 10.6) are the nozzle guide vanes *7*, the turbine casing *2*, the turbine blades *3* and the turbine discs *4* with the turbine shaft *5*. The rotating assembly is carried on bearings *6* mounted in the turbine casing and the turbine shaft may be common to the compressor shaft or connected to it by a self-aligning coupling.

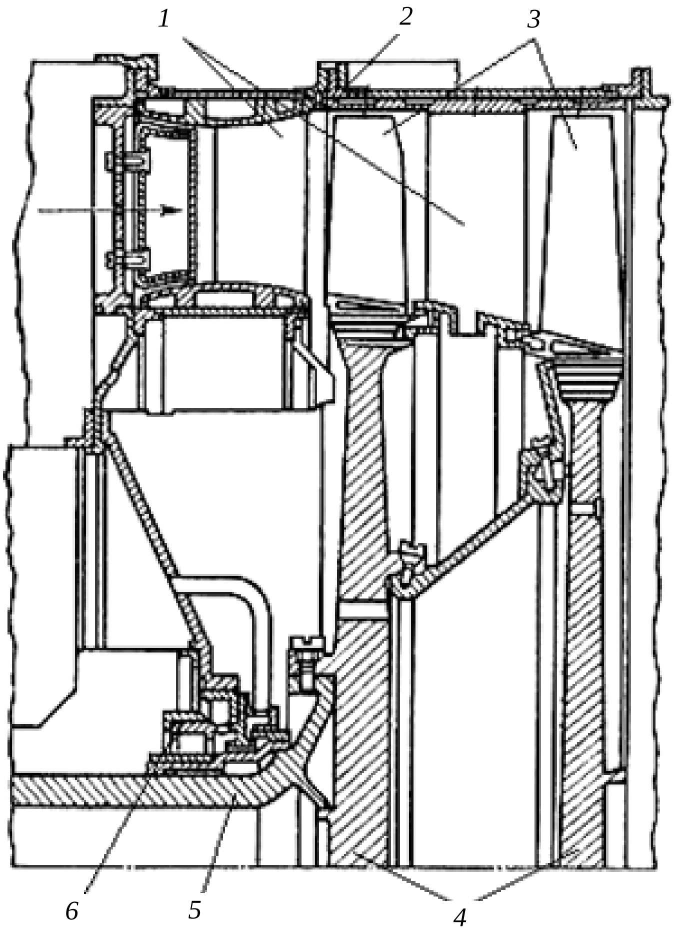


Fig. 10.6. A two-stage axial turbine assembly:
1 — nozzle guide vanes; *2* — turbine casing; *3* — turbine blades; *4* — turbine discs; *5* — turbine shaft; *6* — bearing

The nozzle guide vanes are of an aerofoil shape with the passage between adjacent vanes forming a convergent duct. The vanes are located in the turbine casing in a manner that allows for expansion. The nozzle guide vanes are usually of hollow form and may be cooled by passing compressor delivery air through them to reduce the effects of high thermal stresses and gas loads. The gas temperature and pressure is reduced as it passes through each turbine stage.

As a result, each successive turbine stage must be large to produce an equal share of work. The turbine blade consists of two main parts: profile part and root. The profile part is of an airfoil shape, designed to provide passes between adjacent blades that give a steady acceleration of the gas flow. The actual area of each blade cross-section is fixed by the permitted stress in the material used. High efficiency demands thin trailing edges to the sections, but a compromise has to be made so as to prevent the blades cracking due to the temperature changes during engine operation.

The profile part of turbine blade may be either open or shrouded at their ends, and both types of blades may be used in an engine (Fig. 10.7).

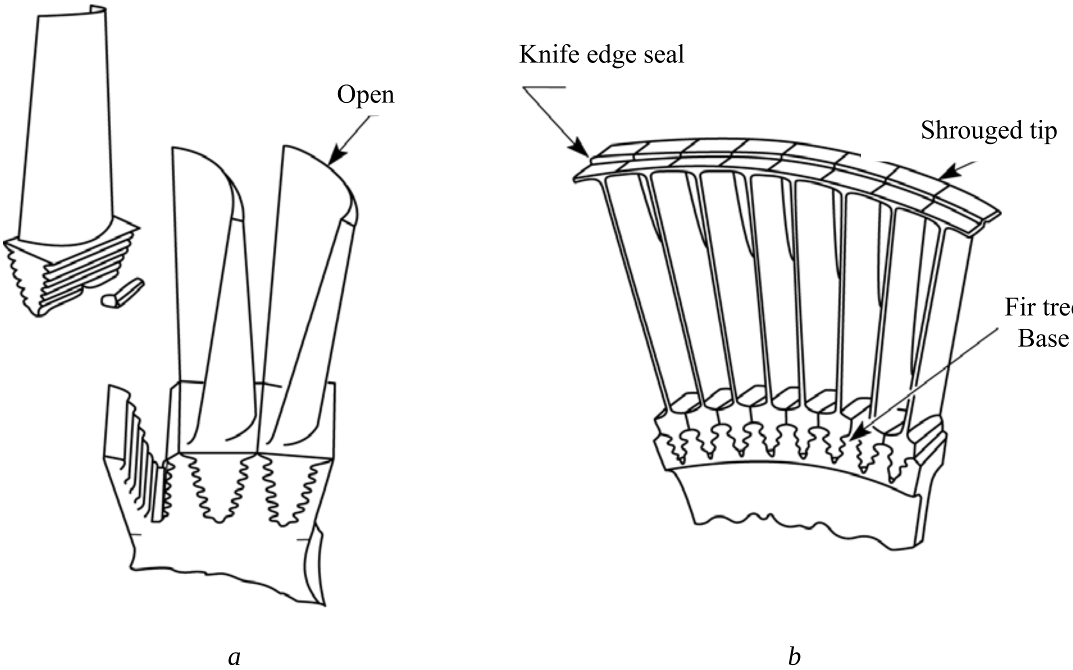


Fig. 10.7. Open and shrouded blades and its attaching to disc:
a — open end type blade; *b* — shrouded type blade

Open end blades are used in the high-speed wheels, and shrouded blades are found in the wheels having slower rotational speeds. Shrouded blades form a band around the perimeter of the wheel, which helps reduce blade vibration, and the weight of the shrouded tip is offset by the blades being thinner and more efficient. A knife-edge seal around the outside of the shroud reduces air losses at the blade tip, keeps the airflow in an axial direction, and minimizes radial losses. The knife-edge seal fits with a close tolerance into a shroud ring mounted in the outer turbine case.

Many modern engines have air-cooled turbine stator vanes and rotor blades. This cooling allows the turbines to operate at much higher temperatures than they could if they were not cooled.

The root of turbine blade serves to attach blade to the turbine disc. The method of attaching the blades to the turbine disc is of considerate importance, since the stress in the disc around the fixing or in the blade root has an important bearing on the limiting rim speed. The most commonly used method is the "fir-tree" design we see in Fig. 10.7. This type of fixing involves very accurate machining to insure that the loading is shared by all the serrations. The blade is free in the serrations when the turbine is stationary and is stiffened in the root by centrifugal loading when the turbine is rotating. The attachment of turbine blades to the turbine disk is usually accomplished by means of "fir-tree" slots broached in the rim of the disk and matching bases cast or machined on the blades. After the blades are inserted in the slots on the rim of the turbine disk, they are held in place by means of metal tabs which are bent over the bases of the blades.

Among the obstacles in the way of using higher turbine entry temperatures have always been the effects of these temperatures on the nozzle guide vanes and turbine blades. The high speed of rotation which imparts tensile stress to the turbine disc and blades is also a limiting factor. Therefore, heat resistance is the property most required from materials used for this components. Nickel alloys are used for nozzle guide vanes, although cooling is required to prevent melting.

A turbine disc or wheel, to which the turbine blades are attached, has to rotate at high speed in a relatively cool environment and is subjected to large rotational stresses. The limiting factor which affects the useful disc life is its resistance to fatigue cracking. Therefore, nickel based alloys are currently used for discs and turbine blades.

10.4. The TB3-117BMA-CBM1 engine turbine

The turbine of the TB3-117BMA-CBM1 engine is a follow-on of the TB3-117BMA helicopter engine turbine with modifications aimed at gaining higher characteristics and ensuring total service life time.

The engine TB3-117BMA-CBM1 turbine (Fig. 10.8) is an axial flow, reaction, four-stage unit that converts the gas flow energy into mechanical rotational energy of the engine compressor, engine accessory drives and propeller. The turbine is situated aft of combustion chamber *2*. The turbine unit consists of a two-stage compressor drive turbine assembly and of a two-stage power turbine assembly.

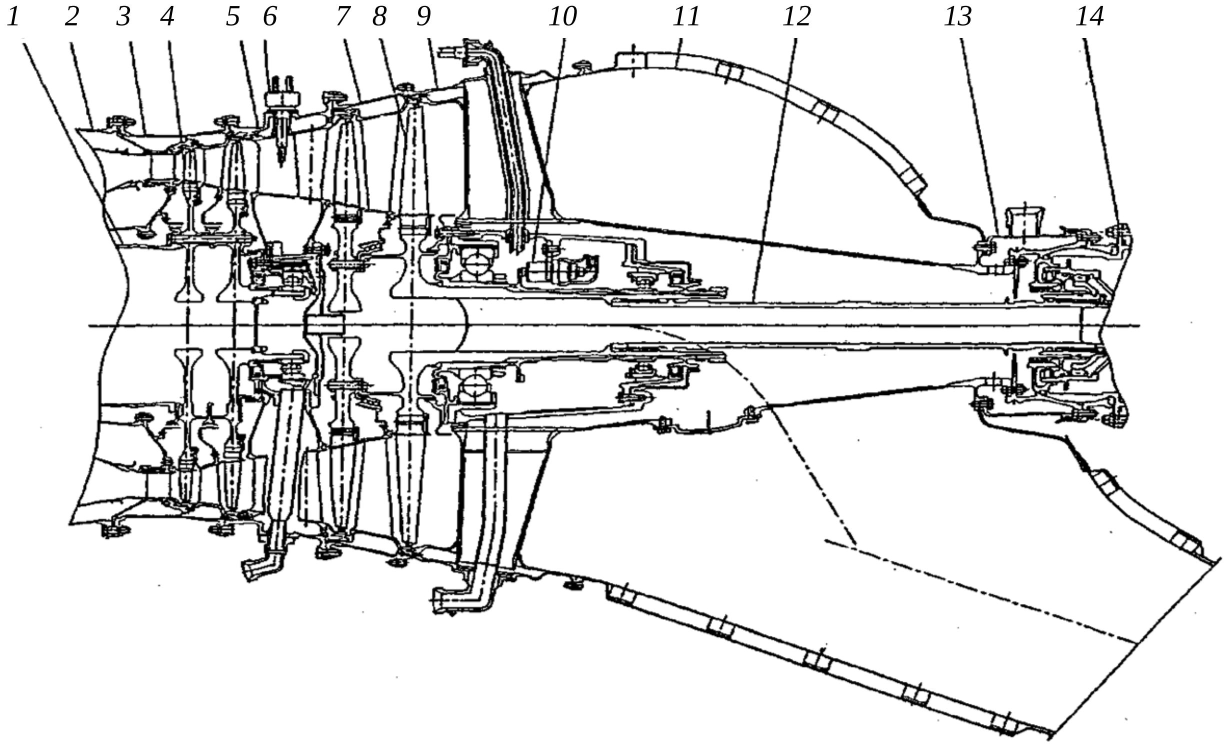


Fig. 10.8. Turbine and exhaust units:
1 — compressor turbine rotor shaft; *2* — combustion chamber; *3* — compressor turbine stator; *4* — compressor turbine rotor; *5* — compressor drive turbine bearing casing; *6* — thermocouple; *7* — power turbine stator; *8* — power turbine rotor; *9* — power turbine bearing casing; *10* — power turbine rotor rotational speed sensor; *11* — exhaust unit; *12* — drive shaft; *13* — receiver; *14* — rear reduction gear casing

Each of the turbine assembly consists, respectively, of rotors *4* and *8*, stators *5* and *7*, bearing casings *5* and *9*. Compressor drive turbine bearing casing *5* consists of welded outer and inner rings. The outer ring has attachment flanges for compressor drive turbine outlet gas temperature measuring thermocouples *6*. The inner ring is unit-casted of alloy ЭИ898-ВД.

