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GENERAL STRUCTURE OF AIRCRAFT ENGINES

AND POWER PLANTS

Lectures summary

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G-20

Разглянуто загальні відомості щодо будови авіаційних двигунів. Наведено основні дані, область застосування, принцип роботи і компонувальні схеми різних типів авіаційних двигунів.

Для англомовних студентів, які вивчають конструкцію авіаційних двигунів та єнергетичних установок.

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This guide will be useful for students studying aircraft engine design in english.

This guide addresses the base technical data, arrangement and construction of aircraft engines. Most diagrams let students get an overview of working process features and construction of main components.

The information is presented in the order suitable for using at practical classes and during individual work. This guide will also be helpful for students studying aerospace engineering to get general overview of air-breathing and rocket engines.

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CONTENTS

INTRODUCTION

The progress in aviation is mostly determined by the progress in aircraft engine design.

The growing requirements to power of aircraft engines at minimum mass and compactness force the modernization of air-breathing engines (ABEs). Gas turbine engines (GTEs) are predominant for the modern aviation. GTE is a heat machine that transforms the chemical energy of fuel into kinetic energy of jet stream or into mechanical energy of shaft rotation. The feature that differs GTEs from other engine types is the gas generator that consists of compressor, combustion chamber and turbine, which function is driving the compressor.

This guide addresses the general arrangement of ABEs. The guide has the chapter that familiarizes students with the liquid propellant engines arrangement. Students can also find the operation principle and construction of GTE base components (intake, compressor, combustion chamber, turbine and nozzle) in this guide. This guide will be useful for students studying the discipline "General structure of aircraft engines and power plants”.

1 SOME FRAGMENTS FROM AIRCRAFT ENGINES HISTORY

The history of aircraft engines starts from first successful flights of Wright brothers in 1903. It is generally divided into two periods:

- first (fourty-year) period is characterized with prevalence of aircrafts with piston engines;
- second (sixty-year) period is a new era of jet aviation.

Wrights succeeded in their first flights because their aircraft was powered by light (for those times) gasoline piston engine. In early forties there was a drastic progress in aviation that was substantially caused by perfection of piston engines. Engines became high power and attained very high structural perfectness.

But permanently growing requirements for higher flight speeds and altitudes of aircrafts at the end of the Second World War met the limits by available power plants with piston engines (PE). Maximum flight speed at the altitude, became the limit for piston aviation. These limitations were caused by the following factors:

– low efficiency and correspondingly low thrust of propeller at high flight speeds;
– practical power limit for multi-cylinder piston engines;
– relatively high mass and overall sizes of piston engines (this factor is caused by periodical working process of piston engines and problems in transferring power from the engine to propeller);
– the complexity of the crank-rod mechanism, etc.

Invention of gas turbine engines (TJEs) eliminated these limits. Unlike piston engines, TJEs operate according to continuous working cycle and have “straight” gas path, which allows high air consumption. The structure of turbojets is simpler than the structure of piston engines; moreover thrust is produced directly, without propeller. Consequently, overall sizes and mass of engines were reduced, maximum practically attainable power limits were augmented. Besides, TJE is higher at high flight speeds. Aviation got new capabilities for further development, especially for supersonic flights.

Piston engines of subsonic airplanes were forced out by turboprop engines (TPEs), which had significantly lower overall sizes and mass, and were able to create more power.

Modern cargo aviation uses not even turboprops, but more perfect turbofans, which are engines of direct reaction. This allows higher flight speeds for aircrafts.

Nowadays, piston engines are used in light aviation and to power small helicopters.

1.1 The beginning of jet engines application in aviation

The first thoughts about benefits of jet engines application for aircrafts appeared in 30th. They turned into set of researches by English, German and Soviet scientists and engineers. The goal of the researches was the designing of experimental specimens of GTEs and jet airplanes. First airplanes were powered by liquid propellant engines (LPE) and next – air-breathing engines (ABE).

First jet airplane (powered by LPE) was German airplane Heinkel He-176 (first flight happened on June 20, 1939).

In 1940 in the USSR happened the first flight of rocket glider by S.P. Korolev, and in 1942 – experimental rocket glider Bi-1 by A.J. Bereznyak.

But ineffectiveness of rocket power plant and difficulties in their maintenance caused the shutdown of rocket glider projects.

By this time, Germany and England succeeded a lot in design of air-breathing engines. The originator of ABE is English engineer, Sir Frank Whittle. He was the first, who started designing TJE in 1928 (that was then patented in 1930). Whittle`s engine was tested in 1937.

The originator of the TJE in Germany was Heinz von Ohain, who tested his simplest TJE in 1937. Engine had centrifugal compressor and centripetal turbine.

Airplane Heinkel He-178 with von Ohain`s engine became the first airplane with TJE in the world. Its first flight took place on August 27, 1939.

First flight of jet airplane with Whittle`s engine in England took place in 1941. In 1943 serial jet fighter Gloster “Meteor” successfully took part in the military operations. Whittle TJE had two-face centrifugal compressor and axial gas turbine. These engines were manufactured by almost all big companies, such as Rolls-Royce, Bristol, etc.

BMW and Junkers have developed first turbojet engines with axial compressors in Germany. Messerschmitt-262 was the first jet fighter with TJE with axial compressor. The manufacturing of this aircraft started in 1944, but because of numerous issues it was not used during the World War II. After the war Germany stopped the development of jet engines.

In these years the USA had no own developments, so engines were manufactured according to English prototypes. Later General Electric, Pratt & Whitney and Allison started designing new engines.

The researches on air-breathing engines (TJE and supersonic ramjet) and gas turbine engines (TPE) in the USSR started much before the World War II. But these researches were cut down because of more urgent problems of military aviation. After the World War II most of the scientific efforts in the USSR were adressed to design and develop the jet engines.

Great scientists B.S. Stechkin, A.M. Lyulka, V.V. Uvarov were founders of soviet school of jet engines design.

In 1929 academician Boris Stechkin put in words the bases of air breathing engine theory.

The founder of soviet research programs in jet engines field was academician, chief designer Arkhip Lyulka, who started researches in 1937. His researches were realized in technical project (RD-1 turbojet) in 1940. Manufacturing of RD-1 turbojet had started in 1940.

When the World War II started “turbojet engine” project was brought to close. In 1941 A. Lyulka patented new turbofan engine scheme (TFE). The first soviet turbojet engine became TR-1, designed by A. Lyulka, TR-1 passed rig tests in 1947. Engine had axial compressor, annular combustion chamber and one stage turbine. This scheme became typical for turbojet engines of next generations.

In 1930 prof. Vladimir Uvarov started researches in gas turbines field and till 1939 made an experimental high-temperature turbofan engine.

These early steps laid the foundation of modern high-thrust engines. Development of aircraft jet engines is of great demand nowadays and is the subject of intensive scientific researches.

1.2 Classification of aircraft engines and their application range

Aircraft engines are heat machines, which are used as power source for aircrafts, which use aerodynamic flight principle for movement and maneuvering in the atmosphere (airplanes, helicopters, winged missiles, aerospace systems). Hence, there is a big variety of available engine types – from piston to rocket. Aircraft engines are divided into three vast classes: piston engines (PE), air breathing engines (ABE) and rocket engines (RE). The classification of the last two classes, especially air-breathing engines (ABEs), will be considered in more details (Fig.1).

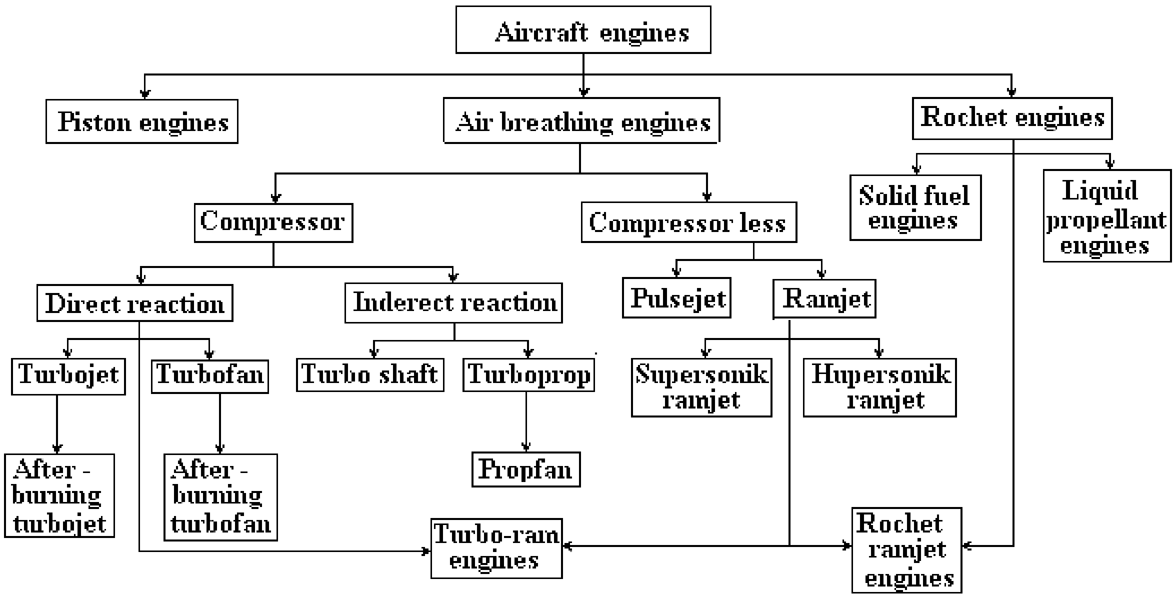


Figure 1 – Classification of aircraft engines

According to the way of air compression, the ABEs are divided into compressor and compressorless. Compressor engines have compressor for mechanical air compression. Compressorless engines are mostly ramjets (air is compressed by the dynamic head) and pulsejets (air is additionally compressed in in special gas dynamic components that operate periodically).

Solid fuel rocket engines (SFRE) do not have special components for working substance compression. It occurs in the beginning of the fuel burning. Fuel charge is burnt in semi-closed volume of combustion chamber.

According to operation principle aircraft engines are divided into engines, which operate according to periodical working cycle (piston engines and pulsejets) and engines, which operate according to continuous working cycle (ABE, GTE and RRE). Engines with continuous working cycle have advantaged in power (thrust), mass etc., which determines their wide application in aviation.

According to the way thrust is produced, ABE are divided into engines of direct and indirect reaction. Direct reaction engines create thrust directly (all rocket engines, turbojet engines without and with afterburning, turbofan engines without and with afterburning, supersonic and hypersonic ramjets, pulsejets and numerical combined engines).

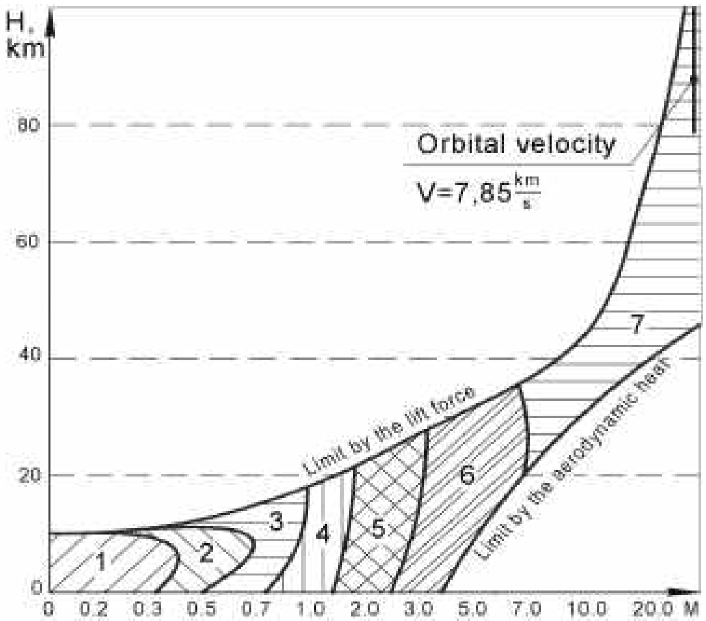


Figure 2 – Application range of different aircraft engines:
1 – TShEs; 2 – TPEs, PFEs; 3 – TFEs; 4 – TJEs; 5 – ATJEs, ATFEs;
6 –supersonic ramjets, LPEs; 7 – hypersonic ramjets, LPEs

Gas turbine engines of indirect reaction transfer produced power to special component, termed propulsor (propeller, propfan, lift rotor of helicopter etc.), which creates thrust (lift force), using jet principle (turboprops, propfans, turboshafts). In this case all engines, which create thrust according to jet principle, belong to air-breathing engines class.

Approximate application ranges of different engine types are presented in Fig. 2. They are limited by lift force and aerodynamic heat. The application area for winged aircrafts is within these limitations.

It is clear that ABEs can be applied for aircrafts in broad altitude and velocity ranges – from $M_{fl}= 0$ (TShE for helicopters and airplanes of vertical takeoff) to $M_{fl} = 10…15$ and more (hypersonic ramjets). Combination of different ABEs and LPEs types gives an opportunity to create the efficient aerospace systems of the future. These systems will be able to perform multiple flights to space being launched horizontally from aerodromes.

1.3 Generations of aircraft engines

Gas turbine engines are complex wares, that is why their improvement was stepwise. Each step took at least 10 years. Scientific and technological progress, inventing materials with better physical properties, permanently growing requirements from aircraft designers made engines with higher structural perfectness and better performances possible. These periods are termed generations. Each generation of engines is characterized by the set of features, such as application purposes (civil or military), type and scheme of the engine and its main components, higher parameters of thermodynamic cycle, reduction of the specific weight and the specific fuel consumption, using new materials etc.

Four first generations of air breathing engines occupied 60 years. Fifth generation started at the beginning of the 90^{th}. Engines of each new generation were more complex and it took more time for their introduction. The beginning of the new generation cannot be set exactly, because in different countries it was different. But the typical technical features of a new generation were very similar everywhere.

The information in Table 1 represents the structure and main parameters of engines that belong to each particular generation. These data show the great seventy-year progress in the designing of aircraft engines: specific weight was ten times decreased, the specific fuel consumption was reduced 2.5 times, the turbine inlet temperature (TIT) grew up from 900 to 1950 K, the pressure ratio increased from 3 to 50 and more. The minimum thrust for light aircrafts stays at initial level (3…5 kN), but maximum thrust of one engine for modern civil two- or four engines powered aircrafts reached 400…500 kN and more.

Let`s consider the features of the fifth generation more precisely, especially military engines.

ATFEs of the fifth generation are designed nowadays mostly for military supersonic maneuverable airplanes and stay the only type of engines of military application. Generally these engines have low bypass ratio, two-spool scheme, usually with contra rotating rotors. Turbine inlet temperature was increased due to monocrystal working blades with intensive convective-film cooling (Table 1).

Higher head of compressor blades, bigger circumferential velocities and new materials let designers decrease the number of compressor stages from 15 – 17 (ATFE of fourth generation) to 10 – 11. New design solutions are applied in the engines of the fifth generation, such as composite casings, powder turbine disk, jointly manufactured compressor blades and disk are manufactured jointly (so-termed “blisk”). Engines of the fifth generation have higher thermodynamic parameters and hence better gas dynamics. Application of new materials and technologies allowed ten times specific thrust increase.

To increase the aircraft maneuverability designers apply engines of the fifth generation with the adjustable nozzles (nozzles with controlled vector jet stream).

More strict requirements are made to engine reliability, lifetime, quick and simple servicing and overhaul. These factors are of prior importance in maintenance.

Subsonic TFEs of the fifth generation are also of high technical perfectness (Table 1). Ecological requirements (noise level, harmful emissions, especially nitrogen dioxide) are the most important for engines of civil aviation. Parameters of interest are high reliability and lifetime, low cost of manufacturing and maintenance.

It is important to point some moments, which characterize qualitative crucial “events” in aircraft engines development as a whole.

1) TJEs and TPEs of the second generation were first time applied for civil aircrafts in 1950s (airplanes Tu -104, Il -18).

2) Cooled turbine blades made it possible to create series of TFEs (third generation, 1960s).

3) Mass introduction of TFE of the third generation for civil airplanes have started since 1960-1970s.

4) Mass introduction of TFEs with high bypass ratios (fourth generation) for subsonic civil aviation have started since 70s).

5) Mass introduction of ATFEs (fourth and fifth generations) for military aviation have started since 1970 - 1990s.

Table 1 – Technical data of aircraft air breathing engines according to generations

Gene-ration, years	Application	Engine scheme	Compressor	Turbine	Specific weight	Specific fuel Consumption M_{fl} =0,8 H=11km	Bypass ratio	$M_{fl\ max}$
1^{st} 1940s	Milit.	TJE ATJE TPE	Single-spool, axial or centrifugal π_c = 3…5,5	No cooling system TIT=900…1150 K	0.6…1.0	1.2…1.4 (TJE)	0	<1.0
2^{nd} 1950s	Milit.	TJE ATJE TPE	Single-spool, axial, with movable GV or two-spool π_c= 7…13	No cooling system, some designs have cooled NVs of the first stage, TIT=1150…1250 K	0.22…0.26	0.8…1.1 (TJE)	0	2…2.3
	Civil	TJE TPE			–	0.25 (TPE)	0	<1.0
3^{d} 1960s	Milit.	ATJE	Two- or one-spool, axial π_c = 10…15 (TJE) π_c = 16…20 (TFE)	Convective cooling of blades TIT=1300…1450 K	0.14…1.8	0.8…0.7	–	2.5…3
		ATFE					0.6…1.5	<2.3
	Civil	TFE					0.5…2.5	<1.0
		TJE					–	2..2.2
4^{th} 1970-1980s	Milit.	ATFE	Two- or three-spool, axial π_c = 20…30	Convective-film cooling of blades TIT=1500…1700 K	0.12…0.1	0.65…0.58	0.4…2	2.2…2.5
		TFE					6…8	<1.0
	Civil	TFE					4…6	<1.0
5^{th} 1990s	Milit.	ATFE	Two-spool, axial π_c= 25…35	Convective-film cooling of monocrystal blades TIT=1850…1950 K	0.1		0.2…0.4	2…2.5
	Civil	TFE	Two- or three-spool, axial π_c = 35…50	TIT^*=1700…1800 K		0.55…0.45	5…12	<1.0

2 APPLICATION OF POWER PLANTS
The concept of “Engine and power plant”

Power plant is a part of an aircraft that must provide propellant force big enough to accelerate an aircraft to velocity, at which lift force exceeds gravity and next – provide its horizontal flight.