Compressor drive turbine transmits the torque to the compressor rotor and the front accessory drive gearbox via shaft *1*. The compressor drive turbine rotor assembly consists of the shaft *1*, two discs with the rotor blades, and turbine casing *5*. The shaft *1* is attached to first stage disc by means of *12* conical bolts with nuts. These bolts serve also to attach the second stage disc to the first disc. The second stage disc of compressor turbine has a rear hub which serves to support the turbine rotor on the roller bearing.

The rotor blades of both stages are noncooled. First stage rotor blades are made of heat resistant alloy by way of directed solidification and second stage rotor blades are made of heat resistant alloy ЖС-6У. Outer platforms with knife-edge ribs, which are located on their tips, form shroud, which serves to seal clearance between the rotor blades and stator case, and thus to minimize tip leakage of gas through this clearance.

The compressor turbine stator consists of outer casing *3* and two turbine guide diaphragms. The first stage stator vanes are cooled by compressor discharge air entering the vane compartments through inserts in outer ends of vanes and exiting through the holes located near the vanes trailing edges. The hollow second stage stator vanes are cooled by compressor discharge air entering the vane compartments through holes in outer ends of vanes and exiting through the holes located in inner end of the vanes. The major parts of the first stage nozzle diaphragm are several nozzle segments, outer support ring and inner ring. Each nozzle segment is assembly of four vanes and inner and outer platforms, which are cast as single-piece construction. The second stage nozzle diaphragm is a ring shaped cast single-piece construction, which is attached to the turbine casing *3*.

The power turbine rotor transmits the torque to the propeller via rear reduction gear and transmission.

The power turbine rotor assembly *8* consists of two discs with blades and turbine shaft and drive shaft *12*. The first disc of this turbine is attached to the second disc by means of several short conical bolts. The power turbine shaft is forged as single-piece construction with last stage disc. This shaft is installed on two bearings, mounted inside of the power turbine bearing casing. The power turbine shaft is spline coupled with drive shaft *12*. Drive shaft *12* transmits the torque to rear reduction gear. The rotor blades of power turbine, like a mentioned above compressor turbine rotor blades, are noncooled and shrouded.

The power turbine stator consists of outer casing *7* and two turbine guide diaphragms. The stator vanes are hollow but noncooled. The turbine guide diaphragms are the ring shaped cast single-piece constructions, which are attached to the turbine casing *7*.

The power turbine bearing casing *9* comprises outer ring, six radial struts and inner conical beam. Rigid casing of ball and roller bearings of this turbine is mounted inside of the inner cavity conical beam.

The rear flange of power turbine bearing casing *9* is connected with rear reduction gear casing *14*.

Exhaust unit *11* is attached by means of its flanges, front and rear, to the respective flanges of power turbine bearing casing *9*. Exhaust unit *11* forms the flow sections of the engine and discharges the gas flow from the turbine to the atmosphere. The exhaust unit rear flange is equipped with receiver *13*, which ensures the air delivery for cooling power turbine bearing casing *9*.

REVIEW QUESTIONS

1. What is the task of the turbine? What main parts does the turbine unit include?
2. How many types of gas turbines do you know? Name them and explain their difference.
3. Define the term “turbine stage”.
4. In what cases do we use several turbine stages? Name the advantages of using these turbines.
5. What is the function of a turbine nozzle guide diaphragm?
6. Explain, how is the internal energy of gas flow transferred into mechanical work in the turbine unit?
7. What is the shape of the nozzle guide intervanes passage?
8. What main parts of the turbine blade do you know?
9. Why are turbine vanes and blades cooled? How is this cooling achieved?
10. What main losses take place in the turbine unit?
11. Draw velocity diagram for stator vanes and rotor blades of the reactive turbine stage.
12. Show graphically variation of pressure, velocity and temperature in the turbine stage passage.
13. Describe the turbine blade-to-disk fitting.
14. Describe the turbine and exhaust unit of the ТВ3-117ВМА-СБМ1 engine structure.
15. What is the purpose of the ТВ3-117ВМА-СБМ1 engine free turbine?
16. What purpose are the knife-edge ribs on turbine rotor elements used for?

Chapter 11. EXHAUST SECTION OF GAS TURBINE ENGINES

11.1. Concept

Exhaust section is the one of five main obligatory units of any type of gas turbine engine. It serves to direct the flow of hot gases rearward in such a manner as to prevent turbulence and to accelerate hot gases before they are discharged directly into the outside atmospheric air. Increasing the velocity of the working fluid (hot gases) increases momentum of gases which results in maximal propulsive force of the engine. The exhaust system must be capable of withstanding high gas temperatures and is therefore manufactured from nickel or titanium. It is also necessary to prevent any heat from being transferred to the surrounding aircraft structure. This is achieved by passing ventilating air around the jet pipe, or by covering the exhaust system with an insulating blanket.

The exhaust section is located directly behind the turbine section and in general case includes the exhaust collector, tailpipe (if required), and jet nozzle or exhaust pipe. In case of turbojet and turbofan engines the exhaust section includes some additional components that serve to control the direction of the engine thrust, to reduce noise level and some other functions. These additional components will be discussed later.

The exhaust unit design depends on desired flight speed of aircraft for which the engine is designed and location of engine on the aircraft. The simplest exhaust unit is used for subsonic turbojet engine mounted under the wing or at the rear portion of fuselage (Fig. 11.1). In this case exhaust unit consists of only two structural components: tail cone and exhaust nozzle. Tail cone protects the rear side of turbine disk against overheating by hot gases and at the same time gives more streamline shape to gas channel to reduce hydraulic losses when ring-shaped channel is transformed into round-shaped channel.

Convergent shaped exhaust nozzle acts as an orifice, the size of which determines the velocity of the gases as they flow out from the engine.

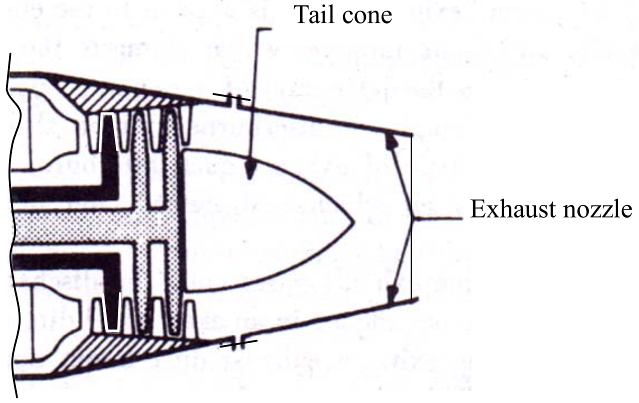


Fig. 11.1. The simplest convergent exhaust duct of pure turbojet engine

11.2. Subsonic turbojet engine exhaust unit

In general case main parts of exhaust system of pure turbojet engine (Fig. 11.2) are the exhaust cone assembly, the jet pipe and the exhaust or propelling nozzle. The exhaust assembly collects the exhaust gases discharged from the turbine buckets and gradually converts them into a solid jet. In performing this, the velocity of the gases is decreased slightly and the pressure increased due to diverging passage between the outer duct and inner cone.

Gas from the engine turbine enters the exhaust system at velocities of 330 to 370 m/s, but because velocities of this order produce high friction losses, the speed of flow is decreased by diffusion. This is accomplished by having an increasing passage area between the exhaust cone and the outer wall, as shown in Fig. 11.1 and Fig. 11.2. The cone also prevents the exhaust gases from flowing across the rear face of the turbine disk. It is usual to hold the velocity at the exhaust unit outlet to a Mach number of about 0.5, or approximately 290 m/s. Additional losses result from the residual whirl (swirl) velocity in the gas stream from the turbine. To reduce these losses, the turbine rear struts (straightening vanes) in the exhaust unit are designed to straighten out the flow before the gases pass into the jet pipe. The exhaust cone assembly consists of an outer shell or duct, an inner cone, three or four radial hollow struts and the necessary number of rods, located inside of the hollow struts to aid the struts in supporting the inner cone from the outer duct.

The outer duct or shell is usually made of stainless steel and is attached to the rear flange of turbine case.

The exhaust gases pass from jet pipe into the atmosphere through the propelling nozzle, which is a convergent duct, thus increasing the gas velocity. In a turbojet engine, the exit velocity of the exhaust gases is subsonic at low-thrust conditions only.

But during most operating conditions for turbojet engine with high level of working process parameters (pressure ratio more than 9:1 and gas temperature more than 1200 K and at high flight velocity), the exit velocity reaches the speed of sound, and the propelling nozzle is then said to be "**choked**" — that is, no further increase in velocity can be obtained unless the temperature is increased.

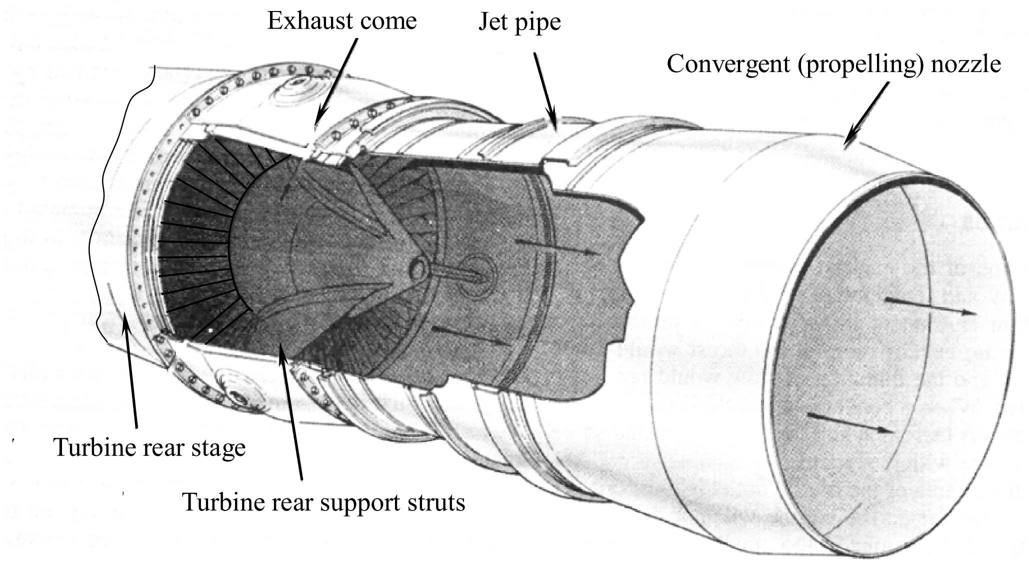


Fig. 11.2. Exhaust unit of subsonic pure turbojet engine

As the upstream total pressure is increased above the value at which the propelling nozzle becomes "choked", the static pressure of the gases at the exit increases above atmospheric pressure. This pressure differential across the nozzle provides what is known as **pressure thrust**. Pressure thrust is additional thrust added to the thrust obtained from the momentum change of the gas stream. With the convergent type of nozzle, a waste of energy occurs, since the gases leaving the exit do not expand rapidly enough to immediately achieve outside air pressure.