Aircraft performances, aircraft flight altitude and velocity depend on power plant perfectness. To increase cruising speed of the aircraft it is necessary to use more powerful engines.

Let`s consider the interaction between terms “Power plant” and “Engine”. Power plant is more general. It may contain several engines, e.g. An-140 or An-74 have two engines.

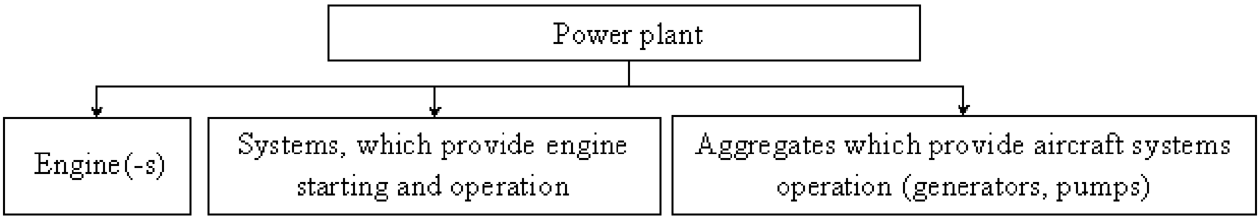


Figure 3 – Power plant structure

Hence it is clear, that engine is a power plant component, which produces thrust (Fig.3).

If thrust is produced by propeller, then engine is known as engine of indirect reaction.

If thrust is produced by engine itself, then engine is known as engine of direct reaction.

2.1 Thrust creation principle

Different engine types are used in aviation, such as reciprocating (piston), gas turbine, rocket engines etc. You already know their classification.

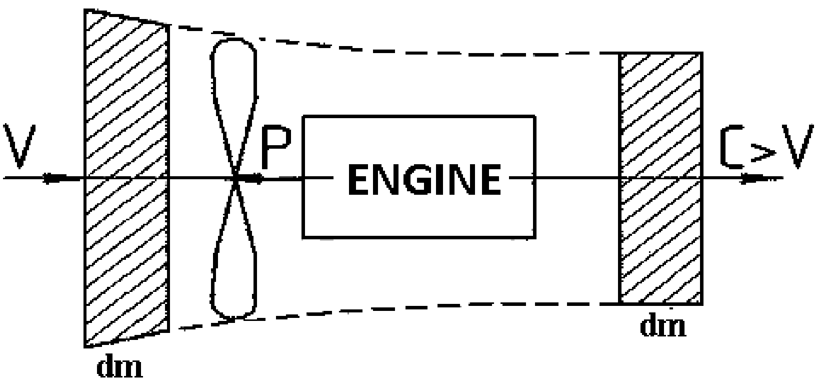


Figure 4 – The scheme of air motion for indirect reaction engine

The main function of any engine is thrust creation. Thrust creation is based on Euler`s momentum change laws:

“The impulse experienced by an object from another object is equal to the change in momentum of that object. The latest object experiences the equal contra directed impulse”.

The force that appears due to momentum change is termed reactive. Let`s apply this law to propeller engine.

For the specific time $\mathbf{dt}$ propeller pushes through small mass of air $\mathbf{dm}$ at flight velocity $\mathbf{V}$ and throws the same mass out at bigger velocity $\mathbf{C > V}$ (Fig. 4).

Momentum changes from $\mathbf{dm \cdot V}$ to $\mathbf{dm \cdot C}$ in the considered volume. Propeller is acted by the momentum, which is equal $\mathbf{Pdt}$. According to the Euler`s law:

$$\mathbf{Pdt = dm(C - V)}; \tag{1}$$

$$\mathbf{P} = \left(\frac{\mathbf{dm}}{\mathbf{dt}}\right)\mathbf{(C - V)} = \mathbf{G_{air}(C - V)}. \tag{2}$$

Eq. (2) is termed the thrust equation. When deducing this equation we have neglected the mass change due to fuel consumption $\mathbf{dm_{air}} \sim \mathbf{dm_{gas}}$. The eq. (2) suites turbojet engine (Fig. 5).

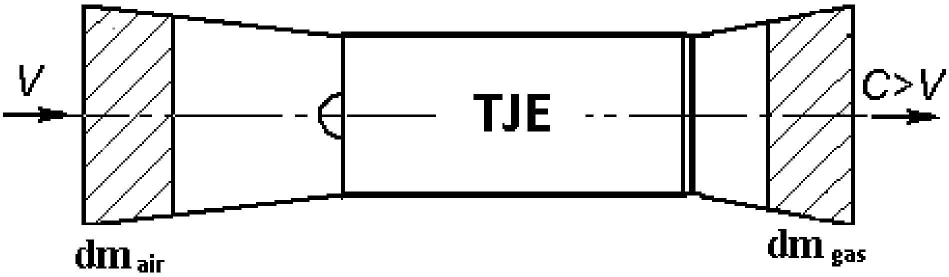


Figure 5 – Air (gas) motion in TJE

2.2 Turbojet engine operation

Let`s consider the operation of the turbojet referring "pressure-volume" diagram. The scheme of turbojet engine in Fig. 6.

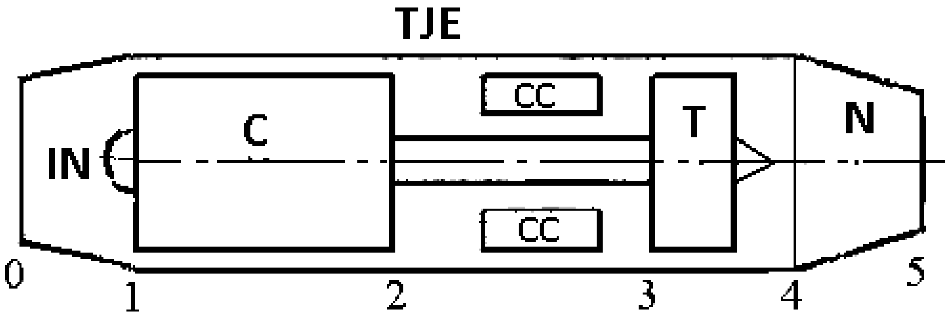


Figure 6 – The scheme of turbojet

During the flight at velocity $\mathbf{V_{fl}}$ incident air flow decelerates and is preliminary compressed in the intake (dynamic compression, point 1). Further air compression occurs in the compressor (point 2).

The dynamic component of pressure rise increases greatly at high supersonic velocities and may contribute a significant part from total pressure rise in the engine.

So, for example, the pressure ratio of Tu-144 engine at flight velocity $\mathbf{V_{fl} = 2200\frac{km}{h}}$ is 18 (half is provided by the intake and the rest – by the compressor).

From compressor air is guided to combustion chamber (CC) where fuel (usually, aviation kerosene) is sprayed. Next fuel and air are mixed in proportions suitable for combustion and burnt. The temperature of combustion products increases to maximum permissible value. The limit is determined by high-temperature strength of “engine`s hot part”. The combustion process on a diagram is presented by the line 2-3 (Fig. 7). The analysis of the diagram shows that volume increase happens at almost constant pressure because of extremely hot gases formation. The potential energy of gases is partially transformed into mechanical work in the turbine (T). The mechanical energy is transmitted to compressor through a shaft.

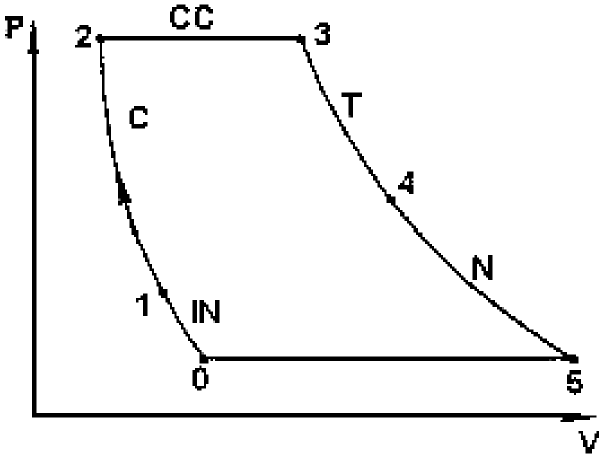


Figure 7 – TJE operation cycle

Turbine pressure ratio (π_T) needed to get mechanical work is spent on air compression in compressor, overcoming friction in bearings and driving auxiliary units. Turbine pressure ratio (π_T) is always lower than compressor pressure ratio (π_C) because of higher combustion products operability (because of high temperature).

Excessive pressure and stagnation temperature at the nozzle inlet are always higher than compressor inlet pressure and total temperature of the incident flow. That is why the velocity of jet stream leaving the nozzle in TJE is higher than flight velocity. Velocity difference forms the thrust.

Schematically, TJE operation can be described as follows:

1. Air is compressed in the compressor.
2. Energy is brought to already compressed air by burning fuel in combustion chamber.
3. Energy is partially retracted from gas flow in the turbine to drive compressor and auxiliaries. This is necessary for normal aircraft operation.
4. Residual part of gas flow energy is used to create thrust by gas flow acceleration in the nozzle.

To increase flight velocity, it is necessary to increase jet stream velocity at nozzle discharge. This is achieved by burning extra fuel in special component termed an afterburning.

Afterburning turbojet (ATJE) differs from previously discussed TJE by afterburning, which is between turbine (T) and jet nozzle (N) (Fig. 8). Extra fuel is supplied to the afterburner through special fuel nozzles. Combustion process is arranged and stabilized in frontal device, which provides mixing of evaporated fuel with airflow. The combustion process in afterburning combustion chamber on a diagram is presented by the line 4-4$^{/}$ (Fig. 9). The analysis of the diagram shows that volume increase happens at almost constant pressure because of extremely hot gases formation.

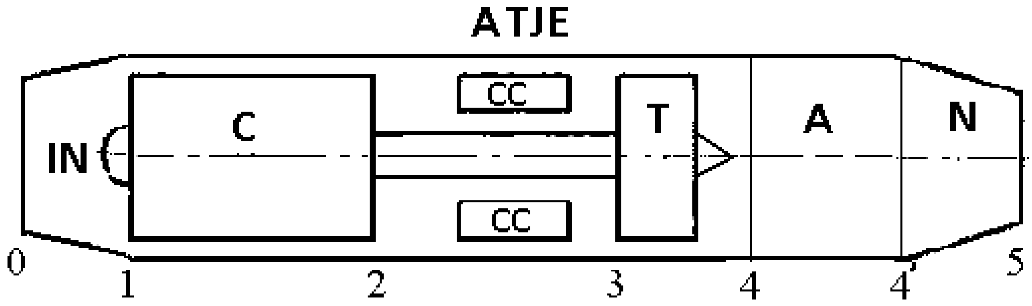


Figure 8 – The scheme of afterburning turbojet

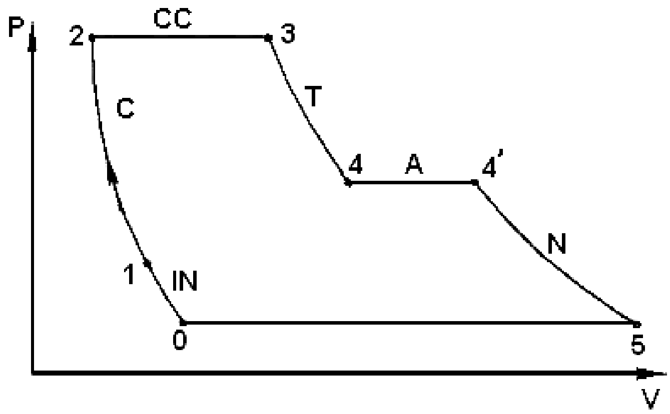


Figure 9 – ATJE operation cycle

The growth in temperature due to burning extra fuel (adding extra energy to the flow) in afterburning combustion chamber increases available combustion products energy, and hence – jet stream velocity at the nozzle discharge. Accordingly, thrust increases. ATJEs are mostly applied for supersonic flight velocities, so they have supersonic are intakes. ATJEs are equipped with Laval`s nozzles (i.e. with expansion portion after “throat”), because expansion ratio in the nozzle at high flight speeds exceeds critical.

Air consumption must be increased to obtain higher efficiency at relatively low velocities. This is achieved by fan application. The fan is driven by its own turbine. This engine type is known as **turbofan engine (TFE).**

TFE is the most abundant gas turbine engine for aircrafts nowadays. Incoming air is compressed in front part of the compressor (F), which is also known as fan. Next airflow is divided into two flows. Primary flow is next compressed in the rear part of the compressor (C) and then is delivered to the combustion chamber (CC). Processes taking place in combustion chamber are similar to the TJE. Combustion products are expanded to the pressures, which are lower than in TJE. Available work of the turbine must be higher than in TJE because of necessity to drive the fan. Fan compresses the air of the secondary flow (Fig.10).

Gas expansion in the turbine is presented by the curve $4–4^{/}$ at the “Pressure-Volume” diagram (Fig.11). During this process the volume of gas increases and its pressure decreases.

Available energy at the nozzle inlet (N) is less than the energy at the same station of the turbojet. At the same time, additional air, passing through the secondary flow, expands in the annular nozzle and hence creates extra thrust. That is why, total thrust increases.

A bypass ratio ($\mathbf{m}$) means the amount of airflow bypassed around the core engine of that which passes through the core (the airflow induced in combustion process).

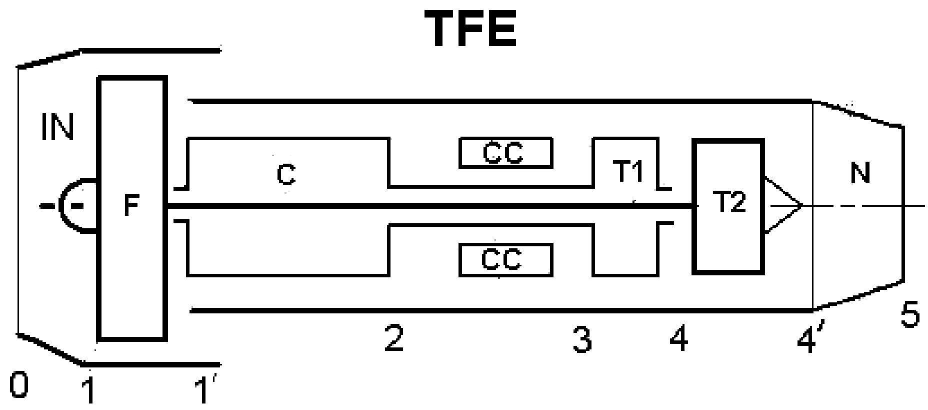


Figure 10 – The scheme of turbofan

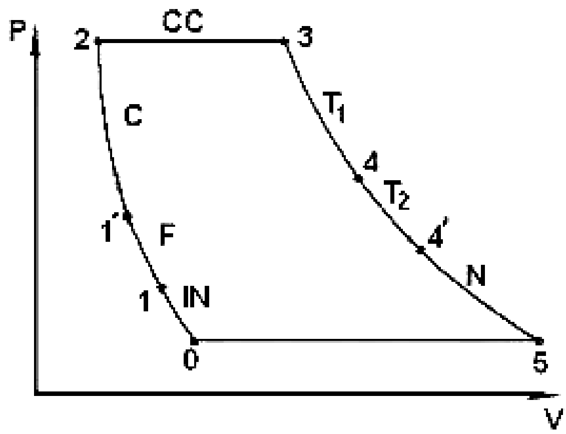


Figure 11 – TFE operation cycle

Afterburning turbofans (ATFE) are widely applied for multipurpose military airplanes. There is a scheme of afterburning turbofan with primary and secondary flows mixing. Here leaving the turbine combustion products are mixed with the secondary air. Next additional heat is supplied to an afterburner, which operates in the similar way to ATJE. Combustion products are exhausted through the common nozzle. Such engine type is termed ATFE with common afterburning combustion chamber (Fig.12). AFTE operation cycle is shown in Fig.13.

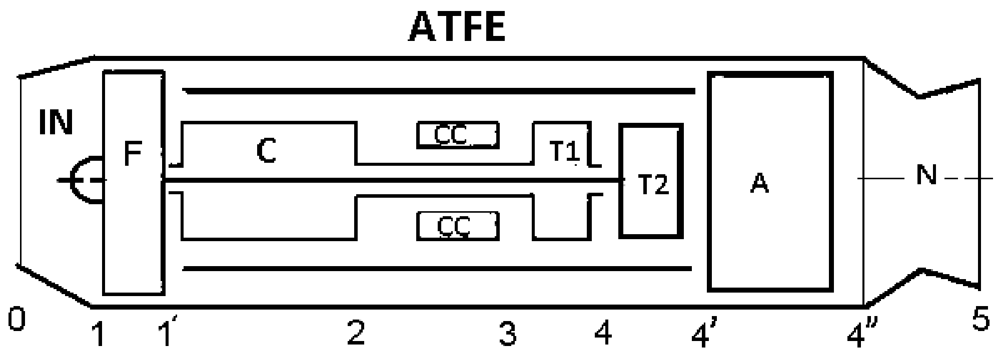


Figure 12 – The scheme of afterburning turbofan

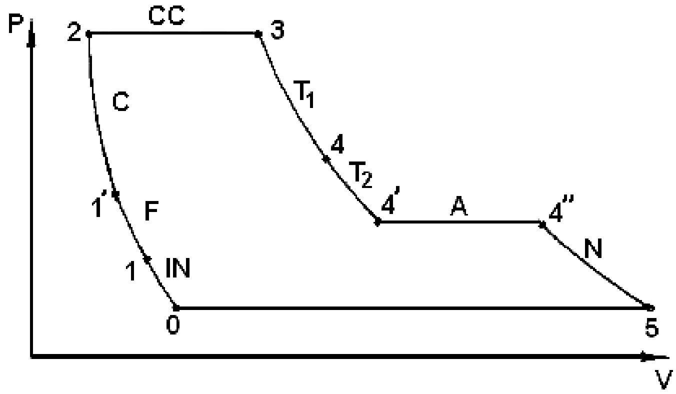


Figure 13 – AFTE operation cycle

Also, **turboprops (TPEs)** are widely applied in aviation. Operation principle and structural scheme of TPEs are similar to TFEs (see Fig. 10). The difference between them is in the way available power is spent. As it was considered, TFEs spent the energy for driving the fan that compresses the secondary air; TPEs – for driving propeller through the gearbox (Fig.14). Gearbox is needed because rotational speed of propeller must be much lower than rotational speed of turbine.