11.3. Convergent-divergent exhaust unit of supersonic turbojet engine

Whenever the engine pressure ratio is high enough to produce exhaust gas velocities which might exceed Mach 1 at the engine exhaust nozzle, more thrust can be gained by using a convergent-divergent type of nozzle (Fig. 11.3). The advantage of a convergent-divergent nozzle is greatest at high Mach numbers because of the resulting higher pressure ratio across the engine exhaust nozzle. To ensure that a constant weight or volume of a gas will flow past any given point after sonic velocity is reached, the rear part of a supersonic exhaust duct is enlarged to accommodate the additional weight or volume of a gas that will flow at supersonic rates. If this is not done, the nozzle will not operate efficiently. This section of the exhaust duct is known as divergent.

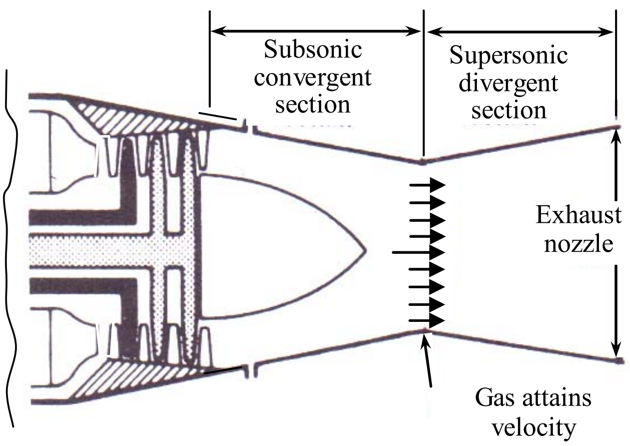


Fig. 11.3. Supersonic convergent-divergent type of exhaust duct (nozzle)

When a divergent duct is used in combination with a conventional exhaust duct, it is called a convergent-divergent exhaust duct. In the convergent/divergent, or C-D nozzle, the convergent section is designed to handle the gases while they remain subsonic, and to deliver the gases to the throat of the nozzle just as they attain sonic velocity.

The divergent section handles the gases, further increasing their velocity, after they emerge from the throat and become supersonic.

11.4. Bypass engine exhaust unit

The bypass engine has two gas streams to eject to the atmosphere: the cool bypass airflow and the hot turbine discharge gases. Construction of exhaust unit of bypass engine depends very considerably on value of bypass ratio. In low-bypass-ratio engines (when bypass ratio is not more than 2:1), the two flows are combined by a **mixer unit**, as illustrated in Fig. 11.4, which allows the bypass air to flow into the turbine exhaust gas flow in a manner that ensures complete mixing of the two streams. The mixer unit comprises corrugated sheath welded in front end to machined flange to be attached by bolts to rear flange of inner duct exhaust cone. The corrugated sheath creates the outer wall for inner duct gas flow and inner wall for outer duct air flow. Due to corrugated shape of the sheath several chutes are created.

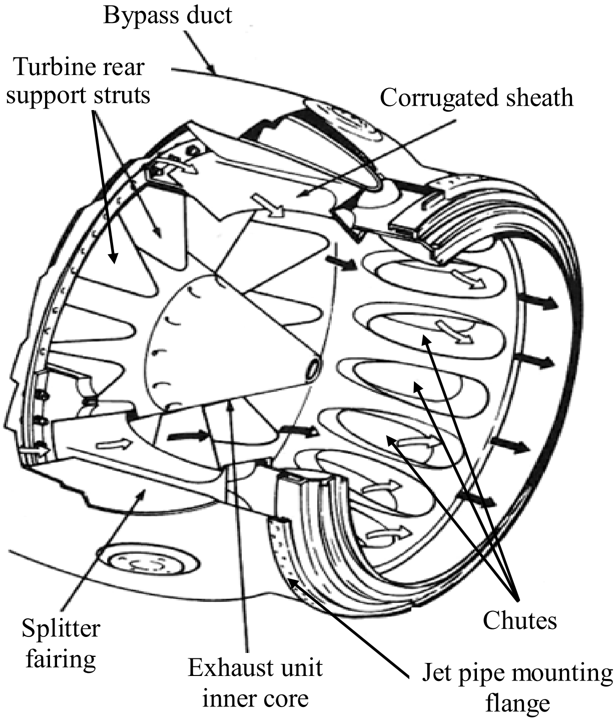


Fig. 11.4. Exhaust cone of low bypass turbofan engine:
□ — bypass air; ■ — exhaust gases

Along these chutes the hot gases of inner duct are directed into air flow of outer duct, and along neighboring chutes the cold air is directed into gas flow of inner duct. So two flows are mixed effectively. This process is accompanied by noise level reducing. Further path of mixed air-gas mixture is absolutely similar with the path of exhaust gas in pure turbojet engine, discussed above. If bypass engine is located on pylon under a wing the exhaust nozzle is fitted directly to the rear mounting flange of exhaust cone. If the engine is installed inside of fuselage, additional exhaust pipe between exhaust cone and exhaust nozzle is used.

In high-bypass-ratio engines (if bypass ratio is more than 4:1), the two streams are usually exhausted separately through two different exhaust nozzles as it is shown in Fig. 11.5. Cold air stream of the bypass fan duct flows out to atmosphere through outer exhaust nozzle, and hot gas from the core duct flows out through the inner exhaust nozzle. In this case only partial external mixing of the two gas streams takes place out of the engine in the atmosphere. Propulsive efficiency of engine is reduced because of kinetic energy losses when two streams with different velocities are interacting.

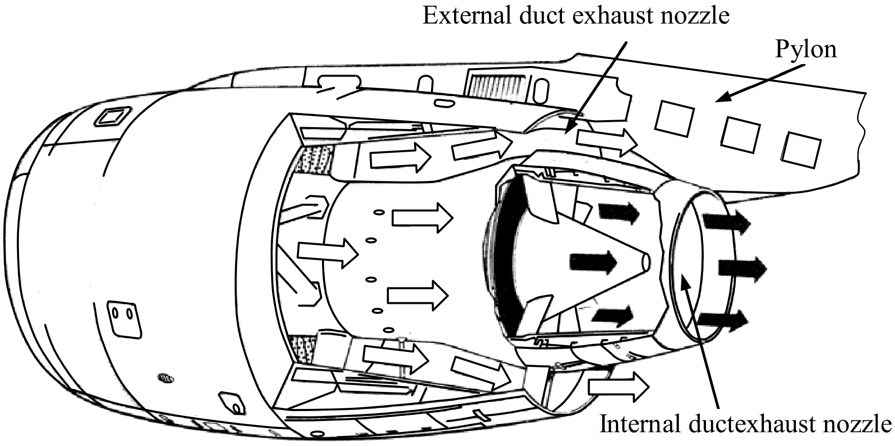


Fig. 11.5. Exhaust system of high bypass ratio turbofan engine
with separate exhaust nozzles of inner and outer ducts:
⇨ — cold bypass (FAN) airflow; ➡ — Hot exhaust gases

However, an improvement can be made by mixing the two gas flows before a common, or integrated, exhaust nozzle. This partially mixes the gas flows prior to ejection to the atmosphere. Examples of such type exhaust unit used for bypass (or turbofan) engine is shown in Fig. 11.6.

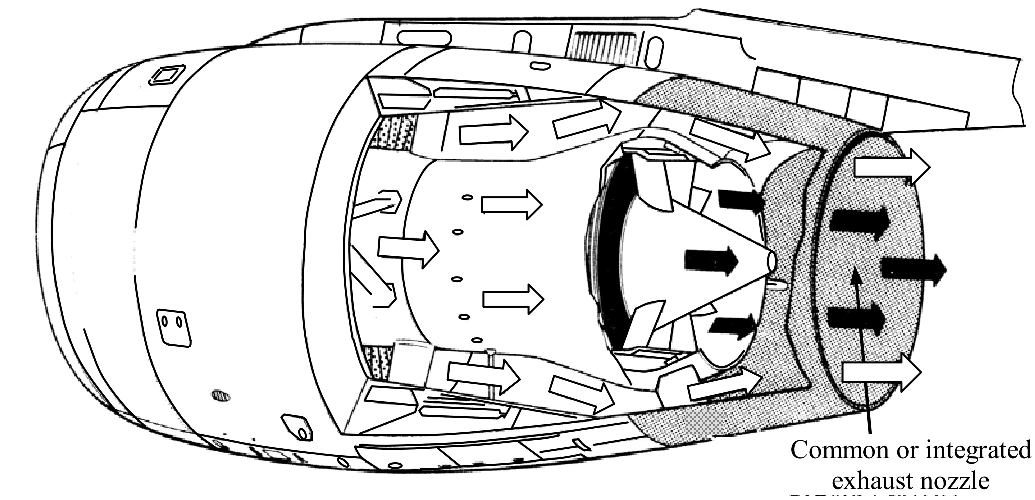


Fig. 11.6. Exhaust system of high bypass ratio turbofan engine with common exhaust nozzle of inner and outer ducts:
— cold bypass (FAN) airflow; — Hot exhaust gases

In this case more effective mixing air flow of outer duct and gas flow of inner duct is achieved. Due to this propulsive efficiency of engine is increased and specific fuel consumption is reduced. Besides, noise level of engine is reduced as well.

11.5. Thrust Reversers

Majority of large passenger and cargo aircraft powered by turbojet and bypass GTE include **thrust reversers** to assist the brake system in slowing the aircraft after landing. The two most commonly used types of thrust reversers are the mechanical blockage system, illustrated in Fig. 11.7, and the aerodynamic blockage system, illustrated in Fig. 11.8.

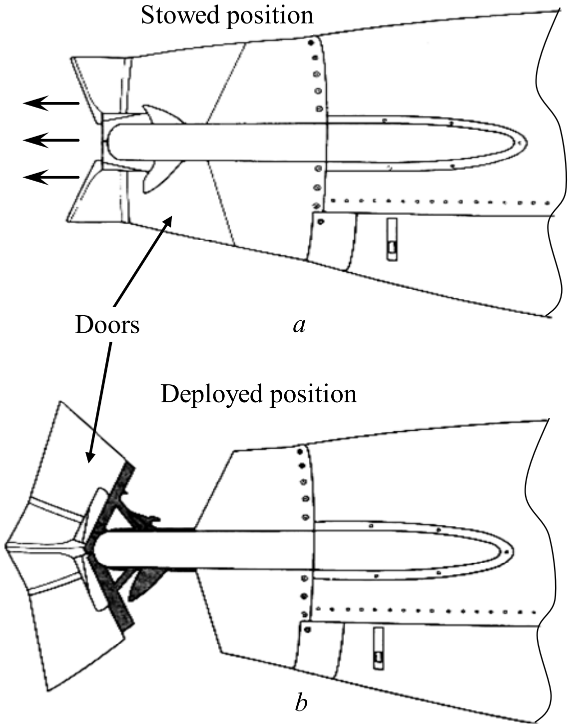


Fig. 11.7. Thrust reversers of mechanical blockage type:
a — normalthrust, *b* — revers thrust

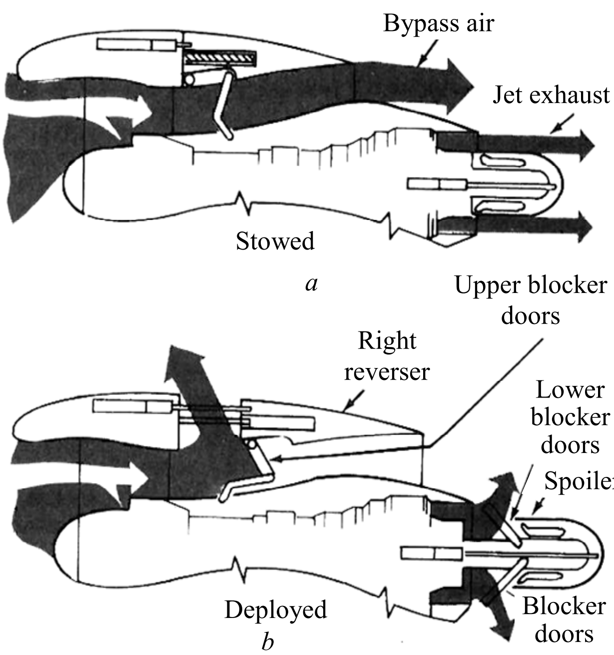


Fig. 11.8. Thrust reversers of aerodynamic blockage type:
a — normalthrust, *b* — revers thrust

Both the aerodynamic system and the mechanical system are subjected to high temperatures and to high gas loads. The components of both systems, especially the doors, are therefore constructed from heat-resistant materials and are of particularly heavy construction.