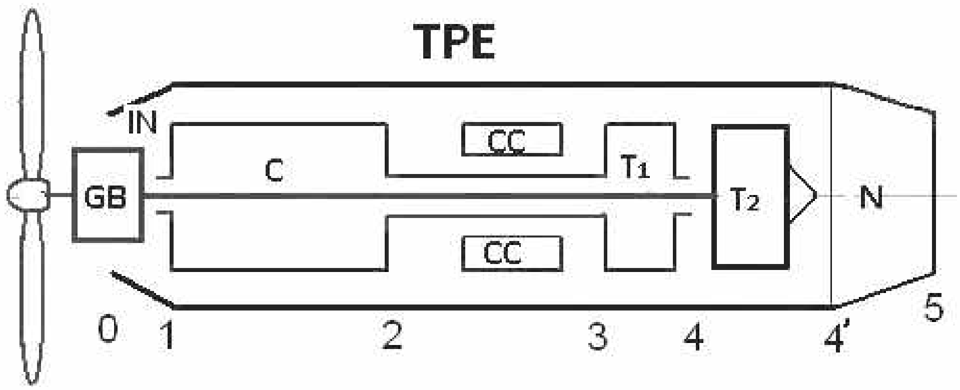


Figure 14 – The scheme of turboprop

Both, propeller and bypass part of the fan have the same function. They accelerate additional airflow to get extra thrust. Propeller produces many times greater thrust than the engine itself. Nozzle produces about 10% from total engine thrust. This is because gas flow energy at nozzle inlet is low. TPE operation cycle is shown in Fig.15.

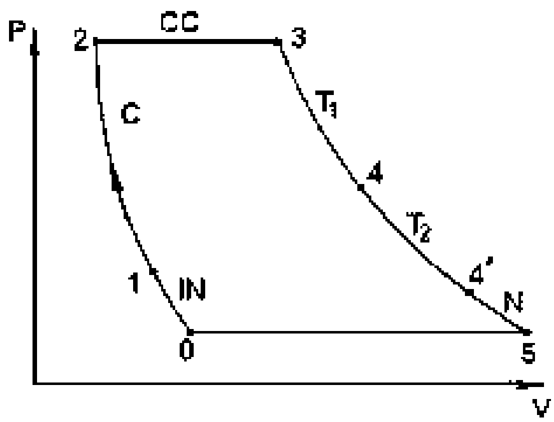


Figure 15 – TPE operation cycle

Turboshaft engines (TShEs) are mostly applied for helicopters. The structural scheme of the TShE is similar to the TPE. Power of the engine is spent to drive the lift rotor of helicopter through gearbox. The thrust, produced by jet stream cannot be leveraged. Because of this reason the energy of the gas flow is entirely retracted in turbine (Fig.16). That is why instead of jet nozzle these engines are equipped with divergent duct.

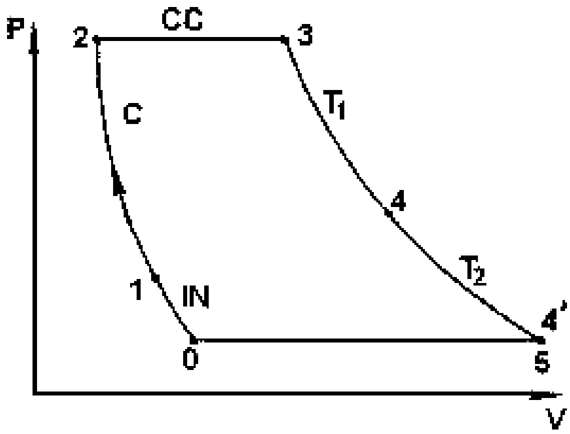


Figure 16 – TShE operation cycle

2.3 Ramjet engines

Ramjets operate according to the same principle as gas turbine engines, but unlike latest, they don`t need neither compressor nor turbine. Incoming air is compressed under aerodynamic head of incoming air. Air, that was compressed in an intake is next delivered to combustion chamber. There it is combusted at maximum possible temperature for fuel combustion. Combustion process is organized and stabilized by frontal device, which mixes the evaporated fuel with airflow. Leaving the combustion chamber gas gets to the jet nozzle (Fig.17). There it is accelerated and exhausted to the atmosphere. This engine type can accelerate the airflow to the highest velocities that cannot be achieved by any other air-breathing engine type. Engine starting cannot be performed from the standstill. So engine must be accelerated up to definite velocity to provide compression enough in the intake.

The idea of ramjet was proposed by Frenchman Rene Loran in 1913. He designed and carried out rig tests of his engine in 1935.

First flight tests of ramjet were carried out by Russian engineer Merkulov in 1939. Ramjet served as booster.

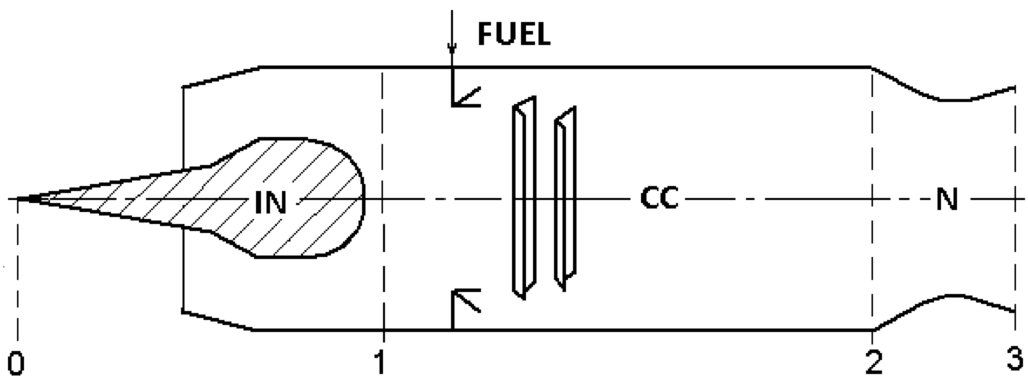


Figure 17 – The scheme of ramjet

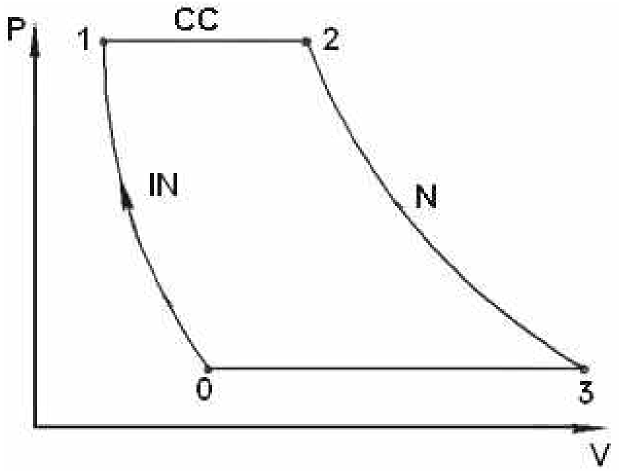


Fig. 18 – Ramjet operation cycle

The strength of "hot part" limits the usage of ramjets. The higher flight speed is, the higher air total temperature in the intake and in the combustion chamber is. The temperature of air combustion is also higher. Ramjet operation cycle is shown in Fig.18.

According to flight speed ramjets may go into three baskets:

– subsonic $(0{,}5 < M \leq 1{,}2)$ (Fig.19);

– supersonic $(2 < M \leq 5)$ (Fig.20);

– hypersonic $(5 < M \leq 10)$ (Fig.21).

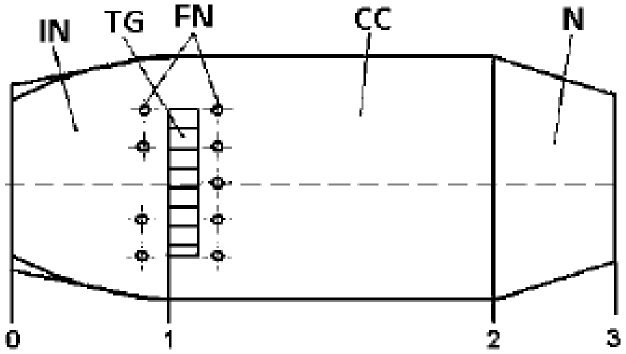


Figure 19 – The scheme of subsonic ramjet

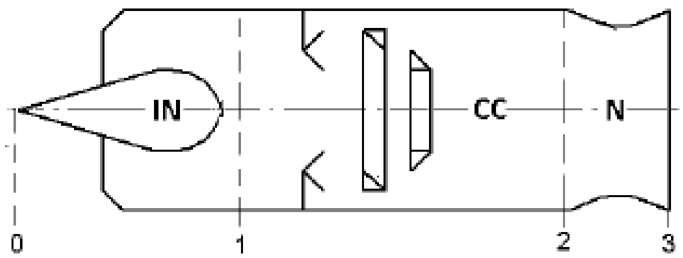


Figure 20 – The scheme of supersonic ramjet

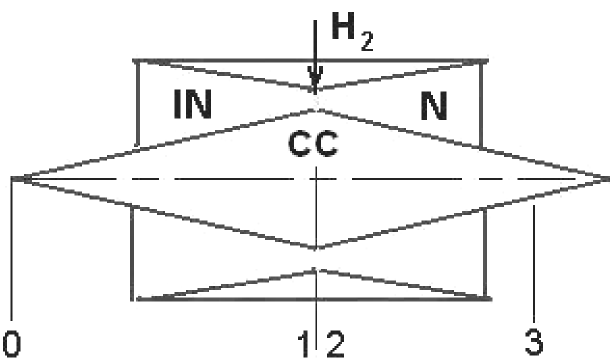


Figure 21 – The scheme of hypersonic ramjet

Having been compressed in the intake of the subsonic ramjet, compressed air gets to the combustion chamber (CC) at subsonic velocity. Combustion finishes before the inlet of convergent jet nozzle (N).

Having been compressed in the intake of the supersonic ramjet, compressed air gets to the combustion chamber (CC) at subsonic velocity. Combustion finishes before the inlet of de Laval nozzle (N).

At extremely high flight speeds that exceed $\mathbf{M_{fl} = 7...8}$, air is compressed in the way not to decelerate the airflow up to subsonic velocity. Air is reasonable to be decelerated up to moderate supersonic velocities, because of lower loses of stagnation pressure in the intake and, hence, working process higher efficiency. Such engine type is known as hypersonic ramjet. Combustion of air/fuel mixture at supersonic velocities is also reasonable for other reasons, for example to facilitate operational conditions of main engine parts. At the same time there are great problems in organizing combustion process at supersonic conditions because of small time air/fuel mixture is in combustion chamber and some other features of supersonic flow. Because of this reason it`s reasonable to burn fuel in normal shock wave. Hydrogen is the most suitable fuel, because its burning temperature is 3550°C.

2.4 Rocket engines

Main difference between rocket engines and ABEs is the necessity to have both fuel and oxidizer onboard. According to the state of oxidizer and fuel, rocket engines can be classified into liquid propellant engines (LPEs) and solid fuel rocket engines (SFREs).

The founder of rocket engines theory is N.Y.Zhukovsky. He published in 1882 the work which addressed the equation to calculate the velocity of jet stream at nozzle discharge.

In 1903 K.E. Tsiolkovsky published the work where he proposed:

– the scheme of LPE;

– to use liquid oxygen as an oxidizer;

– the pumping fuel supplying system;

– to manufacture parts of combustion chamber from graphite and refractory materials;

– to cool combustion chamber with one of propellant components.

First foreign researches in this field were carried out by Frenchman Robert Esnault-Pelterie and American Goddard.

First LPE “OR-1” was tested in 1930–1932 by Zander. It was 49 N thrust and used oxygen/gasoline mixture as a propellant. Next “OR-2” was tested in 1933. It also used liquid oxygen/gasoline mixture as a propellant.

German scientists developed the engine ($\mathbf{P = 250...260\ kN}$) for missile A-4 (V-2). Alcohol was applied as fuel and oxygen – as an oxidizer. The propellant for modern LPEs of rocket-carriers is the mixture of liquid hydrogen (fuel) and liquid oxygen (oxidizer). The thrust of modern engines is $\mathbf{5000...8000\ kN}$.

2.4.1 *Advantages and disadvantages of LPE*

Liquid propellant engines have some advantages compared to air-breathing ones:

– ability to provide thrust in vacuum, because their operation doesn`t depend on ambient conditions (the maximum altitude aircraft with LPEs is able to reach depends only on fuel store in tanks and specific thrust of the engine);

– Unlike any ABE, the higher altitude is, the higher thrust is. Thrust weakly depends on flight velocity in dense atmospheric layers;

– high thrust rate (to 1000 tons) at relatively small overall sizes and low specific weight ($\gamma = 0{,}00102...0{,}00408 \frac{kg}{N}$) of the engine;

– engine weight is less than the weight of any ABE with the same power;

– convenience of engine integration with aircraft because of relatively low overall sizes;

– possibility of reaching high flight speeds and altitudes, which are impossible for ABEs (long-ranged missile A-4 had flight velocity $5500 \frac{km}{h}$ at the altitude 35…37 km);

– engines can be started from the standstill without any auxiliary power units.

Same to other engine types, LPE can perform multiple starting and thrust control. Thrust is controlled by changing the second fuel consumption.

Main LPE disadvantages are:

– very poor efficiency at relatively low flight speeds (high specific fuel consumption);

– short-time operation due to high specific fuel consumption (to $1{,}53...2{,}04 \frac{kg/h}{kgf}$);

– short engine lifetime ($2{,}5s ... 2h$ – LPEs, $500 ... 20000h$ – ABEs, $100 ... 3500h$ – reciprocating engines);

– the bigger distance your aircraft must cover, the more fuel and oxidizer must be onboard (this significantly decreases the payload);

– necessity to carry liquid oxidizer onboard, which safekeeping meets challenges.

Considered advantages and disadvantages of LPE define their application limitations.

As it was mentioned above LPE is a heat machine that transforms one energy type to another. First chemical energy stored in liquid fuel transforms into heat energy. Next heat energy is transformed into kinetic energy of jet stream leaving the nozzle. Chemical energy transformation to other energy types in LPEs occurs with loosing part of the energy. The lower these loses are, the higher level of perfectness engine has.

2.4.2 *Ideal engine operation cycle*

Ideal engine operation cycle means some closed, reversible thermodynamic cycle that consists of the simplest thermodynamic processes. These processes express simplified scheme of real processes in the LPE.

Because of the definition, next assumptions must be made for ideal operation cycle of the engine:

– propellant components are compressed are guided to combustion chamber in an ideal way (without hydraulic resistance in pipes and with negligibly small energy consumption);

– propellant components are sprayed and ideally mixed in combustion chamber (i.e. homogenous mixture is formed);

– both propellant components consumption is time independent;

– propellant is burnt at constant pressure and with full heat generation in combustion chamber;

– combustion products are described as an ideal gas;

– the expansion of combustion products in the nozzle happens adiabatically, i.e. without heat exchange with ambience, without propellant afterburning;

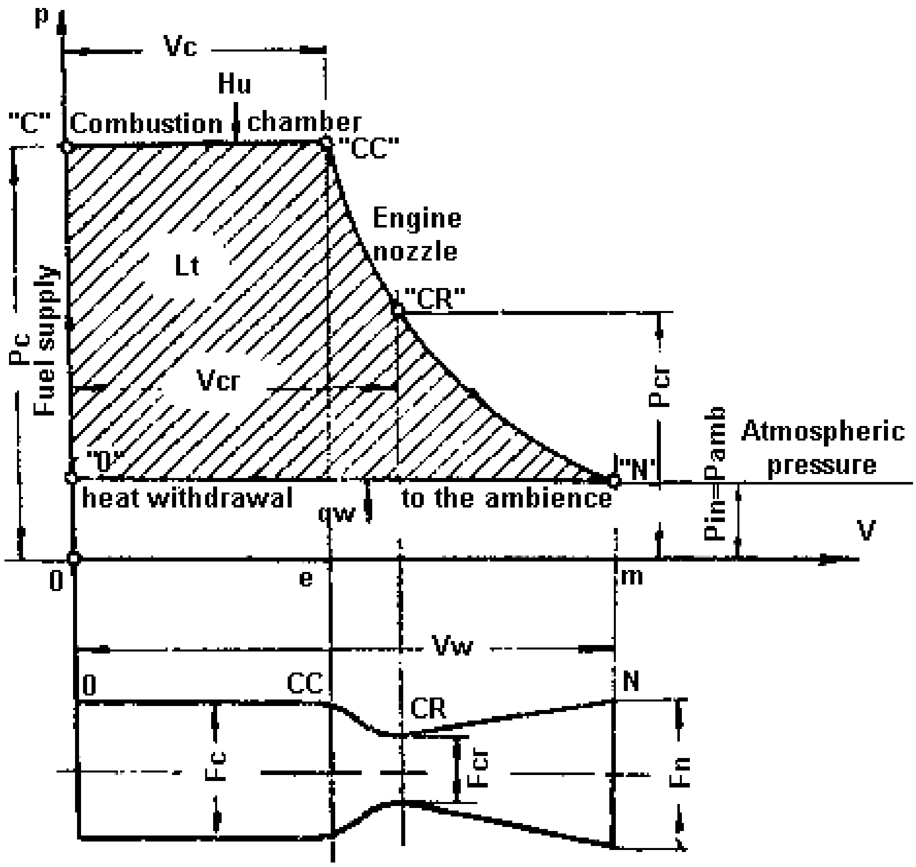


Figure 22 – Ideal LPE operation cycle, expressed in “p-V” coordinates

– there are smooth temperature, pressure and velocity fields in each station of the combustion chamber and the nozzle;

– gas motion at the nozzle discharge is considered as the set of parallel one-dimensional, streaming lines;

– all heat obtained in combustion chamber is used in working process, except the heat of gases that leave through the nozzle.

Hence, ideal operation cycle consists of (Fig. 22):

– compression and fuel supply to combustion chamber (line “0” – “C”, which characterize processes in propellant supply system of LPE);

– fuel burning in combustion chamber (line “C” – “CC”);

– gas expansion in the nozzle (“CC” – “N”);

– heat withdrawal to the ambience (line “N” – “0”, this line is to close of the working cycle).