The mechanical blockage reverser shown in Fig. 11.7 consists of two blocker doors, or “clamshells”, which, when stowed, form the rear part of the engine nacelle. When the doors are deployed, as shown in the lower part of the illustration, they form a barrier to the exhaust gases and deflect them to produce a reverse thrust. This thrust reverser is hydraulically operated and electrically controlled. The reverser cannot be deployed unless the engine rpm is less than 65 percent. Some aircraft utilize a thrust reverser lever separated from the throttle; however, the two are connected to the same cable and clutch systems to prevent the thrust reverser from being deployed above the IDLE position. On other aircraft, the throttle and thrust lever is the same.

The thrust reverser shown in Fig. 11.8 is equipped with vanes and deflectors by which the exhaust gases and cold fan stream air are deflected outward and forward and are controlled through the thrust lever in the cockpit. This system consists of a number of cascade vanes and solenoids with a pneumatic motor operating through gears and shafts to move the cascade vanes to the deployed position.

After the aircraft has landed, the pilot moves the levers to the rear of the IDLE position. This causes the deflecting vanes to move into the main stream of the gas flow through the engine and the cold airstream to reverse the flow direction. At the same time as the thrust levers are moved further rearward, the fuel flow to the engine is increased. When the thrust levers are in the FULL REVERSE position, the engine power output is approximately 75 percent of full-forward thrust capability.

11.6. Turboprop and turboshaft engine exhaust system

Turboprop and turboshaft engine exhaust system is the simplest one in comparison with all descripted above because of more simple task it performs. Actually, the exhaust unit of such engines serves only to discharge the gas flow after turbine into atmosphere. Because of very considerable rate of gas expansion in several stages of turbine unit of such engines the pressure of gas after turbine is approximately equal to ambient pressure, that is why following acceleration of gas is impossible and is not desired. So, the shape of exhaust duct is not convergent or divergent.

The design of turboprop engine exhaust unit is illustrated in Fig. 11.9.

Here exhaust unit includes only two parts: exhaust cone attached to rear flange of turbine casing and exhaust pipe connected directly to rear flange of exhaust cone. The shape of exhaust pipe in this case is constant diameter cylindrical tube and because of absence of excessive gas pressure after turbine unit velocity of gas flow is not increased practically along the exhaust pipe.

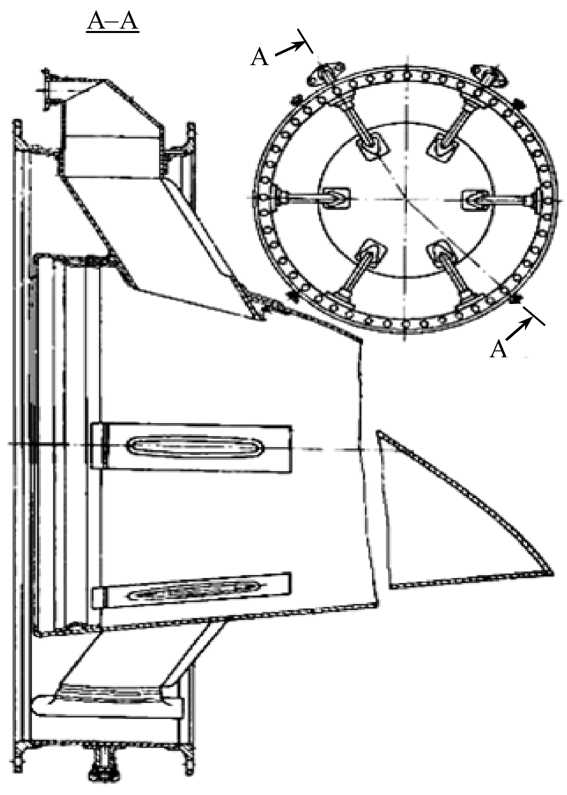


Fig. 11.9. Exhaust unit of turboprop engine

The length and shape of exhaust pipe depend on place of engine location on the airplane and should be sufficient to prevent the impact of the hot exhaust gases on the airframe components of aircraft. In those cases where the engine arrangement on the aircraft requires the remote location of the gas outlet from the engine installation site, then the exhaust channel is supplemented by an extension pipe through which the gas flow is discharged beyond the nacelle.

In the majority of turboshaft engines, the shaft of the free turbine is pulled back, so it crosses the engine exhaust channel. For this reason, the outlet channel after the turbine first has a diffuser part, which allows to lower the pressure behind the turbine. Inside the outlet channel there is a casing in which the output shaft of the free turbine is located with its supports and casing.

The exhaust units of turboshaft engines of helicopters are almost identical to the exhaust devices of turboprop engines.

However, when installing two turboshaft engines connected to a single helicopter gearbox that is installed in upper part of the helicopter body, the direction of exhaust gas exit from the exhaust pipe at the left and right engine provides the direction of outgoing gases to the outside so as to prevent the exhaust gases from entering the gearbox housing.

11.7. The TB3-117BMA-CBM1 engine exhaust unit

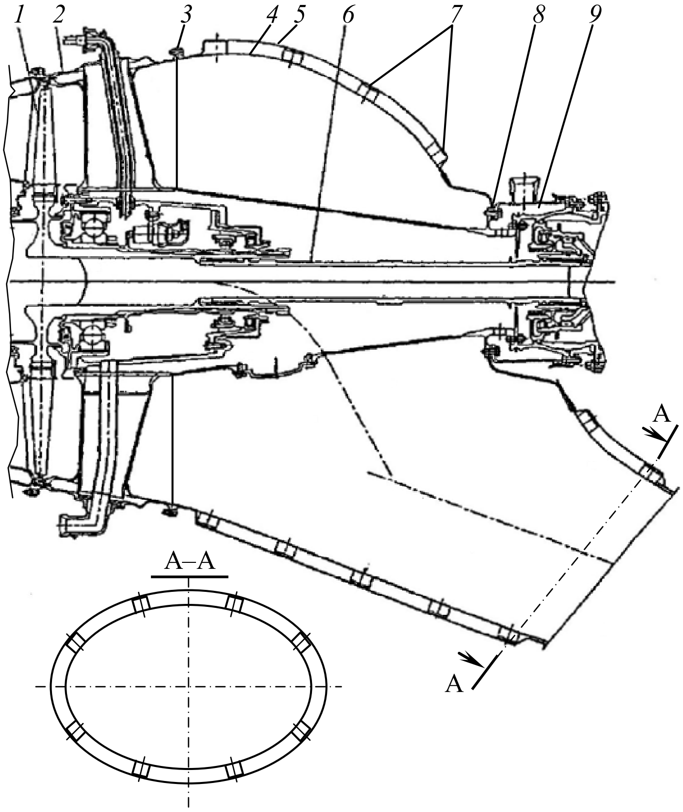


Fig. 11.10. Exhaust nit of TB3-117BMA-CBM1 engine:
1 — power turbine blade; *2* — power turbine bearing casing;
3 — exhaust pipe front flange; *4* — exhaust tube wall;
5 — heat-reflecting jacket; *6* — output shaft of engine;
7 — profiled supports; *8* — exhaust pipe rear flange; *9* — receiver

The TB3-117BMA-CBM1 engine was created on the basis of the turboshaft helicopter engine TB3-117BMA. Therefore, its exhaust device in its shape is largely similar to the exhaust devices of turboshaft engines. The exhaust channel shape of the TB3-117BMA-CBM1 outlet device provides exhaust gas outlet at an angle of 15–20 degrees downward relative to the engine axis. It allows to prevent the exhaust gases from getting on the structural elements of aircraft wing. The design of theTB3-117BMA-CBM1 engine exhaust unit is shown in Fig. 11.10.

Exhaust unit is attached by means of its front flange *3* to power turbine bearing casing *2*. Exhaust unit tube wall 4 forms the flow sections of the engine and discharges the gas flow from the turbine to the atmosphere. As it is visible in Fig. 11.10, the exhaust unit tube wall *4* creates the gas flow path of ring-shaped cross section near the flange *3* and oval-shaped cross section with enlarged horizon axis and with reduced vertical axis near outlet cross section A–A.

The exhaust unit tube is a welded structure

Made of a heat — resistant alloy. As it is shown in Fig. 11.10 the outer wall of the exhaust pipe is covered with a thin-walled heat-reflecting jacket made of a thin aluminum sheet, supported by profiled steps of a height of 5–6 mm. Due to this, a cavity is formed between the outer wall of the exhaust pipe and the casing, through which cooling air, taken from the atmosphere, flows. This prevents the transfer of heat to the surrounding parts of the nacelle. To prevent excessive heating of the shaft of the power turbine and the support elements of this shaft, the space inside the casing is also purged with air taken through the holes on the turbine casing *2* and discharged through the openings in the walls of the receiver *9* into the atmosphere to ensure cooling power turbine bearing casing *2* and output shaft *6* supports.

REVIEW QUESTIONS

1. Name the functions of the exhaust units of turbojet and turbofan gas turbine engines.
2. What parts does exhaust unit of subsonic turbojet engine consist of?
3. What parts does exhaust unit of turbojet engine for supersonic flight speed consist of?
4. How do the main parameters (pressure, temperature, speed) change in convergent and divergent parts of exhaust unit of supersonic turbojet engine?
5. What purpose are the mixing devices used for in the exhaust unit of turbofan engines with a low bypass ratio?
6. What is the purpose of the thrust reverser device used in the exhaust units of turbojet and turbofan engines?
7. How the TB3-117BMA-CБM1 engine power turbine shaft is protected against overheating by the exhaust gases?
8. What kinds of materials the wall of the TB3-117BMA-CБM1 engine exhaust pipe and the heat-shielding casing installed under it are made of?

Chapter 12. GAS TURBINE ENGINE FUNCTIONAL SYSTEMS

12.1. Concept

To operate successfully, any gas turbine engine must be equipped with several functional systems. The main systems are engine fuel supply system, fuel control system, engine lubrication system, engine starting system, power plant instrument and mechanic control system. Besides there are some additional systems such as: deicing system, fire protection system, engine diagnostic and monitoring system. Brief description of the TB3-117BMA-CBM1 engine main functional systems is given below.

12.2. Lubrication system description and operation

The lubrication system (or oil system) of any gas turbine engine is required to provide lubrication and cooling for all gears, bearings and splines. It must also be capable of collecting foreign matter which, if left in a bearing housing or gearbox, can cause rapid wear. Besides, oil must protect lubricated components which are manufactured from non-corrosion resistant materials. Moreover in a turboprop engine oil is also used as working body to operate the propeller pitch control mechanism. Oil must fulfill all these tasks without significant deterioration.