Full effective specific work of gas in ideal engine working cycle is shown in fig.22 (shaded area).

2.4.3 *Main structural elements of LPE*

Generally, main structural elements of LPE are the following:

1) combustion chamber, which is engine component where propellant combustion happens. Here heat energy of gas is transformed into kinetic energy of jet stream.

Engine chamber consists of injector, combustion chamber and jet nozzle. Injector is used to atomize the propellant at definite weight ratio of propellant components.

Propellant mixing, heating and combusting happens in combustion chamber.

The transformation of heat energy into kinetic energy of jet stream occurs in the nozzle. These processes greatly influence engine efficiency and propulsive performance of the engine.

Processes in the combustion chamber and in the nozzle are associated. Part of chemical energy can be exuded in the nozzle because of propellant afterburning and gases molecules recombination. Amount of heat depends on completeness of combustion process in combustion chamber.

2) Fuel supply system usually consists of one, two or more fuel tanks, propellant components supply mechanism, energy source to drive this mechanism, tubings and fittings (pipelines, valves, consumption washers, etc.), which jointly provide normal starting, operation and shutting down of LPE.

In some instances, fuel tanks are not engine components, but aircraft ones.

Engine must have fuel supplying and control systems that can start and shut down engine when necessary.

3) Ignition system is a set of components for fuel ignition while LPE starting.

Ignition system of some LPEs is not structurally integrated in combustion chamber. Some engines may not have the ignition system at all (in case of self-inflammable fuel components application).

4) Power frame used to fasten engine components between each other and to transfer thrust to the aircraft.

There are two systems, autonomous and airborne, to deliver compressed gas to power units. Autonomous system consists of air cylinder with reducers, pipelines, fittings and valves. All these are necessary to provide the head in fuel tanks and for other purposes.

From presented above it becomes absolutely clear that due to high energy concentration in applied fuels, complex physicochemical processes and safety considerations there is a set of demands made for modern LPEs.

An aspiration to make engine operation fully automatic can be explained by the necessity to make all needed processes in order to provide reliable operation for short periods of time. These processes are propellant ignition, increasing its consumption to nominal value, its maintaining constant or its changing according to engine operation program and finally shutting down an engine.

It must be kept in mind that propellant that is supplied to combustion chamber is an explosive gas mixture. Taking into account that propellant consumption is high, it is clear that any violation in correct propellant supply or delay in propellant ignition may cause its storing in combustion chamber with further sudden ignition with drastic pressure rise, or in other words explosion. The same situation may happen at fortuitous fuel burning termination or in case of restarting propellant supply after engine was shut down. Propellant ignition from incandescent chamber surfaces also may cause engine explosion.

Automatic fuel supply system of modern LPEs is of high perfectness, because all operations (engine starting, its switching to the given mode and shutting down) are performed automatically after only one general command.

2.4.4 *The classification of liquid propellant engines*

Let`s consider the classification according to the following features: structural schemes, operation performances, etc. The variety of existing liquid propellant engines can be explained with the following reasons:

– big variety of applied fuels;

– big variety of application purposes (application determines thrust rate, operation program and time);

– some features of combustion process;

– economical, production and other reasons (different design bureaus have the unique design features).

The structure of LPEs is mostly affected by its application purposes and applied fuel.

The classification will help in understanding advantages and disadvantages of each engine type, their application range, structural and operational features. The majority of existing LPEs can be reasonably classified according to next features:

1. According to engine purposes:

– cruise or main engines(operate during the whole flight or its most part);

– starting engines (facilitate the starting of an aircraft; operate jointly with cruise engines);

– boosters (assist main engines to increase thrust and hence flight speed for a short periods of time).

Liquid propellant boosters are used in aviation. They usually have pumping propellant supply. Pumps are driven by the cruise engines via shafting. Boosters can be started many times during the flight.

Besides, LPEs can be disposable (one time used), or reusable (for some flights).

2. According to applied propellants, there exist LPEs with self-igniting and non-self-igniting propellant components. Properties of applied propellant determine engine structure.

Propellants can be single-component (isopropyl nitrate, nitro methane, diamide, etc.), and two-component – fuel and oxidizer. There also exist three-component propellants.

Two-component propellants are most abundant nowadays.

3. According to used oxidizer:

– oxygen (liquid oxygen, its allotropic modifications and compounds with flammable elements);

– nitric acid (nitrogen oxide, its allotropic modifications and compounds);

– hydrogen peroxide (hydrogen peroxide with liquid or solid catalyst);

– fluorine (fluorine, oxygen fluorine and other fluorine containing compounds);

– chloric (chlorine, chlorine oxides and other chlorine oxide containing compounds).

There are propellants with the combined metal suspension (metalloid) and liquid fuel components. The classification according to applied oxidizer is very significant, because properties of oxidizers determine structural scheme of an engine. Engine designed for applying particular oxidizer cannot operate with another one due to differences in oxidizer properties. Engine designing always starts from choosing oxidizer and fuel for this particular engine.

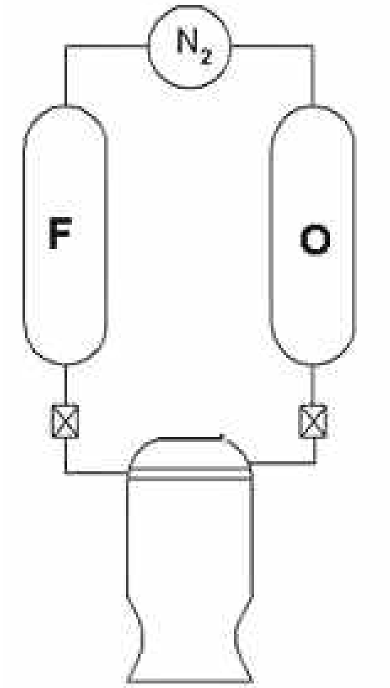


Figure 23 – Pressurized propellant supply system

4. According to propellant supplying system:

– pressurized propellant supply system (Fig. 23), that consists of:

a) high-pressure gas container (HPGC), which contains gas, usually air (HPAC) at high pressure; air from container is delivered to fuel tanks;

b) high-pressure powder container (HPPC), i.e. special chamber with powder charge inside, aimed to generate gases at high pressure by gradually combusting the charge;

c) high-pressure liquid container (HPLC), i.e. one or two gas generators (the chambers where self-inflammable propellant components are mixed and combusted to form gases at high pressure). They are arranged down the tanks.

Pressurized propellant supplying system with high-pressure gas container (HPGC) is also known as gas-container propellant supplying system.

Pumping propellant supply system (Fig. 24), that consists of:

a) turbo-pump unit (TPU), which is the totality of rotary pumps and tubings aimed to deliver propellant components from tanks to engine combustion chamber. These pumps are driven by gas/vapor turbine. This turbine uses gas-vapor, as a working substance. Gas/vapor is obtained in special gas generator (GG) from hydrogen peroxide, isopropyl nitrate, diamide or by combusting propellant components. Gas/vapor turbine may also use gases generated in main combustion chamber;

b) injectors that use kinetic energy of gas during its expansion in special nozzle. Gas is generated in gas/vapor unit).

5. According to heat loading rate:

– “hot” (propellant is burnt at temperature about 2700 – 3600 °C);

– “cool” (decomposition of hydrogen peroxide occurs at 320 – 480 °C).

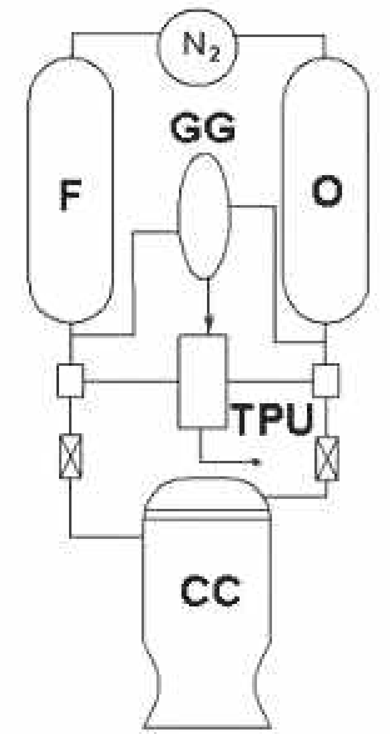


Figure 24 – Pumping propellant supply system

6. According to applied cooling of combustion chamber:

– regenerative cooling(one or both propellant components pass between inner and outer walls of the chamber before getting in combustion chamber and cool inner nozzle casing and combustion chamber;

– effusion cooling (coolant is fed from intercasing volume to inside combustion chamber through small slots in internal casing, which are made from special porous material; next, coolant cools the chamber and forms gas-vapor film on an internal surface of the combustion chamber, preventing internal casing from being overheated by hot gases;

– through-flow water cooling (usually applied at test facilities).

It is also possible to cool the engine chamber with circulating water. Then vaporized water serves as a working substance for turbine of pumping unit (closed regenerative engine cooling).

Cooling type is mostly determined by the calorific intensity of the chamber and the nozzle.

Nowadays, regenerative cooling is the most abundant, reliable and efficient. In this case, heat, taken from internal casing of combustion chamber, is transmitted to coolant and next, delivered to combustion chamber.

Propellant component used as working substance for regenerative cooling must produce least possible corrosive effect, have high heat capacity, conductivity and other favorable for this purpose features.

Oxidizer is used to cool the chamber of a thruster, because amount of fuel onboard is not enough for reliable cooling.

7. According to the way internal casing of engine chamber is protected from being overheated:

– by fuel vapor film, which is created by low flow fuel spraying (fuel nozzles are mounted at periphery);

– by fuel film, which is created in most thermo-stressed parts of the engine chamber. Fuel is supplied on the inner casing surface through special apertures or slots in this casing. Fuel moves downstream internal casing and being heated vaporizes. This prevents casing from extreme overheating;

– by isolating inner surface of the chamber with coatings (ceramics, graphite, metal oxide, etc.).

Effusion cooling by fuel vapor film or fuel film is usually applied only in instances when high thermal stresses render usage of any other, more simple and efficient cooling system impossible or in case applying simpler cooling systems meets structural difficulties.

The cavities of intercasing volume can be slot, spire, slot-spire, etc. From the structural and economical point of view annular cooling cavities are the most profitable.

Casings of chambers can be single-wall or double-wall. First casing type is used for “cool” engines and “hot” engines without cooling. These engines have short lifetime (5…15 seconds). The latest chamber type is used for “hot” engines with cooling, which have long lifetime.

Cooling system must prevent local heat flows being in contact with inner wall of the casing. These flows reach their maximum value near critical cross section of the nozzle at normal operational conditions.

8. According to number of combustion chambers:

– single-chamber (engines with one combustion chamber);

– multi-chamber (engines with some combustion chambers). These chambers can operate simultaneously or separately, depending on required thrust rate.

LPE chambers are made of steel, copper-steel, aluminum, ceramics-steel, etc.

LPE combustion chambers can be cylindrical, conical, convergent, elliptical, pear-shape, spherical, etc.

Optimal shape of the combustion chamber is determined by the sort of applied propellant components, the way they are atomized, pressure in the combustion chamber, needed thrust, engine lifetime, manufacturing process, cost and other factors.

The nozzle of LPE chamber can be:

– conical (cone angle is within the limits 25…35°);

– profiled (ensures axial or almost axial direction of the gas flow at the nozzle discharge).

Nozzles also can be classified according to controllability into adjustable and fixed.

9. According to the way propellant components are atomized:

– spray atomizing;

– atomizing by centrifugal forces;

– atomizing before propellant gets to combustion chamber.

Centrifugal propellant nozzles are divided into single-component or two-component.

Chambers can be welded or assembled. The heads of the chambers may be flat, marquee, spherical, etc. To spherical chamber head belongs the part with propellant atomizing parts and units.

10. According to the way propellant components are ignited during starting:

– chemical ignition (main or starting propellant components are spontaneously ignited getting in contact with each other);

– electrical ignition (by electrical facilities, like sparkling plug or electric arc);

– pyrotechnic ignition by electric squib (igniting flame is formed due to powder block combustion).

11. According to nominal thrust:

– thrusters (about 0,5...5 tons) are widely used for different purposes, like aerial torpedoes, small antiaircraft missiles, starting engines of airplanes;

– intermediate thrust engines (about 5…25 tons) used for big antiaircraft missiles and airplanes, middle-range missiles, transsonic and supersonic interceptors and reconnaissance planes;

– high thrust engines (over 25 tons) are for big intercontinental missiles.

The thrust of engines can be controlled or not controlled during operation. Thrust control is performed by:

– change in propellant consumption $\mathbf{G_{prop}}, \frac{\mathbf{kg}}{\mathbf{s}}$ (the change is usually ensured by varying the pressure in supplying manifold);

– engage or disengage part of fuel nozzles or separate chambers (in case of multi-chamber engine).

If tanks belong to engine components then according to their arrangement LPEs can be:

– engines with in-line arrangement (tanks are arranged one by one along the axis);

– engines with concentric tanks arrangement (one tank is mounted inside another).

Fuel tanks in their turn also can be divided into:

– loads carrying tanks (tanks are components of rocket power frame; they react loads acting the rocket) and loads non-carrying tanks (tanks are arranged inside the rocket case and acted only by working components inside);

– unloaded from the pressure of working gas (pumping propellant supply) and loaded by the pressure of working gas (pressurized fuel supply system, which consists of HPGC, HPLC or HPPC);

– “cool” (for liquid oxygen); “hot” (for propellant component, which is pressurized by hot gas); “normal” (propellant component which is pressurized by cool gas).

– According to engine integration level to the aircraft structure LPEs can be independent from aircraft (engine is fastened to or hanged on aircraft) or integrated to aircraft structure.

2.5 Solid fuel rocket engines

First information about solid fuel rockets (Chinese powder rockets) belongs to XIII century. Nowadays, modern solid fuel rocket engines (SFRE) are widely used as Space shuttle side boosters, submarine ballistic missiles, etc.

SFRE are very simple in design. Essentially, they consist of two main parts – chamber and jet nozzle. SFRE chamber is simultaneously a combustion chamber, which is able to stand high pressure, and the place of fuel stock. Pressure in modern SFRE chambers is within the limits $2{,}94...9{,}8\ \mathrm{MPa}$. SFREs have simple construction. These engines do not have fuel supply system. But operational time of these engines is limited by few seconds or even part of a second and rarely exceeds 2 minutes. That is why these engines are widely applied as boosters, which main goal is to obtain extremely high thrust rates for a very short periods of time. Engines, applied for these purposes, have lower mass than any other engine type. Using boosters as components of auxiliary power plants of aircrafts allows increasing payload and shorting take-off runway. From this point of view, the benefit of power plants with SFREs is that they are always ready for using and do not need tanks refueling before starting. That is why they are applied as cruise engines for missiles of airplanes. Typical example of this is “earth to earth” type projectile. There appeared atomic submarines armed by new powerful solid fuel ballistic missiles, and intercontinental ballistic missiles. Except all advantages, there is a very important disadvantage. After engine was started, burning occurs till all fuel ends, according to definite law and the combustion is uncontrolled. But theoretically, it is possible to stop and reinitiate fuel combustion by controlling pressure in the chamber. Combustion process can be stopped by chamber scavenging or by flame extinguishing by special liquid. Combustion reinitiating is possible only with new igniting charge. Opportune shutting down the engine is feasible nowadays, but re-ignition is still a very complex problem. Combustion control is also very complex. Fuel combustion rate should not significantly depend on pressure and temperature. The thrust control is possible to be performed in preliminary given limits by choosing solid fuel charges of proper geometrical shape and structure. Thrust level control is not the only issue, the direction of thrust is also a problem. Solving this problem requires changing the position of chamber, but the chamber is very big and heavy, because of solid charge inside. There are missiles with complex SFREs with turning nozzles. They do not have problems with the control of thrust direction. The application of SFREs is limited in astronautics. But some powerful SFREs are used for some American rocket-carriers, e.g. rocket-carrier “Titan”. The most important SFRE components are solid fuel charge. Engine performances depend on charge elements, structure and position inside the chamber. There are two main rocket solid fuels: dibasic (colloid) and mixed.

Colloid fuel is a solid homogeneous mixture of substances, molecules of which consist of fuel and oxidizer elements. NC (nitrocellulose) and nitroglycerine are most widely used. Bigger percentage of nitroglycerine in the charge increases the specific impulse of the engine, but propellant explosiveness increases, its stability and mechanical properties become worse. Colloid charges are mostly applied in small engines. Mixed propellant is mechanical mixture of fuel and oxidizer. Inorganic crystalline materials, such as ammonium perchlorate, potassium perchlorate, etc., are usually used as oxidizer in these propellants. Generally this propellant consists of three components: oxidizer, polymer fuel used as sticker, and second fuel in the powdery metallic additive form. Such composition significantly improves propellant performances. Polyester resin, epoxide polybutadiene rubber and others are used as polymer fuel. Powder aluminum, rarely glucinium or magnesiumis are used as second fuel. Mixed fuels generally have bigger specific impulse than colloid fuels. They have higher density, stability, shelf-life, manufacturability. To produce mixed fuel, it is necessary to add dusty oxidizer crystals, metallic powder and other additives to liquid sticker. Next, this compound is thoroughly mixed and put into special forms or directly into the engine casing with air pumped out beforehand. Sticker is polymerized being acted by the added catalysts. Fuel transforms to rubber like substance. Solid fuel is fully disposed in combustion chamber in single or some blocks of definite shape, which are known as charges. Charges are kept by chamber walls or by special lattices, termed diaphragms. The shape of the charge is of great importance. Thrust control law is provided by the shape of charge, preserving coatings of some charge surfaces (these coatings preserve charge surfaces from burning). Thrust control also can be ensured by varying pressure in combustion chamber. There are charges, which provide neutral burning (burning at constant area). Neutral burning can be achieved when charge is burnt from butt-end or simultaneously from inner and outer surfaces (in case when charge has cavity inside) (Fig. 25). In case of regressive burning, burning area decreases on time. Regressive burning can be achieved when cylindrical charge is burnt from outer surface. And finally – charges providing progressive burning. Burning happens with growing pressure in the combustion chamber (burning area increases on time). The simplest example is a charge, which is burnt from inner cylindrical surface. Propellant charge can be ignited pyrotechnically, pyrogenic and chemically.

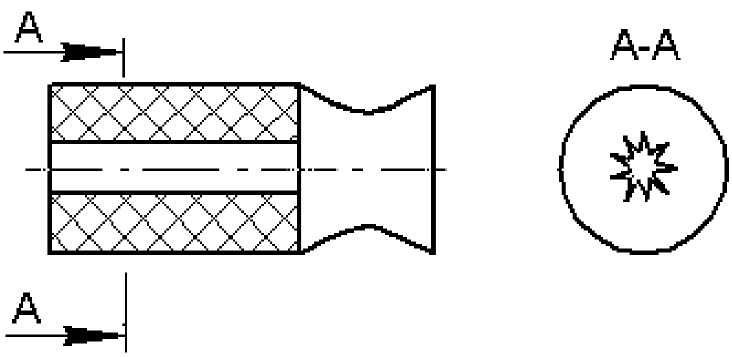


Figure 25 – SFRE scheme

In case of pyrotechnic ignition, electric fuse ignites pyrotechnic igniter, which in its turn ignites main charge. Pyrogenic ignition is performed by gas generator with solid charge inside, which essentially is a miniature SFRE. For chemical ignition, chemically active liquid or gas (oxidizer first) is supplied to combustion chamber, which results in spontaneous ignition. The density of solid fuel is 20…80 % higher than liquid one. This advantage is diminished because of low specific impulse. Fuel in SFREs always depends on shape of engine casing. That is why, ratio between total impulse and total engine weight $\mathbf{G_{EN}}$ (engine weight also considers fuel $\mathbf{G_{fuel}}$) determines engine quality. For majority of modern SFREs $\dfrac{\mathbf{G_{fuel}}}{\mathbf{G_{EN}}} = \mathbf{0,86}$.

If pressure during combustion increases then the specific impulse also increases. But increase in specific impulse results in growth in passive weight of the engine. That is why, the most efficient way is to optimize the ratio of these values.

3 INTAKES

3.1 Requirements and application purposes

Intake is an essential part of a power plant, but simultaneously, structure of the intake is associated with structure of a glider. Intakes deliver air to compressor at increased pressure, and also decelerate airflow.

The following requirements are made to the intakes:

– intake must provide uniform field of air velocities and pressures at compressor inlet;

– minimum hydraulic losses;

– stable operation at all modes;

– simple control;

– intake must be preserved from ice formation at its surfaces;

– minimum mass and overall sizes;

– intakes must be simple, have low manufacturing and overhaul costs;

– high reliability.

Both, designing and development of the intake is performed jointly with the engine.

If flight mode changes or engine operation mode changes, then intake and engine performances change in different ways. Hence, air intake must have control system to coordinate its operation with the compressor.

Intake pressure recovery is used to estimate the perfectness kinetic energy is transformed to potential one. Pressure recovery is a ratio of total pressures at the intake discharge and that of undisturbed flow ahead the intake:

$$\mathbf{\Pi^*_{IN}} = \frac{\mathbf{p^*_{IN}}}{\mathbf{p^*_0}} \quad . \qquad (3)$$

Pressure recovery $\left(\mathbf{\Pi^*_{IN}}\right)$ of intakes of modern aircrafts is $\mathbf{0,96...0,98}$. Pressure recovery of subsonic intakes $\left(\mathbf{\Pi^*_{IN}}\right)$ mostly depends on air friction inside the duct, and of supersonic – also by pressure losses in compression shock waves. Shock waves appear when flow transfers from supersonic to subsonic velocity.

3.2 Intake types

Modern intakes have the variety of types and structural forms.

Intakes are divided into subsonic, transonic and supersonic according to maximum flight speed of an aircraft. The construction of the intake mostly depends on flight speed range of an aircraft, requirements to its maneuverability and engine type.

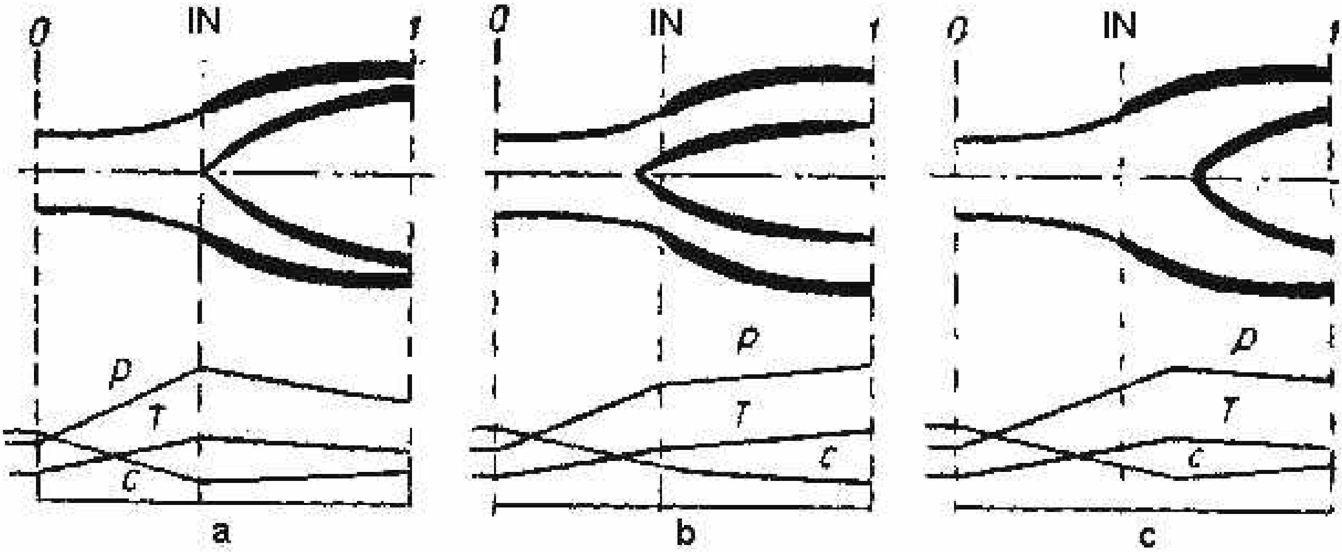


Figure 26 – The schemes of the subsonic air intakes

Subsonic diffusers are applied for subsonic, transonic and supersonic flight speeds (to $\mathbf{M=1,5...1,6}$). Bernoulli equation is an underlying principle of subsonic diffuser operation The equation says, that if flow velocity decreases then static pressure increases. Flow deceleration occurs in divergent duct, termed diffuser. It`s reasonable to make a thick diffuser leading edge to obtain minimum hydraulic loses for subsonic flows.

Different schemes of subsonic intakes are expressed in Fig. 26. They mainly differ by the shape of internal duct. The duct in fig. 26,a is convergent, in Fig. 26,b – divergent, in Fig. 26,c – convergent-divergent. Diagrams in Fig. 26 express the change of airflow parameters in the intake at design mode.

Generally, the shape of the intake is chosen to decelerate the flow before engine inlet at design flight mode. This provides minimum hydraulic compression losses. Next airflow gets inside the duct, where acceleration (convergent duct) or deceleration (divergent duct) happens. The latest improves flow stability and provides uniform field of velocities in station 1.

Airflow parameters alter in the convergent intake in starting and in flight conditions like it is shown in Fig. 27. The flight speed is lower than the speed in station 1.

The size of the boundary layer increases from inlet to discharge. The kinetic energy of the flow in boundary layer is low because of low airflow velocities inside the boundary layer. The width of boundary layer (δ) in the front part of the intake is low, so the cross-section area can be drastically increased (this results is great pressure increments $\frac{\partial \mathbf{p}}{\partial \mathbf{x}}$). But closer to intake rear part generatrix becomes gentler. Such a diffuser is termed isogradient.

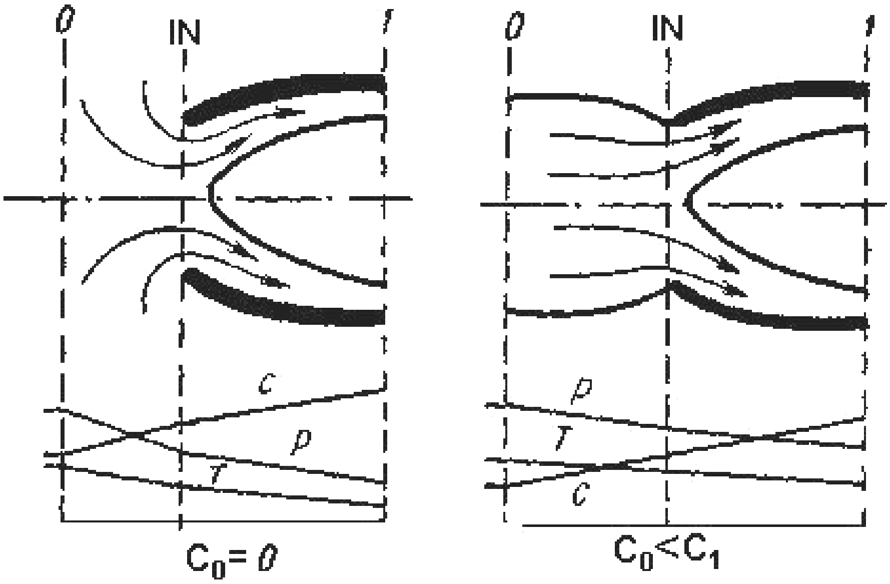


Figure 27 – Different operating modes of the convergent intake

The velocity field at the intake outlet usually has significant nonuniformity. Designers do not use divergent intakes in cases when engine is disposed to unstable operation. To cope with the presented problem designers use convergent intakes that make the velocity field at compressor inlet more uniform. Airflow accelerates a bit, static pressure falls in convergent duct (see Fig. 26,a).

Transonic intakes are applied in maneuverable fighters, which cruising speed is transonic ($\mathbf{V_{fl} = 0,9 - 1,3\ M}$). These intakes provide bigger pressure ratios due to higher dynamic head. The specific feature of these intakes is multimode operation. Airplane intakes are usually designed to be uncontrolled. During supersonic flight, air compression mainly occurs in direct shock wave upstream the engine inlet. Pressure losses in direct shock wave are low up to flight speeds $\mathbf{M_{fl} < 1,4...1,5}$, but next ($\mathbf{M_{fl} > 1,5}$) they grow significantly. This disadvantage is compensated by structural simplicity and low mass. At supersonic velocities (to $\mathbf{M_{fl} \approx 1,6}$) diffuser is designed to have sharp front edges (Fig. 28). In this case, there is a direct shock wave at the intake inlet. Airflow is decelerated from supersonic to subsonic at relatively low stagnation pressure losses ($\mathbf{\Pi^*_{IN} \approx 0,98}$).

The pressure losses in the shock wave directly depend on intensity of this shock wave. The more intense the shock wave is, the higher pressure losses are. Obviously there is a strong need in special supersonic intake, where airflow decelerates to subsonic velocity at low total pressure losses. The best obvious decision is de Laval nozzle with internal compression. Application of this nozzle allows shock-free airflow deceleration. But such intake has single operating mode that is why it cannot have aircraft application.

Supersonic intakes are applied for engines of aircrafts with maximum flight speed ($\mathbf{M_{fl} > 2}$).

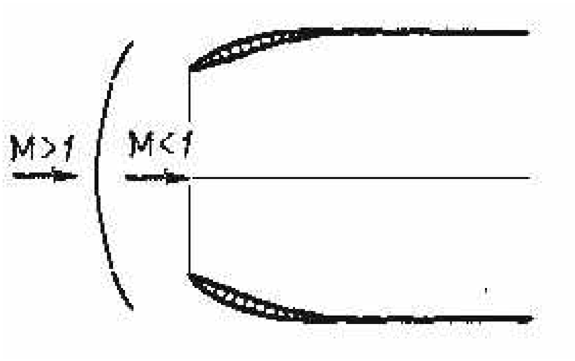


Figure 28 - Intake with sharp front edges

Supersonic air intakes are various. They are classified according to:
– number of braking surfaces and accordingly number of oblique shock waves;
– the position of oblique shock waves relative to intake inlet plane;
– the shape of the inlet cross-section;
– intake position in the aircraft structure.

If the engine is designed for flight velocities, which significantly exceed sonic velocity then incoming flow deceleration is efficient to be performed stepwise (by the set of oblique shock waves). Oblique shock wave set also has losses, but, because airflow parameters undergo less rupture, total losses are lower.

Hence, the pressure after deceleration in oblique shock waves is higher than in single direct shock wave. The higher is velocity of the incident flow the bigger number of oblique shock waves is required for the deceleration and the higher pressure can be obtained at the intake discharge.

Supersonic diffusers can be divided into diffusers with internal, external and combined compression.

Diffuser with internal compression is shown in Fig. 29. There are primary and reflected oblique shock waves, which number depends on the shape of the duct.

Diffusers with internal compression have low external drag. But their wide application is complicated, because it is not possible to provide stable and effective compression in the wide range of flight speeds.

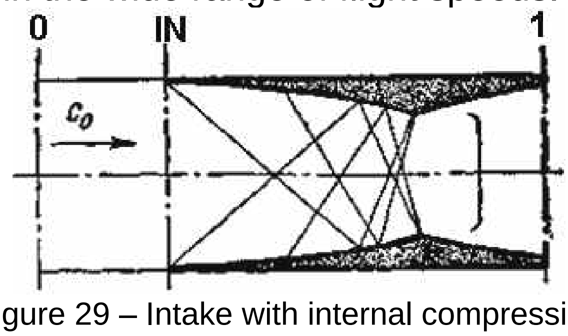


Figure 29 – Intake with internal compression

The most efficient practically implemented method of air compression is when airflow is decelerated to transonic velocities in oblique shock waves or in single oblique shock wave (shock waves are created by specially profiled wedge or cone), and finally – in direct shock wave of low intensity – up to subsonic velocity.

There are two examples of supersonic diffusers with external compression presented in Fig. 30. Oblique shock waves are upstream engine inlet. They are initiated by the specially profiled nose cone mounted flightwise. There is a two-shock waves intake presented in Fig. 30,a. First oblique shock wave decelerates the airflow from the supersonic to subsonic velocity. There is an intake with two oblique shock waves (two stage cone) presented in Fig. 30,b.

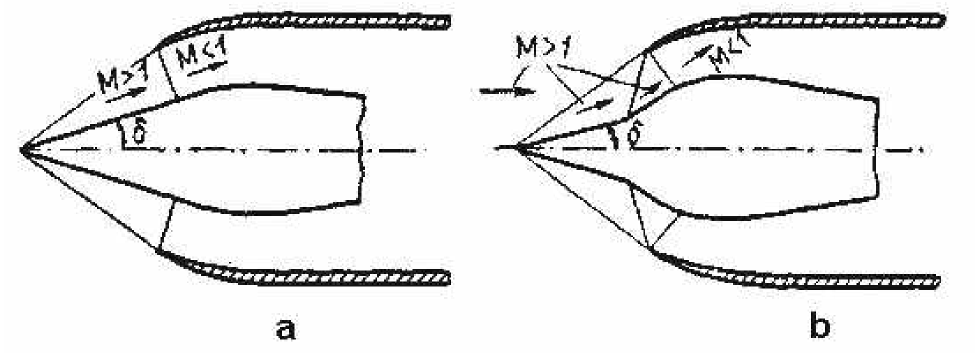


Figure 30 – Intake with external compression

Two considered intakes (Fig. 30) are foreseen to have the subsonic diffuser part for additional air compression after airflow overcomes the direct shock wave. The diffuser part also aims at providing the acceptable flow velocity at outlet of the intake.

Main disadvantage of diffuser with external compression is high external drag. This is because of significant airflow turning and presence of the nose cone, which in its turn causes big tilt angles of external cowl and big basal area.

The more shock waves are arranged in the intake, the lower intensity they have. But necessity in numerous shockwaves results in complicated geometrical structure and control. Extra losses appear when airflow interacts with boundary layer that is present at the wedge or the cone of the intake. The value of losses can be reduced by using the combined compression. Supersonic airflow decelerates in the set of oblique shock waves outside engine inlet (Fig. 31), then in the set of oblique shock waves inside the duct, and finally – in the direct shock wave. Diffusers with combined compression take the intermediate position between diffusers with internal and external compression.

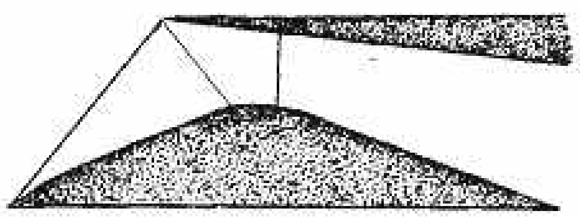


Figure 31 – Combined compression intake

Combined compression intake decreases turning angle of the flow that beneficially affects aerodynamic drag of power plant. But this intake is very responsive to change in flight mode or engine operating mode, as well as to accidental perturbations (afterburning turning on and off, firing aircraft guns, missiles launching, etc.). Control system of this intake arrives very complex that very often makes designers reject its application. For example, supersonic airplane “Concord” has air intakes with external compression, which provide wide range of stable operation with acceptable $\mathbf{\Pi^*_{IN}}$ and simple control system. But at flight velocities $\mathbf{M > 2{,}5...3}$ the benefits of the intake with combined compression become more ponderable than losses caused by control system complication. That is why American airplane SR-71, which cruising speed exceeds three Mach has intakes with combined compression.

Air intakes can also be classified according to the shape of braking surfaces and inlet form into flat and axisymmetric (usually round or semi-round).

Braking surfaces of the flat intakes are formed by the profiled wedge. Engine inlet cross-section is square, sometimes with little fillets in corners. The form internal duct changes gradually from square to round (at engine inlet). The main benefit of controlled air intakes is an ability to vary their geometrical parameters within the wide range. This improves the aerodynamic stability.

Braking surfaces of the axisymmetric air intake are formed by the specially profiled graded cone. The inlet cross-section of the axisymmetric air intake is round (or semi-round), and internal duct gradually transforms from round to annular.

According to the position on aircraft intakes are divided into frontal (arranged in forward part of the fuselage or in the nacelle) and adjoining (arranged near the fuselage of the aircraft, adjoined to it).

Frontal axisymmetric air intakes were widely applied in aircrafts of the first and second generations (like MiG-21, Su-7 etc.), which had cruise flight speeds $\mathbf{M_{fl} = 1{,}8...2{,}3}$. If frontal intake operates at low attack angles then this gives benefits due to uniform velocity field, axisymmetric flow and low pressure losses in the entire operating range. This arrangement also has benefits in weight.