The TB3-117BMA-CEM1 engine oil system (Fig. 12.1) is recirculating lubrication system, with the oil circulating under pressure. It is structurally an independent system, i.e. all components fulfilling lubrication and breather functions are mounted directly on the engine.

This lubrication system provides uninterrupted delivery of oil under pressure to the surfaces subject to friction and seals of bearings of compressors and turbines, to the rotating parts of reduction gear and shaft line. Besides, this system gives oil to the internal drive and accessory drive gearbox for their lubrication and cooling, and performs delivery of oil into the propeller control and torque measurement systems.

The principal components of the lubrication system, as seen in Fig. 12.1, are: oil tank *8*; oil pump block *34*; de-aerator *51* with scavenge pumps; de-aerator *43*, which is mounted in oil pump block; oil filter *23*; air-oil heat exchanger *59*; fuel-oil heat exchanger *47*; drain valves *9*; drain plug *37*; drain valves *5*; centrifugal breather *36*, safety and protective filters *20* and *57* of scavenge pumps *1*, *41* and *50*.

The oil system (Fig. 12.1) is of normal-circuit type. Oil flows by-gravity from the oil tank *8* to two-section oil pressure pump *4* of oil pump block *39*. The pressure stages *4* of the main oil pump deliver oil to two filters *25* and *26* of the filter unit *23*. From one filter oil flows through drilled passages of the reduction gear case to lubricate the gears and is also directed to the oil pump *22* of the torquemeter. This oil is also employed as operating fluid for the propeller speed governor *29*.

From the other filter oil is routed via outer pipes to lubricate the compressor bearings *30* and *31*, compressor turbine bearing *32* and power turbine ball bearing *33* and roller bearing *34*.

From the front casing all oil is drained through the lower rib of the front casing to the entrance of three-gear oil pump scavenge stage *45*. From the compressor rotor rear bearing *31* and the compressor turbine rotor bearing *32* the oil flows through the protective filters into a scavenge gear pumps *40* and *44*, from whence it is routed through the chip detector *43* to the centrifugal de-aerator *43*, where air is separated from oil. After this oil is directed through a check valve to fuel-oil heat exchanger *47* and after partial cooling oil flows to the engine air inlet section *6* for heating this section for preventing ice formation on the its surface. From this section oil is scavenged by oil pump *50* to entrance of de-aerator *51*. To this point oil from the four-gear scavenge pump *2* and from sections of pump *1* for scavenging oil from front reduction gear comes too.

From the de-aerator *51* the centrifuged oil flows to the air-oil heat exchanger *55* with thermal valve *58*. Cooled oil then is directed to the oil tank *8*.

The oil pressure downstream of the main oil pump *4* is controlled by the pressure-reducing relief valve *3*.

The engine is drained of oil through the drain valves (cocks) *9* and *56*, arranged in the oil tank and in the air-oil heat exchanger *59*.

The oil cavity of the front reduction gear, shafting casing cavity and rear reduction gear cavity communicates with the centrifugal breather *36* mounted on the rear reduction gear casing. Oil mixed with the air is centrifuged here and returns into the rear casing cavity, while the air is routed into the jet nozzle. Air from the labyrinths of the compressor rear bearing and the turbine bearing is also directed into the jet nozzle.

This lubrication system is capable of delivering oil during a maximum of *8* s in the event of negative and near-zero g-loads. In addition, in the fuel-oil heat exchanger the fuel at the inlet of the fuel filter is heated by the oil scavenged from the engine hot bearing supports up to the temperature preventing ice formation on the filter.

For preventing ice formation on the surface of the engine air inlet section, hot oil is passed through the inner duct of the engine air inlet section.

For monitoring the current values of the engine inlet oil temperature and pressure, the lubrication system is provided with the indicator. For prompt detection of the defects in the lubrication and breather system parts, assemblies and units washed with oil, the system is provided with transmitters and warning units.

Jointly with the electronic engine control (EEC) these transmitters and warning units ensure operation of the following warning annunciators:

- MIN OIL PRESS (МАСЛО МИН) — at the minimum permissible oil pressure;
- LIMIT OIL TEMP (МАСЛО ПРЕДЕЛЬНОЕ) — at the limit oil temperature at the outlet from the hot bearing supports;
- LOW OIL (МАСЛО МАСЛО) — at the minimum permissible level of the oil in the oil tank;
- CHIPS (СТРУЖКА) — at the presence of ferromagnetic chips in the scavenged oil;
- OI\L FILTER (МАСЛОФИЛЬТР) — in the case of clogging of the oil filter.

For automation of the oil tank filling process and preventing dirt and dust from entering the oil being filled into the oil tank, the standard pressure-filling connection is installed on the engine.

12.3. Engine fuel, controlling and indicating systems description and operation

The functions of the fuel system of any gas turbine engine are to provide the engine with fuel in a form suitable for combustion and to control the flow to the required quantity necessary for easy starting, acceleration and stable running, at all engine operating conditions. To do this, one or more fuel pumps are used to deliver the fuel to the fuel spray nozzles, which inject it into combustion chamber in the form of an atomized spray. Because the flow rate must vary according to the amount of air of air passing through the engine to maintain a constant selected engine speed or pressure ratio, the controlling devices are fully automatic with the exception of engine power selection, which is achieved by a manual throttle or power lever. A fuel shut-off valve (cock) control lever is also used to stop the engine, although in some instances these two manual controls are combined for single lever operation.

The control of power or thrust of the gas turbine engine is effected by regulating the quantity of fuel injected into the combustion system. When a higher thrust is required, the throttle is opened and the pressure to the fuel spray nozzles increases due to the greater fuel flow.

This has the effect of increasing the gas temperature, which in turn increases the acceleration of the gases through the turbine to give a higher engine speed and a correspondingly greater airflow, consequently producing an increase in engine thrust.

With the turbo-propeller engine, changes in propeller speed and pitch have to be taken into account due to their effect on the power output of the engine. Thus, it is usual to interconnect the throttle lever and propeller controller unit, for by so doing the correct relationship between fuel flow and airflow is maintained at all engine speeds and the pilot is given single-lever control of the engine. Although the maximum speed of the engine is normally determined by the propeller speed controller, overspending is ultimately prevented by a governor in the fuel system.

The fuel system often provides for ancillary functions, such as oil cooling and the hydraulic control of various engine control systems: for example, compressor airflow control.

The engine ТВ3-117ВМА-СЕМ1 fuel, controlling and indicating system consists of the following systems: fuel feed system, automatic control system, and indicating and fault diagnosis system.

The fuel feed system (Fig. 12.2) is intended to supply fuel into the combustion chamber in accordance with the preset laws, ensuring reliable operation of the engine under any operating conditions.

The fuel feed system comprises: engine-driven centrifugal pump ДЦН-104 (hereinafter in the text called centrifugal pump), fuel control unit НР-2000 (hereinafter in the text called FCU), fuel distributor РТ-2000 (hereinafter in the text called fuel distributor), fuel-oil heat exchanger ТМА 7068 (hereinafter in the text called fuel-oil heat exchanger), fuel manifold, main fuel nozzles and pipelines.

The automatic control system (ACS) is designed to fulfill the following functions: engine automatic control, propeller automatic control, and indicating and fault diagnosis.

The block diagram showing interaction of the engine fuel automatic controlling, indicating and fault diagnosis system with the aircraft and engine systems as well as interconnection between the accessories of the automatic controlling, indicating and fault diagnosis system is shown in Fig. 12.3.

The functions of the engine automatic control system (hereinafter in the text abbreviated to EACS) are as follows: automatic fuel management, compressor control.

The automatic fuel management system consists of: main ACS and alternate ACS. The main ACS is an electronic hydromechanical system comprising the following units: electronic engine control (РЭД-2000) (hereinafter in the text abbreviated to EEC) — electronic control portion, FCU — hydromechanical actuating portion, fuel distributor — hydromechanical actuating portion.

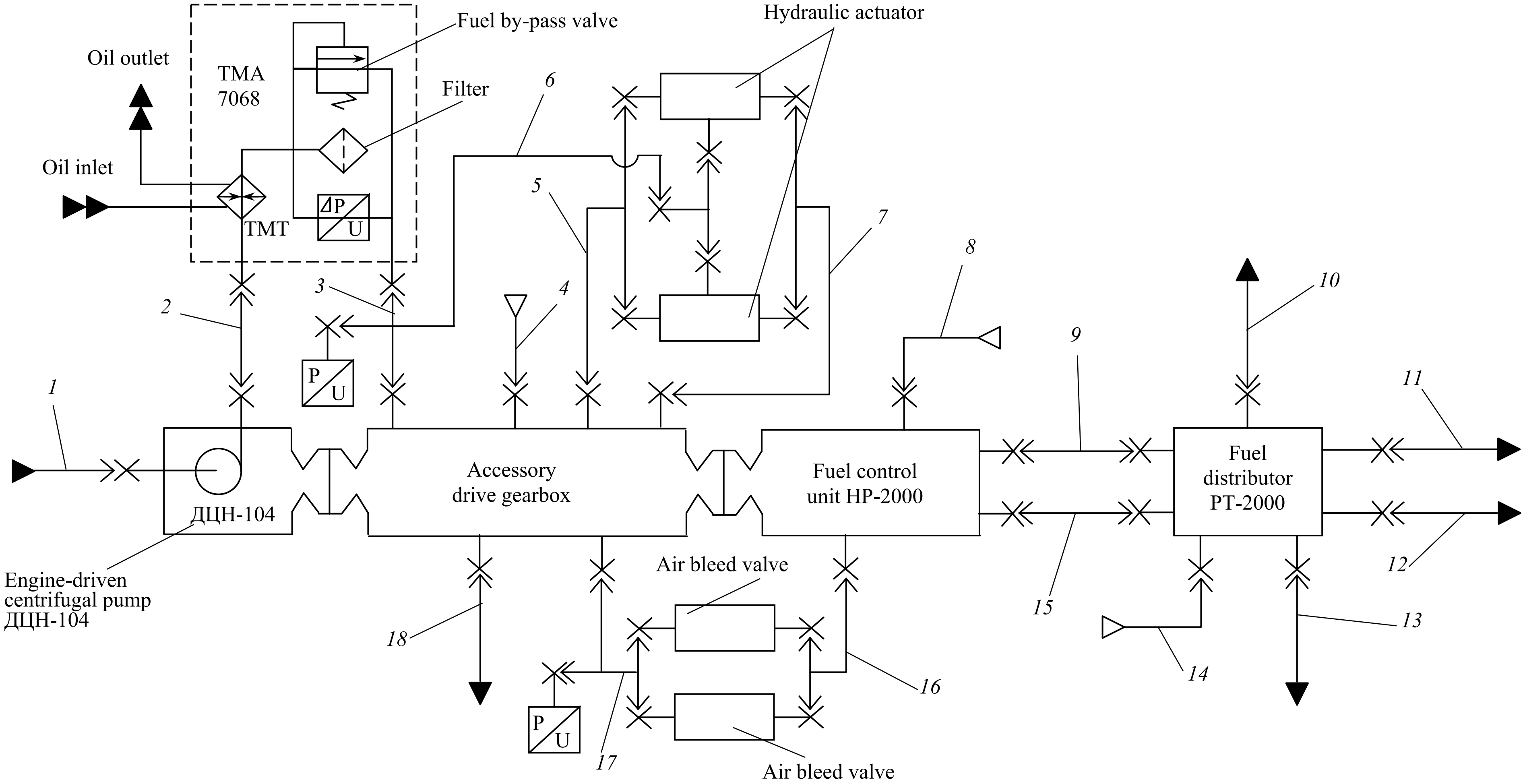


Fig. 12.2. Diagram of the TB3-117 TMA-CEM1 engine fuel system :
1 — fuel inlet to engine-driven centrifugal pump; *2* — fuel supply from engine-driven centrifugal pump to fuel-oil heat exchanger with filter; *3* — fuel supply from fuel-oil heat exchanger with filter to FCU; *4* — delivery of engine inlet air total pressure to FCU; *5* — fuel delivery from FCU to hydraulic actuator; *6* — fuel return from hydraulic actuator to FCU inlet; *7* — fuel delivery from FCU to hydraulic actuator; *8* — supply of HP compressor discharge total pressure to FCU; *9* — fuel supply from FCU to fuel distributor; *10* — fuel drain from combustion chamber; *11* — fuel supply from fuel distributor to primary fuel nozzle manifold; *12* — fuel supply from fuel distributor to main fuel nozzle manifold; *13* — fuel drain from main fuel nozzles and combustion chamber; *14* — air supply from pneumatic turbine starter to fuel distributor; *15* — fuel supply from FCU to fuel distributor; *16* — fuel return from air bleed valve into FCU; *17* — fuel supply from FCU; *18* — fuel drain from sealing cavities of engine-driven centrifugal pump and FCU;
— air flow direction; — fuel flow direction; — oil flow direction

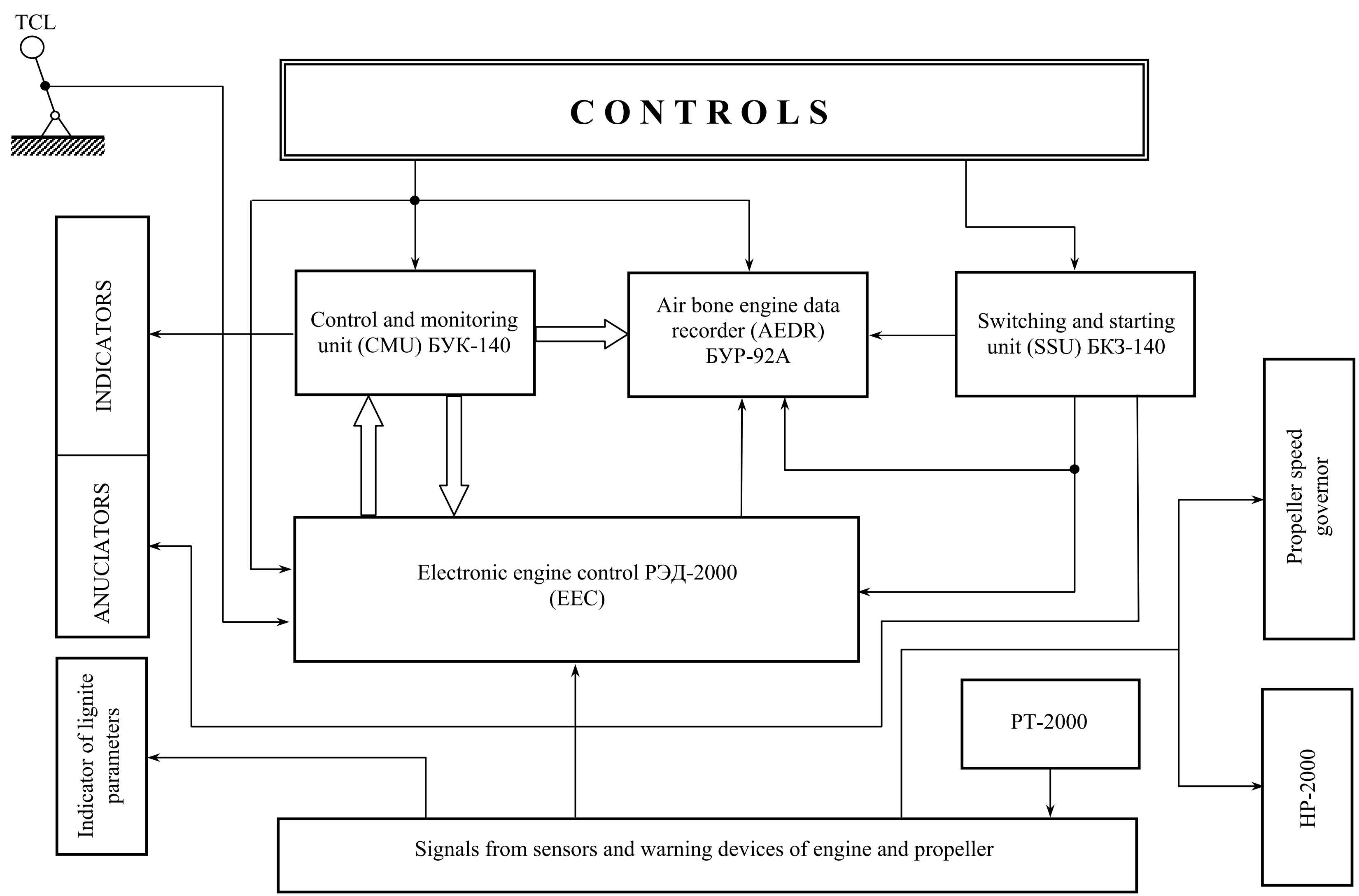


Fig. 12.3. Block diagram of the ТВ3-117 ТМА-СВМ1 engine automatic control system

The alternate ACS is a hydromechanical system comprising: FCU and fuel distributor. During operation of the main ACS the EEC receives information on the following: engine operating parameters, environmental conditions (ambient air pressure and temperature), position of the engine controls located in the aircraft, flight speed.

Fuel feed system operates in the following way (Fig. 12.2): fuel from the aircraft fuel system flows to the inlet *1* of the engine-driven centrifugal pump intended to increase fuel pressure. From the pump fuel flows by the pass *2* to the fuel-oil heat exchanger (TMA 7068) with filter to be heated there due to heat transfer from oil discharged from the engine oil system and filtered. Pressure difference across the fuel filter rises, when the filter gets clogged. If that of is the case, annunciator FUEL FILTER (ТОПЛ. ФИЛЬТР) gets alight in response to operation the differential pressure switch Д1. If the pressure difference across the fuel filter still further increases, the fuel by-pass valve of the fuel filter opens to by-pass excess fuel in a round about way. When the fuel by-pass valve is open, limit switch installed on the fuel-oil heat exchanger with filter operates and puts out a signal on the open state of the by-pass valve to the engine electronic control (EEC). The EEC transmits this signal to the airborne engine data recorder (AEDR) and to the MESSAGE (ЕСТЬ СООБЩЕНИЕ) enunciator.

From the fuel-oil heat exchanger fuel flows through an adapter on the engine accessories drive gearbox to get into the FCU (HP-2000). Installed in the FCU delivery pipe is a pressure switch operating in the event of minimum allowable fuel pressure at the inlet to the FCU. The fuel control unit is designed to meter fuel flow at any operating rating of the engine. From the FCU the metered fuel flow is supplied through the pipes *9* and *15* to the fuel distributor PT-2000. The fuel distributor divides fuel flow accordingly and supplies divided fuel flows into the primary *11* and main *12* fuel nozzle manifolds at any operating rating of the engine.

Fuel flow automatic control system operates in the following way (Fig. 12.2): prior to engine start, the fuel distributor connects the fuel nozzles primary and main fuel manifolds with the engine drain system. After the ENG START (ЗАПУСК ДВИГ) button is depressed the EEC generates and puts out control signals to the signal transducer, in response to which the FCU fuel flow valve assumes a position corresponding to fuel flow rate within (35 ± 5) kg/h. To keep the fuel flow rate control valve in that position, the EEC generates and puts out respective control signals to signal transducer until the combustion chamber gives symptoms of torching.

As a consequence, the FCU starts and maintains fuel supply into the nozzles primary fuel manifold in the following manner: at a fuel flow rate in accordance with the combustion chamber fuel feed pump delivery rate, if the pump capacity is below (35 ± 5) kg/h or at a fuel flow rate within (35 ± 5) kg/h, if the gas generator rotor rotational speed is not less than 13 %. The fuel distributor disconnects the nozzles primary and main fuel manifolds from the drain system not later than fuel starts flowing into the primary fuel manifold. The fuel distributor provides blowing of air into the nozzles main fuel manifold from the pneumatic turbine starter. The fuel distributor ensures discontinuation of air blow into the nozzles main fuel manifold as soon as fuel pressure in the primary fuel manifold rises to 6 kgf/cm^2. After an indication of torching in the combustion chamber emerges, the EEC generates and puts out control signals to FCU — G_f, thereby ensuring acceleration of the gas generator rotor in accordance with the starting law.

In sustaining gas generator rotor acceleration the EEC generates and puts out control signals to FCU — G_f signal transducer, thus restricting fuel flow rate in accordance with the starting law, but keeping it not less than fuel flow rate calculated by the EEC in accordance with the formula G_f = 20 + 30,3 Pin and not over 660 kg/h. The distributor maintains fuel flow rate into the nozzles main fuel manifold with the fuel flow rate in the primary fuel manifold kept within (85 ± 5) kg/h.

When the engine starting is completed, EEC ensure fuel flow control according to steady-state engine operating conditions lows. Engine power ratings during operation of the main or alternate automatic control system are selected by respectively setting the throttle control lever (hereinafter in the text abbreviated to TCL), installed in the flight compartment.

The TCL is mechanically linked up with the TCL position resolver and with fuel flow rate control lever installed on the FCU. When the TCL is moved, the mechanical control system moves the fuel flow rate control lever accordingly to the respective position. Information on the TCL position is conveyed to the control and monitoring unit (CMU) and via the data exchange channel (DEC) to the EEC. When the main automatic control system is in action, the EEC, in response to the received information about position of the TCL, generates and puts out control signals, thereby ensuring the engine operating conditions as a function of the TCL setting.

In case of failure of the main automatic control system catering for fuel flow control the directly ensures required fuel supply into the combustion chamber, depending on the position of the fuel flow control lever.

REFERENCES

1. *Авіаційні* двигуни : навч. посібник / А. М. Вєтров, І. І. Гвоздецький, Є. М. Карпов та ін. — К. : КМУЦА, 1999. — 120 с.

2. *Алемасов В. Е.* Теория ракетных двигателей: учеб. пособие / В. Е. Алемасов, А. Ф. Дрегалин, А. П. Тишин. — М. : Машиностроение, 1969. — 540 с.

3. *Региональный* пассажирский самолёт Ан-140: краткое техническое описание. — К. : АНТК им. О. К. Антонова, 1996. — 102 с.

4. *Турбовинтовой* двигатель ТВЗ-117ВМА-СБ2: руководство по технической эксплуатации. — Запорожье : НПО «Мотор-Сич», 1997. — 102 с.

5. *Otis C. E.* Aircraft gas turbine powerplants / C. E. Otis, A. Peter, 3-d ed. — Vosbury — USA : Jeppesen. — 2002. — 511 p.

6. *Crane D.* Powerplant section textbook / D. Crane, C. Otis, F. Delp. IAP Inc., 1983. — 563 p.

7. *The jet engines*. Derby, England : Rolls-Royce pblc., 1986. — 277 p.

8. *Kroes M. J.* Aircraft Powerplants / M. J. Kroes, T. W.Wild. — 7-th ed. : McGRAW-HILL International Editions, 1994. — 694 p.

9. *Treager I. E.* Aircraft Gas Turbine Engine Technology / I. E. Treager. 3-rd edit.: Tata McGraw-Hill, New Dehly, 1997. — 677 p.

10. *Turner M. J. L.*, Rocket and Spacecraft Propulsion: Principles, Practice and New Developments/ M. J. L.Turner. — 3-rd ed. — Pracxis publishing Ltd. Chichester, UK, 2008. — 329 p.