But if attack angles are increased then the performances of the air intake deteriorate and stability factor drastically decreases, especially at supersonic flight velocities. To provide robust operation of the air intake control system must correct attack angle.

Adjoining air intakes generally have flat shape and are structurally more complex. They are applied for maneuverable airplanes of the third and the fourth generations to improve power plant performances. This is especially important when aircraft operates at big attack and slide angles. Flat air intake allows reducing the length of intake duct and its mass by beneficial using interference of the air intake and the airplane. According to the position concerning fuselage air intakes can be various: under the wing, under fuselage, above fuselage and attached from the side.

Main problem of flat intakes is their positioning near airplane surface in the way to provide lesser local attack and slide angles change in intake zone corresponding to attack and sliding angles change of airplane as a whole. This is achieved by shielding effect of adjoined intake surfaces. But air vortexes, which are formed when air flows airplane frontal surfaces, are inadmissible to get inside the air intake. Using oblique shock waves, which are formed when air flows wing or forward fuselage, are expedient to be used as the first stage of preliminary airflow compression. Such intake and airplane coordination improves supersonic intake performances.

To prevent the boundary layer getting inside air intake, it is fastened at some distance from corresponding surface. Hence, boundary layer can be flushed through special slot between the fuselage or the wing and the air intake.

Disadvantages of adjoining air intakes are higher nonuniformity and unsteadiness of the airflow and at engine inlet. There are two sources of this problem: higher airflow nonuniformity at the inlet of the air intake and short intake duct, which length is not enough to smooth the airflow.

4 COMPRESSORS
Compressor application purposes

Compressor is a blade machine, where mechanical energy is supplied to working substance for its compression.

According to airflow movement inside compressor duct, compressors are divided into axial (Fig. 32), centrifugal, diagonal and combined (axial-centrifugal).

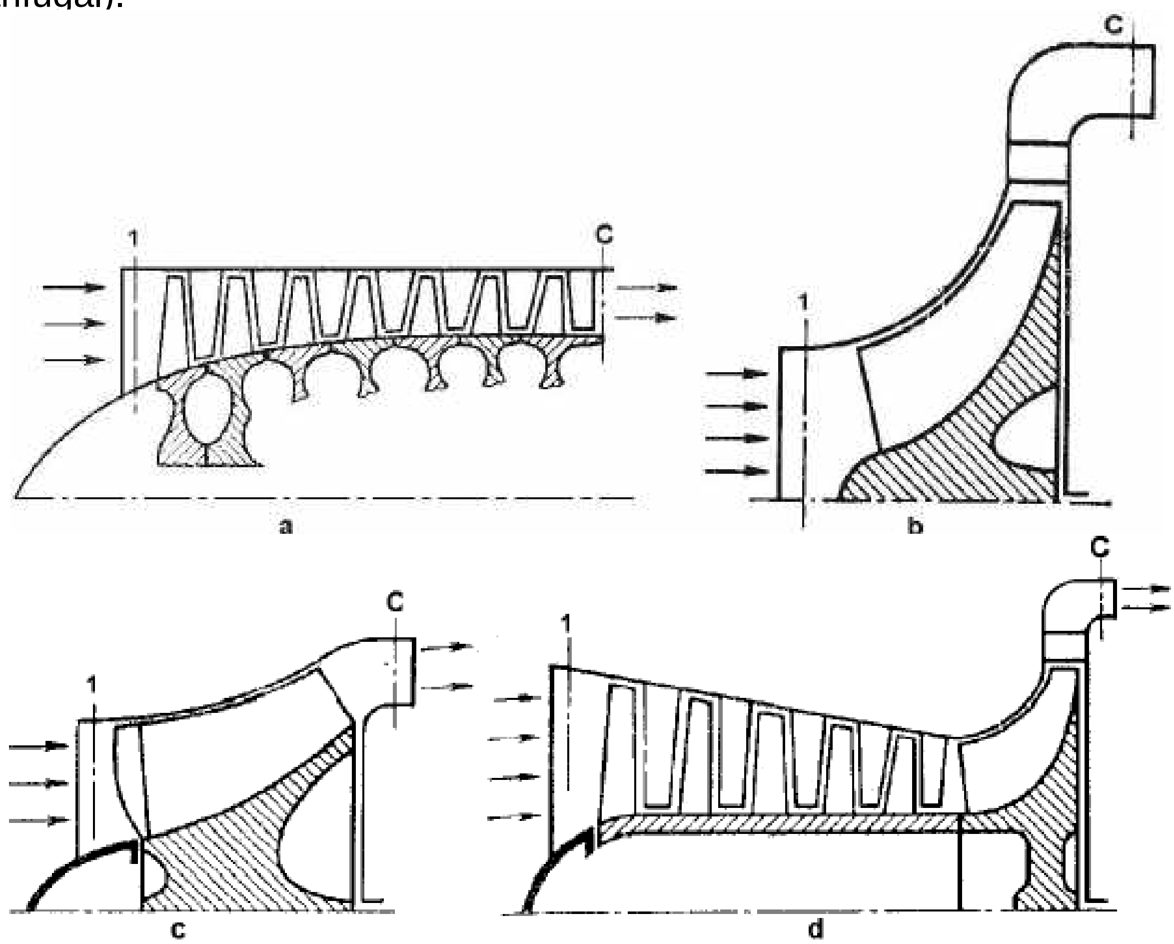


Figure 32 – Compressor types:
a – axial; b – centrifugal; c – diagonal; d – combined

Compressors serve for air compression before delivering it to combustion chamber. At the same time, compressor must compress big amounts of spinning air (air velocity is about $180...250\frac{m}{s}$) at the highest possible level of efficiency.

Main parameters to characterize compressor operation are:

– G_{air}, $\frac{kg}{s}$ – mass airflow;

– $\pi_C^* = \frac{p_C^*}{p_{IN}^*}$ – compressor pressure ratio;

– n, rpm – rotational speed;

– η_{ad} – adiabatic efficiency.

Axial compressors are most widely used for GTEs because they can provide high airflows and high pressure ratios at comparatively low diametrical sizes.

Centrifugal compressors are used in the following cases:

– low thrust (power) engines, which have low airflow and comparatively low pressure ratios;

– as rear stages of combined compressors with high pressure ratios.

All compressors operate in the similar way. They preliminary accelerate the airflow with its further braking in divergent ducts. In the ducts air velocity decreases and pressure rises. Airflow acceleration is ensured by working blades. There are stationary guide vanes (GV) followed by working blades. GV are rigidly fastened in cases. The adjacent GV form divergent ducts.

4.1 Parameters and working process of compressor stage

Axial compressors are widely used in GTEs. Pressure increasing happens in the series of separate stages.

Compressor stage (Fig. 33) consists of impeller and guide vanes. Energy is supplied to airflow in the impeller.

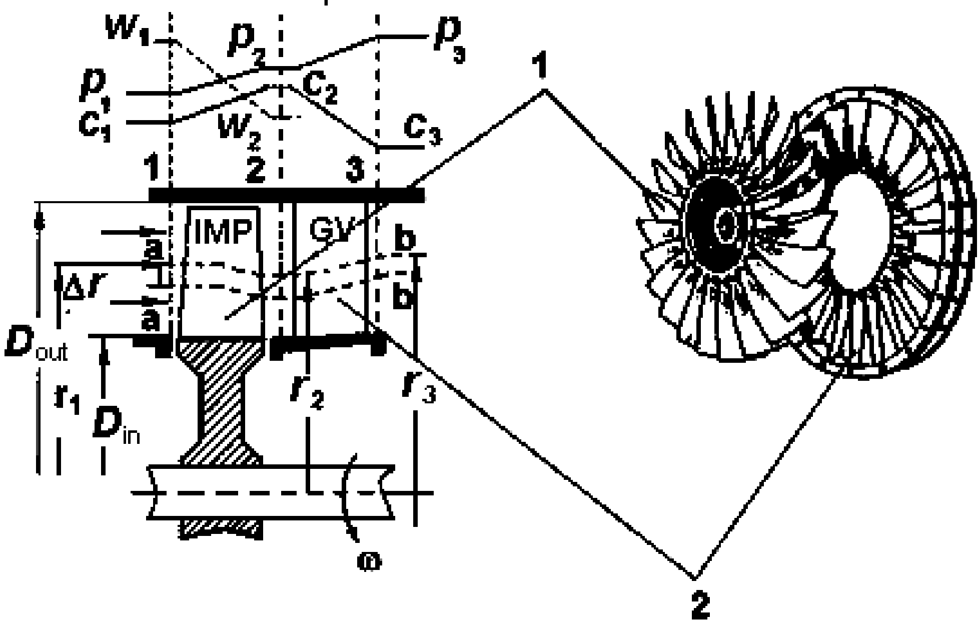


Figure 33 – Compressor stage with typical variation of airflow parameters:
1 – impeller; 2 – guide vanes

To get the elementary stage of axial compressor one must dissect compressor stage by two cylindrical surfaces, which are distant from each other at the distance Δr. Next one must unroll the cylindrical cross-section to get the plane. These scheme will be next used for an analysis (Fig. 34).

There are next velocities denoted in Fig. 34: $\mathbf{c}$ – absolute (as “seen” by an external observer standing next to the engine); $\mathbf{u}$ – circumferential; $\mathbf{w}$ – relative (as seen by an observer “sitting” on the rotating blade and moving with it):

$$\vec{c} = \vec{w} + \vec{u} \quad (4)$$

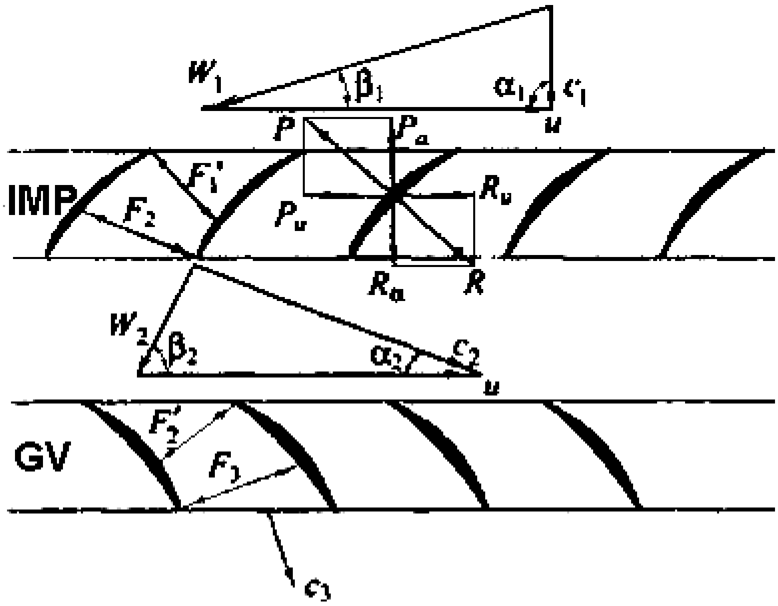


Figure 34 – The scheme of air movement in the elementary axial compressor stage

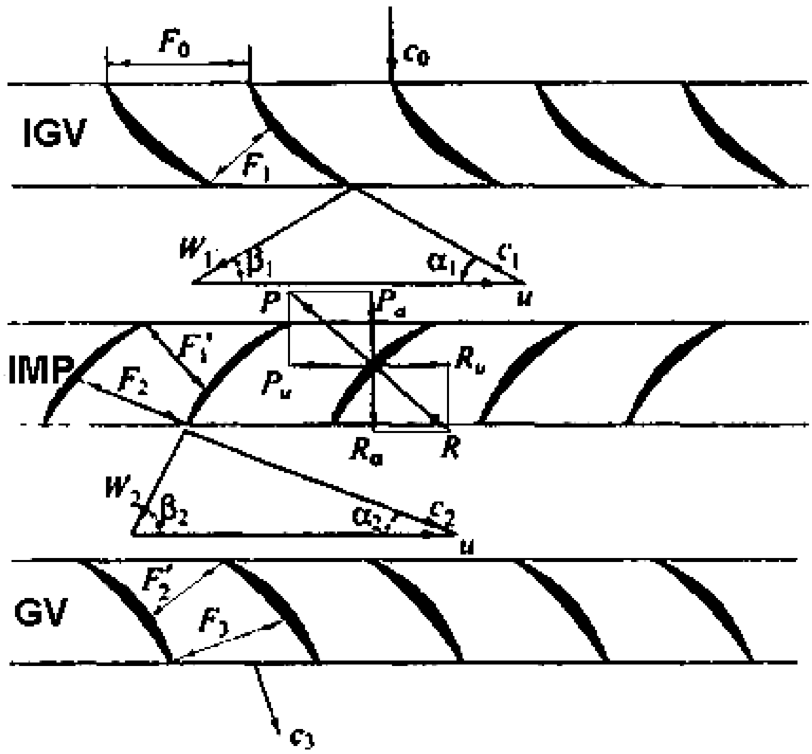


Figure 35 – The scheme of air movement in the elementary axial compressor stage with IGV

Inlet guide vane (IGV) mounted upstream the first stage provide optimal flow direction for the first impeller (Fig. 35).

Areas of outlet stations $\mathbf{F_2}$ and $\mathbf{F_3}$ (see Fig. 34) are always bigger than areas of inlet stations $\mathbf{F_1'}$ and $\mathbf{F_2'}$ in both, rotor blade channel and vane duct. That means that channels are divergent. It means that the velocity of the airflow decreases and pressure increases in these channels.

The analysis of the Fig. 36 shows that using IGV allows decreasing relative velocity $\mathbf{w_1}$ or increasing stage work by increasing the circumferential velocity.

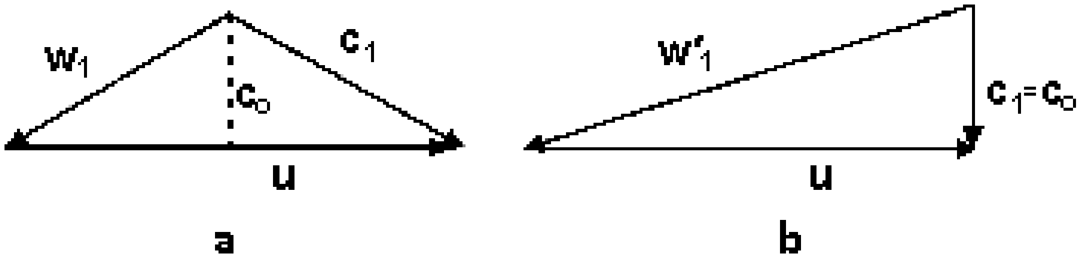


Figure 36 – The comparison of velocity triangles at the first stage inlet:
a - with IGV; b - without IGV

Blades of elementary stage are profiled in the special way. They are acted by aerodynamic forces **P**, that can be decomposed into axial $\mathbf{P}_a$ and circumferential $\mathbf{P}_u$ components. Circumferential component $\mathbf{P}_u$ creates the section modulus of compressor which must be overcome by the applied work. Force $\mathbf{R}_a$, which is equal to $\mathbf{P}_a$ but has an opposite direction, produces work in the impeller. Hence the absolute velocity $\mathbf{C}_2$ at impeller discharge becomes higher than the same velocity at impeller inlet $\mathbf{C}_1$. Total pressure correspondingly rises: $\mathbf{p_2^* < p_1^*}$. Static pressure increases along the flow path between adjacent vanes. Airflow velocity decreases in divergent duct.

4.2 Radial and axial clearances

Radial clearance between rotor blades and facing them casing is assigned to meet requirements that radial clearance must be positive at all operating modes, considering transients. Radial clearance is mostly affected by the centrifugal forces and thermal expansion. Manufacturing tolerance also must be considered.

Radial clearance significantly influences the airflow at the blade tip. Air partially flows from high pressure to low pressure side of the blade through this clearance. Leaked air does no work making compression less efficient. Efficiency is influenced by relative radial clearance. One percent clearance increase results in 1.5…3% efficiency drop, and 3…5% aerodynamic head drop. That is why the efficiency of rear stages is lower because they are shorter.

Axial clearances significantly influence the efficiency of blade operation. They also determine the loads caused by blade vibration. Air that leaves the trailing edge of the blade forms vortexes. The intensity of vortexes decreases by moving away from the trailing edge of the blade. Airflow is smoothed at the distance $0,6...0,8$ of airfoil chord. But these clearances significantly increase engine mass and length.

Non uniform airflow results in cyclic aerodynamic forces, which may initiate dangerous blades oscillations of the next airfoil set. These oscillations may cause higher noise levels.

Generally axial clearances are $0,15...0,25$ of rotor blade chord. Single stage fans have axial clearances between rotor blade and following guide vanes. These clearances equal $1,0...1,5$ of chord for noise level reduction.

4.3 Multistage compressors

As it was mentioned, the channel formed by the adjacent blades is divergent that results in pressure increase. Pressure ratio of subsonic stage does not exceed $\pi^*_C = 1,2...1,4$, and supersonic – $\pi^*_C = 1,5...2,5$. Bigger pressure ratios can be obtained in the multistage compressors ($\pi^*_C = 20...40$). The pressure ratio of multistage compressor can be calculated as follows:

$$\pi^*_C = \pi^*_{ST\ 1} \cdot \pi^*_{ST\ 2} \cdot ... \cdot \pi^*_{ST\ n} . \tag{5}$$

Parameters of air passing through the compressor are changed: pressure increases, density of air increases, axial component of airflow velocity stays almost constant. The cross-sectional area of gas-path, which is formed by inner and outer casings of the compressor decreases from stage to stage. Elementary compressor component is a stage. Stage is formed by impeller and guide vanes. The absolute majority of modern compressors are multistage. There is an example of four stage compressor shown in Fig. 37.