11. Dieter K. Huzel. Modern Engineering for Design of Liquid-Propellant Rocket Engines / K. Huzel Dieter, David H. Huang. — Volume 147. — American Institute of Aeronautics and Astronautics,1992. — 433 p.

12. *Gvozdetskyi I.* Design of Aviation Machines. Modern Turboprop Engines: Manual / I. Gvozdetskyi, L. Vollianskay, L. Morozov, M. Karpenko. — К. : NAU-druk publishing Hous, 2010. — 96 p.

13. *TV3-117VMA-SBM1 Engine Assembly/Training Manual*: — Zaporozche, Motor-Sich JSC, 2001–329 p.

14. https://www.slideshare.net/jschrell/rd-120-paper.

15. http://www.buran-energia.com/energia/moteur-fusee-rocket-engine-desc.php.

16. http://www.braeunig.us/space/propel.htm.

This glossary is provided as a ready reference of terms as used in this text these definitions may differ from those of standard dictionaries, but are more common in reference to the gas turbine engine.

Acceleration due to gravity. The acceleration of a freely falling body due to the attraction of gravity, expressed as the rate of increase of velocity per unit of time. In a vacuum the rate is 32.2 feet per second per second near sea level (9.81 m/s^2).

Absolute. The magnitude of a pressure or temperature above a perfect vacuum or absolute zero, respectively. For temperature, absolute zero is theoretically equal to –273.18 °C.

Absolute pressure. Pressure above zero pressure as read on a barometer type instrument, e.g. Standard Day, 14.7 psi-A. (760 mm Hg.)

Additive. A material added to an oil or a fuel to change its characteristics or quality.

Afterburner. A tubular combustion chamber with a variable-size exhaust outlet attached to the rear of a gas turbine engine into which fuel is injected through a set of spray bars. An example is the Concorde SST aircraft. Burning of fuel in the exhaust supplements the normal thrust of the engine by increasing the acceleration of the air mass through an additional temperature rise.

Airfoil. Any surface designed to obtain a useful reaction upon its surfaces from the air through which it moves. Velocity increases over the cambered side producing lift on the underside.

Air, ambient. The atmospheric air surrounding all sides of the aircraft or engine. Expressed in units of Ibs./sq. inch or in. Hg.

Air (standard). Sea-level atmospheric conditions of temperature at 15 °C (59.0 °F) and air pressure at $1.013 \cdot 10^5$ Pa (14.7 psi).

Air start. The process of starting an aircraft engine in flight.

Alloy. A solid solution of two or more metallic constituents. One metal is usually predominant, and to it are added smaller amounts of other metals to improve strength and heat resistance.

Alternator. A mechanical device which produces AC current through induction.

Annular combustor. A cylindrical (a ring-shaped) one piece combustion chamber, sometimes referred to as a single basket type combustor.

Atmosphere (standard) *See* **air (standard).**

Atomize. To break a liquid up into minute particles.

Atomizer. A device through which fuel is forced so that it enters the combustor as a fine spray.

Autorotation. A rotorcraft flight condition in which the lifting rotor is driven entirely by action of the air as the rotorcraft is in motion.

Auxiliary power unit. A type of gas turbine, usually located in the aircraft fuselage or in other part of airframe, whose purpose is to provide either electrical power, air pressure for starting main engines, or both. Similar in design to ground power units.

Axial. Motion along a real or imaginary straight line on which an object supposedly or actually rotates. The engine centerline.

Axial flow compressor. Compressor with airflow parallel to the axis of the engine. The numerous compressor stages raise pressure of air but essentially make no change in direction of airflow.

Axial-flow turbine. A turbine in which the energy of flowing air is converted to shaft power while the air follows a path parallel to the turbine's axis of rotation.

Bearing (mechanical). Part of a machine that supports a journal, pivot, or pin that rotates, oscillates, or slides.

Bernoulli's theorem. Principle which states pressure and velocity of a gas or fluid passing through a duct (at constant subsonic flow rate) are inversely proportional.

Blade. A rotating airfoil utilized in a compressor as a means of compressing air or in a turbine for extracting energy from the flowing gases.

Blade angle. The angle between the blade section chord line and the plane of rotation of the propeller or crankshaft.

Blade chord. A straight line through the center of a propeller blade, perpendicular to its span, between its leading edge and its trailing edge.

Brayton cycle. A thermodynamic cycle of operation that may be used to explain the operating principles of the gas turbine engine. It is sometimes referred to as the continuous combustion, or constant pressure cycle.

British Thermal Unit (BTU). A unit of heat. One BTU equals the heat energy required to raise one pound of water one degree Fahrenheit (e.g. one pound of jet fuel contains approximately 18.600 BTU).

Bucket. Accepted jargon for turbine blade.

Can-annular combustor. A set, generally of 6 to 10 liners within one outer annulus (combustor outer case).

Centrifugal flow compressor. An impeller shaped device which receives air at its center and slings air outward at high velocity into a diffuser to increase pressure. Sometimes referred to as a radial outflow compressor.

Centrifugal force. The outward force an object exerts on a restraining agent when the motion of the object is circular.

Choked airflow. An airflow condition from a convergent shaped nozzle, where the gas is traveling at the speed of sound and cannot be further accelerated. Any increase in internal pressure will pass out the nozzle in the form of pressure.

Chord line. The imaginary line which extends from the leading edge to the trailing edge of an airfoil section.

Combustor. The section of the engine into which fuel is injected and burned to create expansion of the gases.

Compressor. An impeller or a multi-bladed rotor assembly. A component which is driven by a turbine rotor for the purpose of compressing incoming air.

Compressor pressure ratio. The result of compressor discharge pressure divided by compressor inlet pressure, e.g. a large turbofan may have a compressor pressure ratio of 25:1.

Compressor stage. A rotor blade set followed by a stator vane set. Simply stated the rotating airfoils create air velocity which then changes to pressure in the numerous diverging ducts formed by the stator vanes.

Compressor stall. A condition in an axial-flow compressor in which one or more stages of rotor blades fail to pass air smoothly to the succeeding stages. A stall condition is caused by a pressure ratio that is incompatible with the engine rpm. Compressor stall will be indicated by a rise in exhaust temperature or rpm fluctuation, and if allowed to continue, may result in flameout and physical damage to the engine.

Compressor surge. An operating region of violent pulsating airflow usually outside the operating limits of the engine.

Controllable-pitch propeller. A propeller whose pitch can be changed in flight.

Convergent duct. A cone-shaped passage or channel in which a gas may be made to flow from its largest area to its smallest area, resulting in an increase in velocity and a decrease in pressure. Referred to as nozzle shaped. An in verse proportion is present if the weight of airflow remains constant.

Crankshaft. The main shaft of the piston engine, which, in conjunction with the connecting rods, changes the linear reciprocating motion of the piston into rotary motion.

Cycle. A series of occurrences for which conditions at the end are the same as they were at the beginning.

Detonation. After normal ignition, the explosion of the remaining air-fuel mixture due to above-normal combustion chamber pressure or temperature.

Duct. A passage or tube used for directing gases.

Diffuser. The divergent section of the engine which is used to convert the velocity energy in the compressor discharge air to pressure energy. Aircraft inlet ducts and compressor stator vanes are also described as diffusers due to their effect on air in raising pressure.

Divergent duct. A cone-shaped passage or channel in which a gas may be made to flow from its smallest to its largest area resulting in an increase in pressure and a decrease in velocity. An inverse proportion is present if the weight of airflow remains constant, e.g. the engine diffuser.

Efficiency. The ratio of power output to power input. (Power input equals power output plus power wasted.)

Energy. Inherent power or the capacity for performing work. When a portion of matter is stationary, it often has energy due to its position in relation to other portions of mailer. This is called potential energy. If the matter is moving, it is said to have kinetic energy, or energy due to motion.

Engine mount. The part of an aircraft structure that is designed for attachment of the engine.

Engine cycle. Cycles are recorded as one takeoff and landing, and are used to compute time between overhaul of engines and components where operating hours are not used.

Engine pressure ratio (EPR). The ratio of turbine discharge pressure divided by compressor inlet pressure. Displayed in the cockpit as an indication of engine thrust.

Engine stations. Numbered locations along the engine length, or along the gas path used for the purpose of identifying pressure and temperature points, component locations and the like.

Exhaust gas temperature (EGT). Temperature taken at the turbine exit. Often referred to as T_T.

Exhaust nozzle. Also referred to as (he jet nozzle, this is the rear-most pan of the engine.

False start. An unsuccessful or aborted engine start.

Flame out. An unintentional extinction of combustion due to a blowout (too much fuel) or die-out (too little fuel).

Fan jet engine. A gas-turbine engine that employs a fan to accelerate a large volume of air through a bypass duct to increase thrust and engine efficiency.

Feathering propeller. A propeller that can have its blades turned so that they are facing directly into the airstream as the aircraft is moving through the air.

Fixed-pitch propeller. A propeller whose blade angles cannot be changed except by bending the blades to a new pitch.

Flameout. An unintended extinction of flame.

Fluid. Any substance having elementary particles that move easily with respect to each other—i.e., liquids (incompressible fluids) and gases (compressible fluids).

Foreign object damage (FOD). Compressor damage from ingestion of foreign objects into the engine inlet.

Free turbine. A turbine which operates independent shafts for high- and low-pressure rotors.

Fuel control unit. The main fuel scheduling device which receives a mechanical input signal from the power lever and various other signals, such as PI (p_B*), TI (T_B*), etc. These signals provide for automatic scheduling of fuel at all ambient conditions of ground and flight operation.

Fuel flow. Rate at which fuel is consumed by the engine in pounds per hour (p/h) or kilograms per hour (kg/h).

Fuel-air ratio. The ratio by weight of fuel to air.

Fuel nozzle. The nozzle in a gas-turbine engine combustor through which the fuel is discharged. The spray pattern from the fuel nozzle is such that the flame is always centered in the burner so that it will not overheat it.

Gas generator turbine. High pressure turbine wheel(s) which drive the compressor of a turboshaft or turboprop engine.

Gas turbine. Engine consisting of a compressor, combustor and turbine, using a gaseous fluid as a working medium and producing either shaft horsepower, jet thrust, or both.

Gear ratio. A gear relationship, usually expressed numerically, used to compare input to output speed.

Governor. The speed-sensing propeller control device that adjusts and maintains system rpm by adjusting oil flow to and from certain types of constant-speed propellers.

Ground idle. The engine speed that is normally used for operating a gas-turbine engine on the ground so that it will produce the minimum amount of thrust.

Ground power unit. A type of small gas turbine whose purpose is to provide either electrical power, air pressure for starting aircraft engines, or both. A ground unit is connected to the aircraft when needed. Similar to an aircraft installed auxiliary power unit.

Guide vanes. Stationary airfoil sections which direct the flow of air or gases from one major part of the engine to another.

Horsepower. Unit of power equal to 33.000 foot pounds of work per minute, 550 foot pounds per second, or 375 mile pounds per hour.

Hot start. A start which occurs with normal engine rotation, but exhaust temperature exceeds prescribed limits. This is usually caused by an excessively rich mixture in the combustor. The fuel to the engine must be terminated immediately to prevent engine damage.

Hung start. A condition of normal light off but with rpm remaining at some low value rather than increasing to the normal idle rpm. This is often the result of insufficient power to the engine from the starter. In the event of a hung start, the engine should be shut down.

Idle. A percent rpm setting, the value of which changes from engine to engine. It is the lowest engine operating speed authorized.

Igniter plug. An electrical sparking device used to start the burning of the fuel-air mixture in a combustor.

Ignition system. The means of igniting the air-fuel mixture in the cylinders; it
includes spark plugs, high-tension leads, ignition switches, and magnetos.