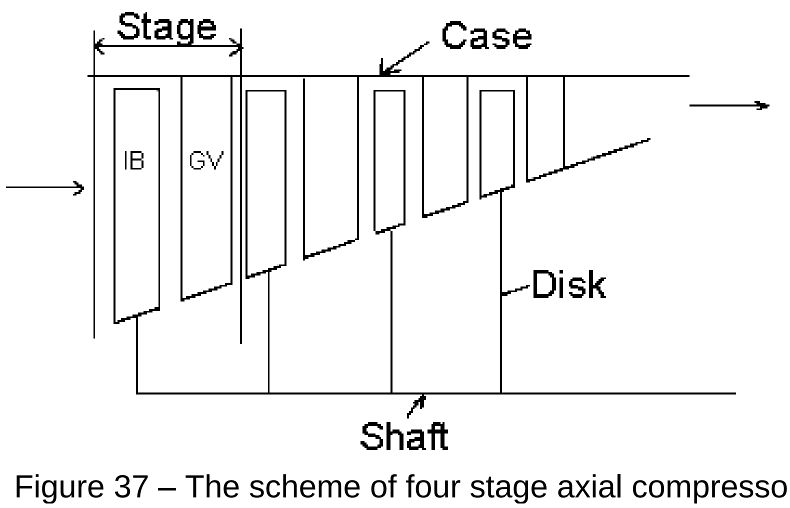


Figure 37 – The scheme of four stage axial compressor

Designing new compressor is carried out in terms of mass and overall sizes reduction and efficiency improvement. To provide above mentioned requirements designers must choose gas path shape, calculate number of compressor stages and distribute available work between compressor stages.

Work is distributed under the following logic:

– work, done by front stages, is below the average value because of the non-uniform airflow at engine inlet;

– work, done by rear stages, is also below the average value because of big relative radial clearances.

4.4 Axial compressor classification

1) According to airflow velocity:
– subsonic;
– supersonic.
2) According to number of spools:
– single spool;
– two spool;
– three spool.
3) According to rotor structure:
– drum type rotor;
– disk type rotor;
– drum and disk type rotor.
4) According to gas path shape:
– $\mathbf{D_h = const}$ (hub diameter is constant);
– $\mathbf{D_t = const}$ (tip diameter is constant);
– $\mathbf{D_m = const}$ (mean diameter is constant);
– combined.
5) According to casing structure:
– non-detachable (single-piece) casing;
– detachable (casing with splits).
6) According to the arrangements to prevent the surge:
– compressor with movable guide vanes;
– compressor with ABVs (air bleed valves);
– compressor with both, movable guide vanes and ABVs.

4.5 The construction of axial compressor elements

1. Casing. All immovable parts form the casing (external casing, casing of guide vanes, casings of supports).
2. Rotor is the compressor component, which is formed by all movable parts (shaft, disks, drum, rotor blades).

The classification of rotors

Drum type rotor is a cylindrical or conical drum which is used to attach blades. Rotor is closed from the face and from the rear by the covers. There are cylindrical surfaces for rotor supports arrangement from the front and the rear.

Advantages: simple construction, high bending rigidity.

Disadvantages: limited circumferential velocity ($\mathbf{U = 180...200\,m/s}$) to meet the strength requirements.

Disk type rotor is the shaft directly joined with disks. Blades are also fastened to disks.

Advantages: opportunity to operate at high circumferential velocities (to $U = 250...360\, m/s$). This rotor type is mostly used for heavily loaded compressors.

Disadvantages: low bending rigidity, increased mass.

Drum and disk type rotor is formed by disks, coupled together by drum fragments, which transmit the torque.

Advantages: unites advantages of both, disk and drum rotor types (high circumferential velocities and high bending rigidity).

Disadvantages: increased mass, complex construction.

Rotor blade consists of blade body and lock.

Blade body is a specially profiled airfoil. This profiling is coordinated with strength analysis and oscillation analysis. Blade airfoil surfaces must have high microinch accuracy (▼ 8 – 10). This sounds reasonable because high microinch accuracy reduces friction loses and improves fatigue strength of the blade.

Lock of the blade:

The most widely used locks are:

– dovetail (trapezoidal lock), mounted in axial type dovetail groove;

– dovetail (trapezoidal lock), mounted in circumferential type dovetail groove;

– hinged lock;

– fir-tree lock (used rarely).

Casings

Compressor casing consists of front and rear frames.

Compressor casing is tube-like or truncated cone-like construction, which shape is determined by gas path profiling law.

Ball bearing support forms the front frame, and straightening guide vanes jointly with the casing of the rear bearing form rear frame.

Casing can be detachable or non-detachable.

Detachable casing can be classified according to splits configuration:

– with transversal split;

– with longitudinal split;

– with longitudinal and transversal splits.

Inlet guide vanes, guide vanes and straightening guide vanes are locked in the compressor casing.

Locking can be:

– cantilever (mostly for short vanes);

– two-sides (mostly movable guide vanes and vanes that transfer loads from supports).

Expanding range of axial compressor stable operation

Aircraft engine operates in wide ranges of flight speeds and altitudes. Its operation mode changes from idle $n_{gg} = 0,5...0,7 \cdot n_{gg\ max}$ to maximum $n_{gg\ max} = 1$.

That is why increasing attack angles can be followed by the stall at compressor blades. Stall is a reduction in the lift coefficient generated by an airfoil as angle of attack increases. This results in pressure vibration. This, in its turn, leads to airflow velocity change in gas path and to surge.

There are three methods to provide stable operation:

– movable inlet guide vanes;
– movable guide vanes;
– air bleeding valves.

4.6 Classification of centrifugal compressors

Sometimes, centrifugal compressors are applied instead of axial. In centrifugal compressors the airflow changes its direction in the impeller from axial to radial.

Impeller blades accelerate and twist the airflow. Centrifugal forces intensify compression of the airflow passing through the impeller. Hence, centrifugal stage produces higher pressure ratios than axial but at lower efficiencies. Centrifugal compressors nowadays are applied in auxiliary power plants and sometimes – in small GTEs for helicopters. Their main advantage is compactness.

Centrifugal compressors are classified according to next features:

1. Structure:
   – single-face compressors (used for engines with low airflows);
   – two-face compressors with two-side inlet (used for engines with high airflows).
2. Construction of impeller blades:
   – compressors with radial blades arrangement;
   – compressors with blades arranged at an angle to radial direction:
      a) blades, twisted to hand of rotation (higher aerodynamic head);
      b) blades, twisted to opposite to hand of rotation (better compressor efficiency).
3. Number of stages:
   – single stage compressors;
   – two stage compressors;
   – multi stage compressors.
4. Diffuser type:
   – vaneless diffuser;
   – vaned diffuser.

Construction of centrifugal compressors

Impeller.

Usually impeller blades are manufactured bodily with disk from aluminum or titanium alloys and steels. They are forged with further mechanical treatment (low size impellers are casted).

Movable guide vanes furnish the airflow to the following blade raw at minimum losses. Usually, it is a separate part, which is fastened to the impeller.

Casing

Casings usually consist of some parts:

– casing of the intake duct;
– diffuser casing (bladed annular duct);

– structural elements (power frame);

Casing may have immovable guide vanes. They aim to twist incoming air to hand of rotation. Twisting the airflow decreases the relative airflow velocity at impeller inlet (Fig. 38).

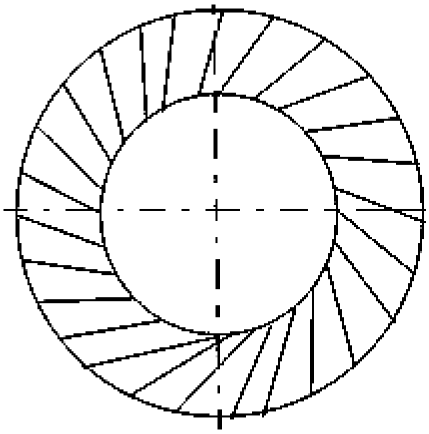


Figure 38 – Casing of centrifugal compressor

5 GAS TURBINES

Turbine is an engine component that serves to drive compressor, propeller or lift rotor of a helicopter. Some turbines (e.g. auxiliary power units) may be used to drive electric generators or auxiliaries.

As well as compressors, turbines also can be classified according to the direction of the airflow into axial, radial, diagonal and combined. But the vast majority of aircraft turbines are axial.

Axial gas turbine, same to compressor is a blade machine. But, opposite to compressor, which consumes mechanical work, turbine produces it. The turbine transforms the energy of the gas flow to mechanical work. Compressor uses up the work produced by the turbine to compress the air.

As well as compressor operation, turbine operation obeys the rule of “Acceleration-Deceleration”. But there are some different features. The acceleration of the airflow in compressor is provided by mechanical work that is supplied by the turbine, and deceleration – basing on gas dynamic laws. The acceleration of the gasflow in the turbine happens basing on gas dynamic laws, and deceleration is performed mechanically.

5.1 Working process and main parameters of the turbine stage

Turbine stage consists of nozzle vanes and impeller. Nozzle vanes form the convergent curvilinear channels. That is why, flow velocity rises there, and potential energy is transformed to kinetic. Nozzle vanes also provide the needed flow twisting at impeller inlet.

Rotating blades also form convergent channels, which make the gas moving through them expand. When gas expands, the relative velocity rises and temperature and pressure decreases. Absolute velocity at impeller discharge $\mathbf{C_2}$ is much higher than impeller inlet velocity. $\mathbf{C_1}$. Parameters change because gas does work, which is transferred to shaft via blades and disk. The density of gas decreases and gas-path cross-sectional area and blades length increase from stage to stage (Fig. 39).

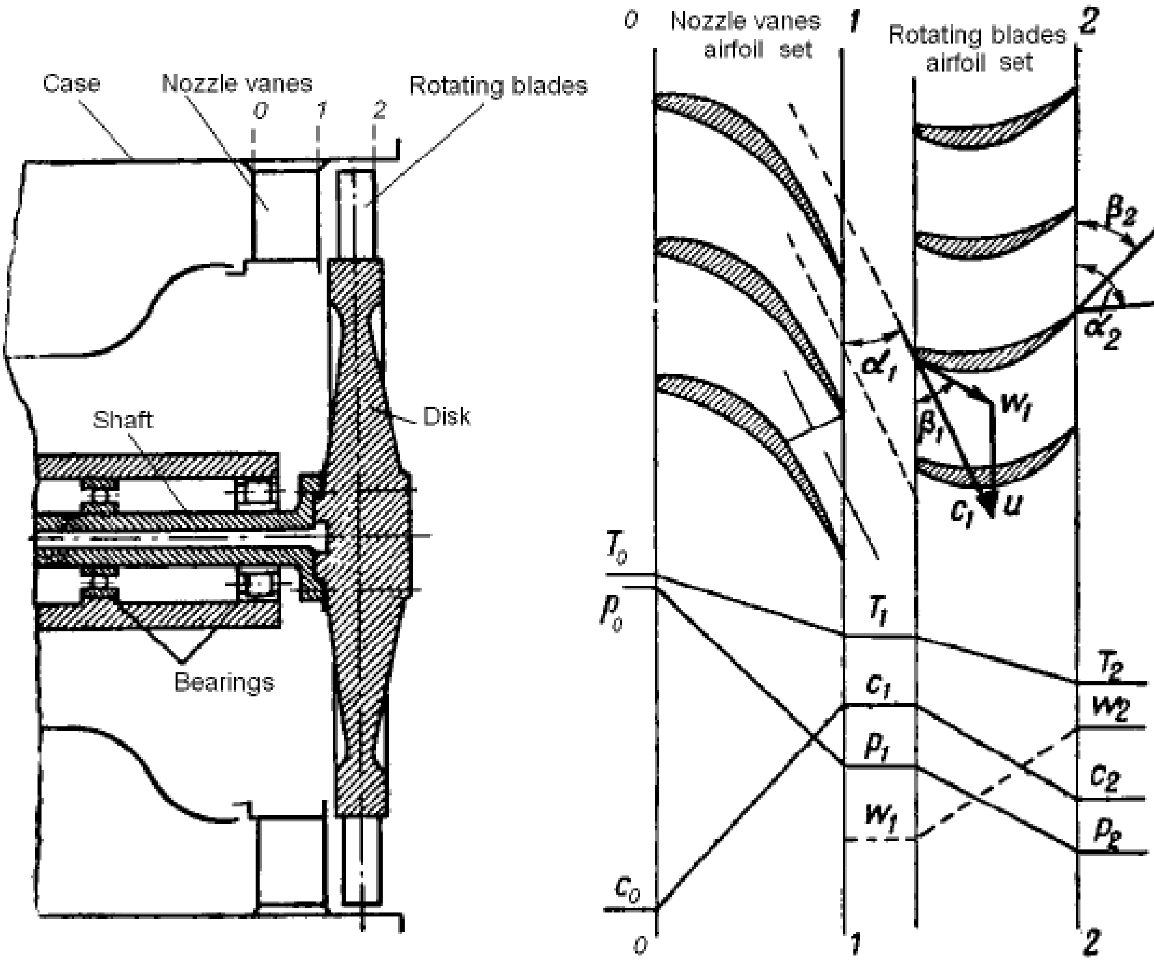


Figure 39 – The scheme of the single-stage axial turbine

As it is seen from Fig. 39 the maximum velocity of the gas flow is at nozzle vane discharge. The velocity may reach sonic values. This is especially typical for high pressure turbine, which is designed to be single-stage. Pressure ratios of the HPT are big ($\pi^*_T = 3{,}5 - 5$). Reaching sonic velocity in minimal channel section (nozzle throat) at nozzle, gas flow gets to divergent channel (known as oblique shear) and accelerates to supersonic velocity. Gas flow from nozzle vanes gets to the inlets of rotating blades, which are mounted in the disk. Circumferentially directed force appears due to interaction between gas flow and rotating blades. Disk starts rotating.

The fundamental difference between the impeller channels of compressor and turbine is the form of the channel (compressor – divergent, turbine – convergent). Gas flow in convergent channel is more robust, which ensures wide range of turbine stable operation. Channels can be designed more curved (with bigger flow turning). That is why the work of turbine stage and pressure drop are higher than of compressor stage. Main parameters of the stage are turbine pressure ratio π^*_{st}, efficiency η^*_{st}, and stage work. L_{st}. The change of radial clearances in the turbine is bigger than in the compressor due to temperature impact. The temperature in turbine causes significant strains which are much bigger than in compressor.

5.2 The classification of gas turbines

1. According to gas movement:
– axial;
– radial.

Only axial turbines are applied in aircraft GTEs. Radial turbines compared to axial are heavier and bigger.

2. According to number of stages:
– single stage;
– multi-stage.

3. According to the number of shafts:
– single shaft;
– two shaft;
– three shaft.

4. According to construction:
a) according to supports position:
– disks are arranged in cantilever,
– discs are arranged between supports (to reduce bending stresses in turbines with big mass);
b) according to rotor structure:
– with undetachable rotor,
– with detachable rotor.

5. According to cooling system:
– uncooled turbines (neither rotating blades nor nozzle vanes are cooled);
– air cooled turbines (rotating blades or/and nozzle vanes are cooled. Cooling is performed by pushing cooling air through the hollow blades and its blowing out to the gas path. Such cooling system is termed open);
– liquid cooled turbines.

Fuel (kerosene) can be used for cooling. The disadvantage of this method of cooling is the necessity for leak tightness, that in its turn increases the mass of the power plant.

Turbine rotor

Turbine rotor consists of shaft, disks and impeller blades (sometimes turbine disks are covered with “cover disks”).

Disks are joined together and to the shaft. These joints can be detachable and non detachable.

Joints operate being acted by the following loads:
– big torque to transmit;
– high temperature, due heat transfers from the blades to the disk;
– vibration loads, which appear because of rotor misbalance and time-variable gas flow pulsing (pulsations act turbine blades);
– bending loads during aircraft maneuvers;
– possibility of parts centering defect, because of their different temperatures.

Rotating blades

Rotating blade has blade body, lock, and often – shroud plate at its tip.

Rotating blades are profiled basing calculation results and equations of gas turbine theory.

There are channels inside blades for cooling purposes. Cooling air passes through them and takes heat from the blade body.

If blades are small and there is no opportunity for the cooling channels, then blades are manufactured without cooling.

Cooled blades are classified:

According to heat exchange:

– convective heat exchange;

– convective-film heat exchange.

According to cooling air movement:

– with longitudinal cooling air movement;

– with transversal cooling air movement;

– with combined (longitudinal-transversal) air movement.

Blade locks

Turbine blade locks are heavily loaded blade parts, as turbine blades (everything else being equal) have bigger mass than compressor ones. They also operate at high temperatures, which result in lower strength thresholds.

“Fir-tree” lock is the most abundant for turbine nowadays.

The construction of the lock: lock has 3-6 teeth pairs with locking plate above. Some locks may have elongated blade shank.

Lock advantages:

1. The material of the blade root part and the disk peripheral part are rationally utilized (blades and disks are the lightest in this case).

2. The root part of the lock is small that allow mounting greater number of blades in the disk.

3. Blades are loosely fitted in the disks, which prevents thermal stresses. These stresses may arise when the blade is heated.

4. Loosely fitted blades can self-center under the centrifugal force. Such fitting provides minimum bending loads.

5. The loose placement allows damping the oscillations. The energy of oscillations is dissipated when overcoming friction in the lock.

6. Damaged blades are very easy to be substituted.

Lock disadvantages:

1. Small contact area causes poor heat withdrawal from blades to disk.

2. Teeth of the lock have small rounding radiuses. Small radius results in stress concentration, and fatigue cracks formation and even tooth break in tooth slot zone.

3. It is necessary to ensure high lock production tolerance, especially by step and angle. Blades are manufactured by casting with further mechanical treatment.

Turbine stator

Turbine stator consists of casing, nozzle vanes and power elements, which join support housing with external casing. Turbine casing is a part of the engine power scheme and an outer wall of the gas path.

Casing is cylindrical or truncated cone structure. Multistage turbine usually has composite casing with only transversal splits.

Casing of the combustion chamber is fastened to the front part of the turbine casing and afterburner casing – to the rear part of the turbine casing (in case of afterburning engine).

Nozzle vanes

Nozzle vanes usually have airfoil part, inner and outer platforms. Gas temperature is maximum at the airfoil part, that is why this part requires to be cooled. Inner and outer platforms limit gas path. Nozzle vanes can be attached from two-sides or cantilevered. Vanes must be attached in a way to provide free thermal expansion, because blades are heated to extremely high temperatures during operation:

– outer platform is rigidly fastened to outer casing, and inner platform is free (with clearance);

– inner platform is rigidly fastened to inner case, and outer platform is free;

– both platforms are attached with clearances.

Nozzle vanes do not comprise the power scheme of the engine. For this reason elements of power scheme are arranged inside the nozzle vanes or housing of turbine support is fastened to outer casing with the profiled struts. In both variants force that appears in bearings is transmitted to outer casing.