Impeller . Name given to the centrifugal flow compressor rotor.

Inertia. The opposition of a body to a change in its state of rest or motion.

Inlet duct. The ambient air entrance duct which directs air into the engine.

Inlet guide vane. Stationary airfoil which proceeds the first stage compressor rotor blades. These guide vanes form straight through passages and are present to direct air onto the blades at the optimum angle.

Jet. A small, tubelike device through which a fluid or gas flows.

Jet engine. A reaction engine which derives its thrust from the acceleration of an air mass through an orifice. There are four common types: rocket, ramjet, pulsejet, and turbojet.

Journal. A shaft machined to fit a bearing.

Kinetic energy. Energy due to motion.

Labyrinth seal. A high-speed seal which provides interlocking passages to discourage the flow of air, oil, etc., from one area to another.

Mach number. The ratio of the speed of the airplane to (he speed of sound (at the temperature in which the airplane is operating).

Mass. A basic property of matter. Mass is referred to as weight when in the field of gravity such as that of the earth.

Momentum. The tendency of a body to continue in motion after being placed in motion.

Nozzle, fuel nozzle. A spray device which directs atomized fuel into a combustion chamber.

Nozzle, turbine. A convergent duct through which hot gases are directed to the turbine blades.

Overspeed. Engine speed which exceeds the selected rpm by a set percentage.

Overtemperature. Any time exhaust gas temperature exceeds the maximum allowable limits.

Plane of rotation. The plane in which a propeller blade rotates.

Port. A hole through which gases may enter or exit.

Potential energy. Energy due to position.

Power lever. The cockpit lever which connects to the fuel control unit for scheduling fuel flow to the combustor. Also called power control lever or throttle.

Power turbine. A turbine rotor connected to an output reduction gearbox. Also referred to as free power turbine.

Pressure ratio. In a gas-turbine engine, the ratio of compressor discharge pressure to compressor inlet pressure.

Pressure, static. The pressure measured in a duct containing air, a gas or a liquid in which no velocity (ram) pressure is allowed to enter the measuring device. Symbol (Ps).

Pressure, total. Static pressure plus ram pressure. Total pressure can be measured by use of a specially shaped probe which stops a small portion of the gas or liquid flowing in a duct thereby changing velocity (ram) energy to pressure energy. Symbol (P_t or p*).

Primary air. The portion of the compressor output air that is used for the actual combustion of fuel, usually 20 to 25 percent.

Probe. A sensing device that extends into the air-stream or gas-stream for measuring pressure, velocity or temperature. In the case of pressure, it is used to measure total pressure. For temperature it measures total temperature.

Propeller. A device used for converting brake horsepower, or torque, into thrust to propel an aircraft forward.

Propeller feathering. Rotation of the propeller blades to eliminate the drag of a windmilling propeller on a multi-engine aircraft in the event of engine failure.

Propulsive efficiency. External efficiency of an engine expressed as a percentage.

Propeller governing. A mode of engine operation wherein the propeller governor selects the blade pitch to control engine rpm and the fuel flow is established manually.

Propeller reversing. Causing the blades of a propeller used on a turboprop engine to rotate at a negative angle to produce a reversing thrust that brakes the aircraft.

Propeller track. The plane of rotation of a propeller blade.

Ram pressure rise. Pressure rise in the inlet due to the forward speed of the aircraft, e.g. at M = 0.85 a pressure of 1.6 times above ambient will typically occur.

Rated horsepower. The horsepower that the manufacturer of an aircraft engine guarantees that the engine will produce under a given set of conditions.

Rotor. Either compressor or turbine. A rotating disk or drum to which a series of blades are attached.

S.A.E. Society of Automotive Engineers.

Secondary air. The portion of compressor output air that is used for cooling combustion gases and engine parts.

Shaft horsepower (shp). The horsepower that is actually available at a rotating shaft.

Shock stall. Turbulent airflow on an airfoil which occurs when the speed of sound is reached. The shock wave distorts the aerodynamic airflow, causing a stall and loss of lift.

Shroud. A cover or housing used to aid in confining an air or gas flow to a desired path.

Spark plug. A device inserted into the combustion chamber of an engine to deliver the spark needed for combustion.

Speed of sound. The terminal velocity of sound waves in air at a specific air temperature. Referred to as Mach-one. Symbol (M).

Splined shaft. A cylindrical crankshaft which has splines on its outer surface to prevent propeller rotation on the shaft.

Symbols

A acceleration
A_j area of jet nozzle
C_p specific heal
C_c local speed of sound
F_g static or gross thrust
F_n net thrust
G gravity
K_e kinetic energy
m mass airflow
M mach number
N_c speed, single compressor

N_1 speed, low pressure compressor
N_2 speed, high pressure compressor
N_f speed, free turbine
N_g speed, gas producer turbine
P_{am} pressure, ambient
P_b pressure, burner
P_j pressure at jet nozzle
P_s pressure, static
P_t pressure, total
T_t temperature, total
V_f velocity of aircraft
V_z velocity, jet nozzle
W_f weight of fuel, fuel flow rate
γ gamma, ratio of specific heats (C_P/C_V)
η eta, efficiency
ρ rho, density.

Thermal efficiency. Internal engine efficiency or fuel energy available versus work produced, expressed as a percentage.

Thermocouple. A pair of jointed wires of two dissimilar metals. A dc voltage is produced at one joint when the other joint is at a higher temperature.

Thrust. A pushing force exerted by one mass against another, which tends to produce motion in the masses. In jet propulsion, thrust is the forward force in the direction of motion caused by the pressure forces acting on the inner surfaces of the engine. Or, in other words, it is the reaction to the exhaust gases exiting the nozzle. Thrust force is generally measured in Newton's (N), pounds (p) or kilograms (kg).

Thrust, gross. The force which the engine exerts against its mounts while it is operating but not moving. Also called static thrust. Symbol (Fg).

Thrust, net. The effective thrust developed by the engine during flight, taking into consideration the initial momentum of the air mass prior to entering the influence of the engine. Symbol (F_n).

Thrust specific fuel consumption (TSFO. An equation: TSFC = W_f / F_n where: W_f is fuel flow in pounds per hour, and F_n is net thrust in pounds; used to calculate fuel consumed and as a means of comparison between engines.

Time between overhaul (TBO) The time in hours or engine cycles the manufacturer recommends as the engine, or engine component, service life from new to overhaul, or from one overhaul to the next.

Torque. A force multiplied by its lever arm, acting at right angles to an axis.

Torquemeter indicator. A turboprop or turboshaft cockpit instrument used to indicate engine power output. The propeller or rotor inputs a twisting force to an electronic or oil operated torquemeter which sends a signal to the indicator.

Turbine. A rotating device turned by either direct or reactive forces (or a combination of the two) and used to transform some of the kinetic energy of the exhaust gases into shaft horsepower to drive the compressor(s) and accessories.

Turbine blade. A fin mounted on the turbine disk and so shaped and positioned as to extract energy from the exhaust gases to rotate the disk.

Turbine stage. A stage consists of a turbine stator vane set followed by a turbine rotor blade set.

Turbine wheel. A rotating device actuated by either reaction, impulse or a combination of both, and used to transform some of the kinetic energy of the exhaust gases into shaft horsepower to drive the compressor(s) and accessories.

Turbojet. A gas-turbine engine whose entire propulsive output is delivered by the jet of gases through the turbine nozzle.

Turboprop. A type of gas-turbine engine that converts heat energy into propeller shaft work and some jet thrust.

Turbulence. An agitation of, or disturbance in, the normal flow pattern.

Vaporize. To change a liquid to a gaseous form.

Vector. A line which, by scaled length, indicates magnitude, and whose arrow point represents direction of action.

Velocity. The actual change of distance with respect to time. The average velocity is equal to total distance divided by total time. Usually expressed in miles per hour (m/h) or kilometers per hour (km/h), or foot per second (f/s) or meters per second (m/s).

Viscosity. A fluid's resistance to flow under an applied force.

APPENDIX

SYSTEMS OF PHISICAL UNITS

Conversion Factors for recalculation of physical values from technical system of units to international system of units (SI)

Unit abbreviations		
in = inch — (дюйм)	s = second	g = gram
ft = foot — (фут)	min = minute	kg = kilogram (g × 1000)
yd = yard — (ярд)	H = hour	N = newton
oz = ounce	F = force	kN = kilo Newton
Ib = pound	W = watt	Pa = Pascal
BTU = British thermal unit	KW = kilowatt (WxlOOO)	kPa = kilopascal
hp = horsepower	mm = millimetre (m × 0.001)	J = joule
Hg = mercury (mm Hg – міліметри ртутного стовпчика)	m = metre	kJ = kilojoule(J × l000)
	km = kilometre (m × l000)	MJ = megajoule (J × 1000 000)
Conversion factors		
Length	l in	25.4 mm
	1 ft	0.3048 m
	1 mile	1.60934 km
	1 International nautical mile	1.852 km
Area	1 in^2	645.16 mm^2
	1 ft^2	92903.04 mm^2
Volume	1 UK fluid ounce	28413.1 mm^3
	1 US fluid ounce	29573.5 mm^3
	1 Imperial pint	568261.0 mm^3
	1 US liquid pint	473176.0 mm^3
	1 UK gallon	4546090.0 mm^3
	1 US gallon	3785410.0 mm^3
	1 in^3	16387.1 mm^3
	1 ft^3	0.0283168 m^3
Mass	1 oz (avoir)	28.3495 g
	1 Ib	0.45359237 kg
	1 UK ton	1 .01605 tone
	1 short ton (2000 Ib)	0.907 tone
Density	1 Ib/in^3	27679.9 kg/m^3
	1 Ib/ft^3	160185 kg/m^3
Velocity	1 in/min	0.42333 mm/s
	1 ft/mm	0.00508 m/s
	1 ft/s	0.3048 m/s
	1 mile/h	1. 60934 km/h
	1 International knot	1.852 km/h

Acceleration	1 ft/s^2	0.3048 m/s^2
Mass flow rate	1 lb/h	1.25998×10^{-4} kg/s
Force	1 lbf	4.44822 N
	1 lbf	9.80665 N
	1 ton f	9964.02 N
Pressure	1 in Hg (0.0338639 bar)	3386.39 Pa
	1 Ibf/in^2 0.0689476 bar)	6894.76 Pa
	1 bar	100.0 kPa
	1 standard atmosphere	101.325 kPa
Moment	1 lbf in	0.112985 Nm
	1 lbf ft	1.35582 Nm
Energy /heat/work	1 hp	2.68452 MJ
	1 therm	105.506 MJ
	1 Btu	1.05506 kJ
	1 kWh	3.6 MJ
Heat flow rate	1 Btu/h	0.293071 W
Power	1 hp (550 ft Ibf/s)	0.745700 kW
Kinematic viscosity	1 ft^2/s	929.03 stokes = 0.092903 m^2/s
Specific enthalpy	1 Btu/ft^3	37.2589 kJ/m^2
	1 Btu/lb	2.326 kJ/kg
Plane angle	1 radian (rad)	57.2958 degrees
	1 degree	0.0174533 rad = 1.1111 grade
	1 second	4.84814×10^{-6} rad = 0.0003 grade
	1 minute	2.90888×10^{-4} rad = 0.0185 grade
Velocity of rotation	1 revolution/min	0.104720 rad/s

CONTENTS

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КОНСТРУКЦІЯ АВІАЦІЙНОЇ ТЕХНІКИ

АВІАЦІЙНІ ТА РАКЕТНІ ДВИГУНИ

Навчальний посібник

(*Англійською мовою*)

Друге видання, перероблене і доповнене

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