Radial clearances

Turbine has rotating and stationary parts, hence it must have clearances to prevent touching. Radial clearances differ from mode to mode, because rotor and facing it casing are heated to different temperatures.

Factors affecting the clearances

1. Temperature strains of turbine casing, blades and disk.
2. Strains of blades and disk, which are caused by centrifugal forces.
3. Strain of parts due to long-term operation.
4. Strains caused by inertia during aircraft maneuvers.

Let`s consider the way radial clearance changes at different engine operating modes. There is an initial clearance **δ** between working blades and the casing of the engine in the work of engine (Fig. 40).

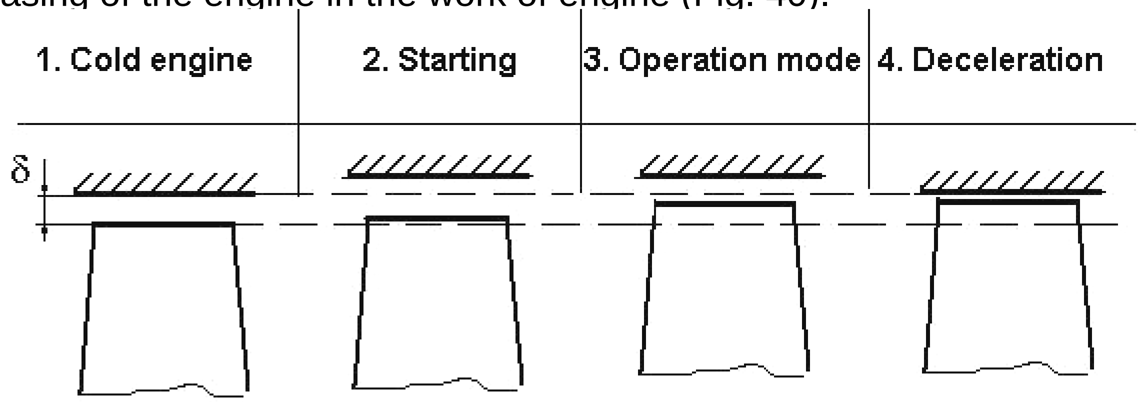


Figure 40 – Radial clearance change

When engine is started casing is heated up quicker than the rotary parts. The clearance between rotating parts and casing increases. When engine has switched up to idle mode, turbine disk is being heated-up reducing clearance up to operating value. After engine was switched off engine rotor decelerates. Casing is cooled quicker than the rotor, hence clearance becomes smaller making working blades touching the casing possible.

All factors affecting radial clearance are very hard to be considered, hence engine casings are equipped with special easily abradable inserts to prevent blades touching the casing. The inserts are facing the blade tips.

1. Metal-ceramic inserts are mounted in trapezoidal grooves, and then are turned to given diameter (Fig. 41,a).

2. Honeycomb seal are manufactured from heat-resistant sheet of 0.3…0.5 mm width and are soldered to the engine external casing (Fig. 41,b).

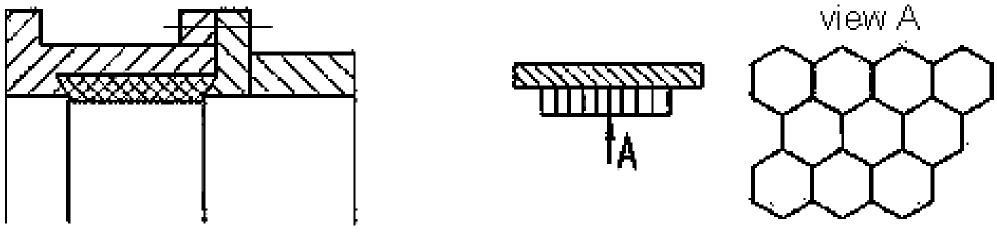


Figure 41- Special easily abradable inserts:
a – metal-ceramic insert; b - honeycomb seal

5.3 Cooling turbine parts

Purposes: Cooling system is applied for ensuring maximum permissible temperatures of the turbine parts. Physical properties of the material at these temperatures are minimally necessary to provide the needed operational reliability and given lifetime.

The benefits of cooling system application are higher temperatures of working substance in the working cycle, higher engine thrust with the same airflow, higher efficiency of working process, usage of cheaper materials with lower portions of nickel and cobalt.

Cooled elements are rotary blades and nozzle vanes, disks and turbine casings. Cooling rotary blades is the most difficult task, because the blades are of complex structure and operate under dynamic loads.

Requirements made to cooling systems:

– high efficiency (temperature must be reduced to needed value at minimum air bleeds for these purposes);

– structural simplicity and high reliability;

Aircraft engines usually have open cooling systems, where air, after it had cooled parts, is blown out to the gas path.

Whence, it follows:

– the necessity of efficient air consuming;

– pressure in the secondary air system must be higher than in the gas path.

Working blades cooling

1. By heat withdrawal to the disk.

Temperature of blades decreases thanks to heat withdrawal to the disk. Disks are cooled by the air. The application range for these cooling scheme is $\mathbf{T_G^* = 1200...1250\ K}$.

2. By blowing the cooling air through the cavities inside the blades.

The example of this cooling scheme is convective cooling. Cooling air passes through specially profiled channels inside the blade and accumulates heat from the internal surfaces of the blade. Then airflow gets to the gas path in the turbine.

3. By joint blowing the cooling air through the cavities inside the blades and preventing hot air getting in contact with hot air from the gas path (by creating the film of cooling air).

Convective cooling of the blade can be divided into groups according to channels arrangement inside the blade:

– with longitudinal channels arrangement;
– with transversal channels arrangement;
– with mixed (longitudinal-transversal) channels arrangement.

Disks cooling

– by blowing on lateral surfaces of the disk (radial blowing);
– by blowing the disk rim (stream blowing);
– by blowing the cooling air through mounting clearances between disk and blade lock;
– combined (e.g. combination of 1 and 3).

In case of radial blowing on lateral surfaces of the disk, air is guided to central disk part and then – to disk lateral surfaces. Deflectors or covering disks are applied to guide air to lateral surfaces.

Advantages: high efficiency.

Disadvantages: big temperature difference between disk rim and central disk part, which results in big temperature stresses.

In case of blowing the disk rim air is supplied through special nozzles to disk rim.

Advantages: low temperature difference.

Disadvantages: low efficiency.

The advantage of blowing the cooling air through mounting clearances between disk and blade lock is high cooling efficiency due to big surface of heat exchange.

For these reasons the latest variant of cooling is usually combined with the first one.

6 COMBUSTION CHAMBERS

6.1 Application purposes of combustion chambers
Operation conditions and requirements to combustion chambers

Combustion chambers are GTE components applied for supplying heat to the working substance. The heat energy appears by combusting the fuel in combustion chamber (chemical energy is transformed to heat). Parameters of the combustion chamber significantly affect the efficiency of the engine as a whole, its reliability and ecological perfectness.

Combustion chamber operates at very hard conditions:

– walls of combustion chamber are in contact with extremely hot gases (gas temperature in the burning core is up to 2300K);

– chamber elements operate at high temperatures (casings – to 600…650 K, flame tubes – to 1000…1200K) and at big nonuniformity temperature field (gas temperature is not uniform and can vary circumferentially up to 75…100 K and radially – up to 40…70 K);

– elements combustion chamber, especially flame tubes and mixers are flown by chemically active gases at high pressures (to 2,5…4,0 MPa) and velocities.

Major sources of non-uniform temperature field are flow division on separate flows by swirl vanes, finite number of fuel spray nozzles and mixers. This results in distortion of chamber elements shape and even blowholes. Blowholes distort gas flow even more leading to overheating, vibrations and structural damages.

Processes in combustion chamber are very complex for theoretical analysis or simulation. So, experimental methods of designing and development come to the fore. Experimental results with further complex and labor-consuming development of manufactured patterns at special test beds are basic methods in designing process.

Except general requirements, such as lower mass and overall sizes, maximal simplicity and reliability, combustion chamber must also meet specific requirements:

– fast and reliable start-up and operational stability of combustion chamber at all operating modes and all flight modes;

– high efficiency of fuel combustion (modern engines have $\eta_{CC} = 0{,}95...0{,}99$) at minimum heating of elements of aircraft and engine and minimal heat losses to the ambience;

– low emissions;

– low hydraulic losses in the chamber (hydraulic efficiency of modern combustion chamber is $\sigma_{CC} = 0{,}95...0{,}97$);

– short flame torch, which will shorten combustion chamber and avoid flame getting to nozzle vanes and rotating blades;

– uniform gas velocity, temperature and pressure field at combustion chamber discharge. Non-uniform temperature field and long flame torch may cause local overheating the vanes and the rotating blades and even their burnout. The non-uniform velocity and pressure field may cause rotating blades vibrations and fatigue damages;

– manufacturing, maintenance and overhaul simplicity.

Stable fuel combustion is ensured by forming the reverse flow zone. The reverse flow zone is a zone where hot gases move in the opposite to air/fuel mixture flow direction. At the same time, air and fuel are mixed, fuel is evaporated and air/fuel mixture is ignited. The reverse flow zone is created by the flame holders. Flame holders form the low pressure zone at combustion chamber entrance. Mixture of hot combustion products and fresh air moves from high pressure zone to low pressure.

High thermal density of working volume is ensured by supplying the precise amount of fuel to ensure excessive air/fuel ratio α close to one (14,8 kg of air are needed to burn 1 kg of kerosene; such mixture is known as stoichiometric). Gas temperature in the primary zone is 1800…2300 K. Preparing air/fuel mixture at such temperature conditions provides better fuel evaporation, intense oxidation. The combustion process becomes more intense, stable and complete. Low hydraulic loses are ensured by proper profiling of combustion chamber walls. The gas path has no sharp flow turns and lips, which prevents unplanned turbulent flows. Combustion chamber shortening is provided by complete fuel combustion in the primary zone (extra fresh air is supplied to provide complete air/fuel mixture combustion in the primary zone) and by special profiling the chamber (e.g. dual-dome combustion chamber). Constructive solutions also determine temperature field uniformity at combustion chamber discharge. The best variant is when temperature is distributed along turbine blade to provide negative gradient of gas temperature to blade root (Fig. 42).

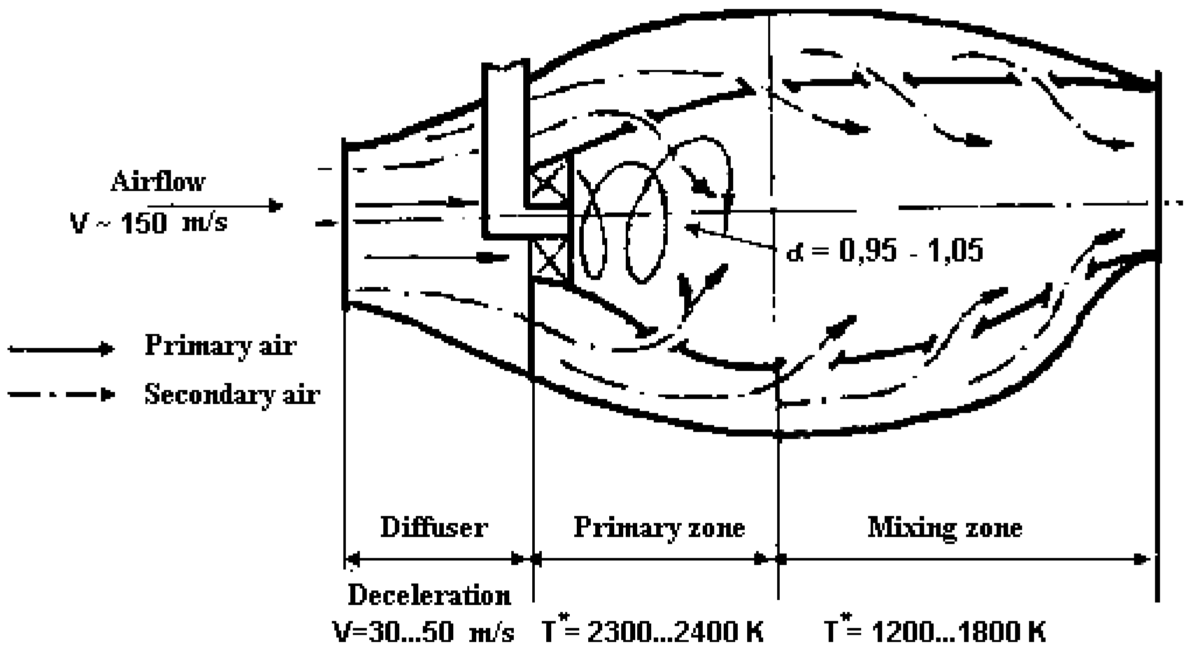


Figure 42 – Air motion in combustion chamber

6.2 The schemes of combustion chambers

Combustion chambers of GTE are classified according to:

a) air/gas flow direction:

– through-flow (D-36, AM-3, R-11 etc.);

– reverse-flow (GTD-350);

– mixed-flow (loop combustion chambers – AI-9, AI-450 and semi-loop GTD-3F);

b) fuel supply to burning zone:

– vaporizing (fuel is supplied in a vapor state – AuxPP TA-6, TShE GTD T-700-GE-700 by General Electric, USA);

– spraying (fuel is supplied in liquid state, means in drops – AL-7, AI-25, etc.);

c) structure and position in the engine:

– tubular or individual – VK-1, Walter M-701(Fig. 43, a);

– tube-annular – R-11, AM-3, D025V, AL-21 (Fig. 43, b);

– annular – AL-7, AI-25, D-36, RD-33 (Fig. 43, c).

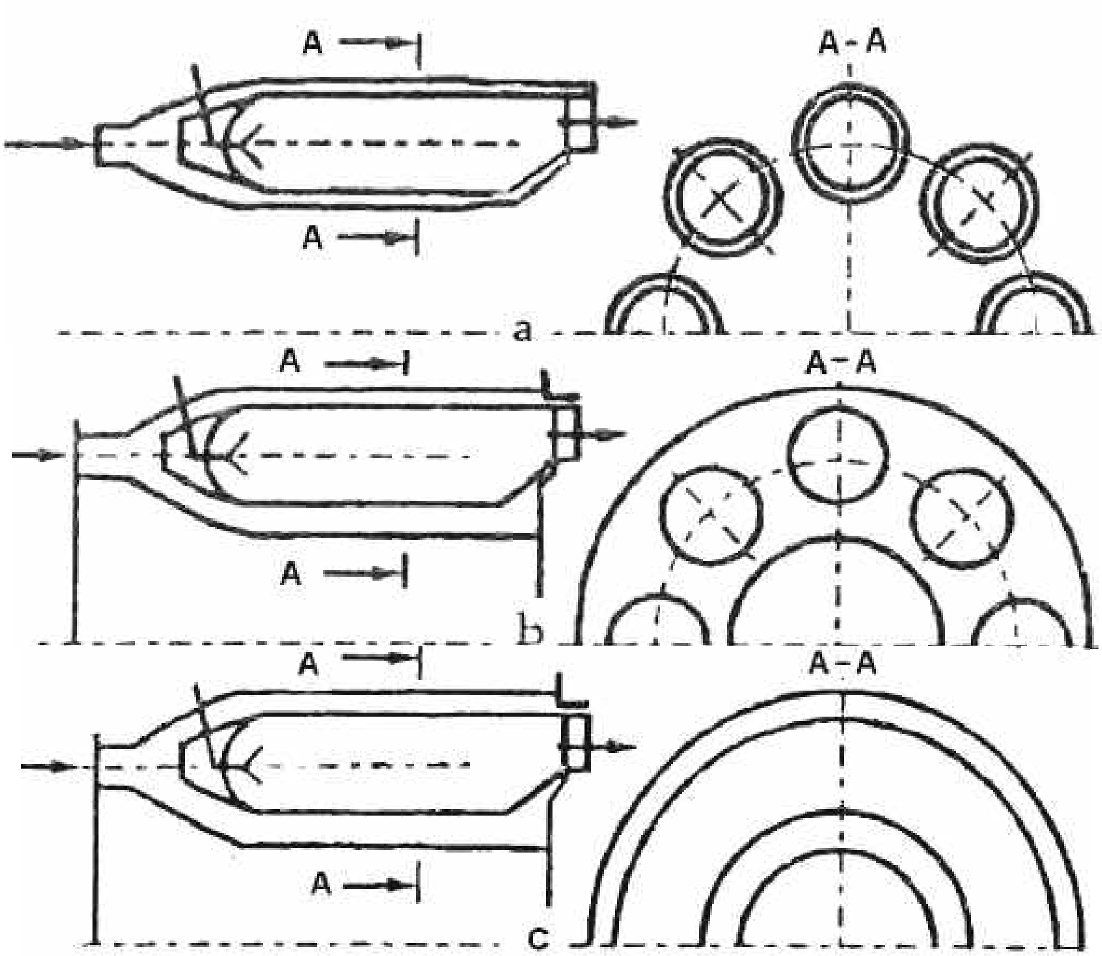


Figure 43 – Schemes of combustion chambers: a – tubular; b - tube-annular; c – annular

Through-flow combustion chambers are the most abundant in modern GTE, because they are of the highest hydraulic perfectness. Their diametric dimensions do not exceed sizes of compressor or turbine.

The disadvantage of through-flow combustion chambers is their big axial size, which in its turn increases engine length and distance between supports and leads to the necessity in more complex engine construction.

Engines with loop and semi-loop combustion chambers have smaller axial sizes, but higher hydraulic losses, comparing to through-flow chambers.

Same to loop and semi-loop combustion chambers, reverse flow combustion chambers (Fig. 44) also significantly decrease distance between rotor supports. They are used in small GTEs or auxiliary power plants, which means that they are expedient in case, when requirements to engine mass and overall sizes are most important, even if it will cause additional hydraulic loses in combustion chamber.

Vaporizing chambers (fuel supply is provided in vapor state) ensure high fuel combustion efficiency and low emissions level. But they are rare to be used because of problems with evaporation system. The evaporation system consists of pipes mounted in primary zone. Fuel evaporation and its partial thermal cracking inside pipes may cause carbonization inside fuel supplying pipes, their overheat and finally burnout. In case of pipe burnout there will form the rich air/fuel mixture (mixture with excess air/fuel ratios $\alpha \approx 0{,}25...0{,}3$) which in its turn may cause engine explosion.

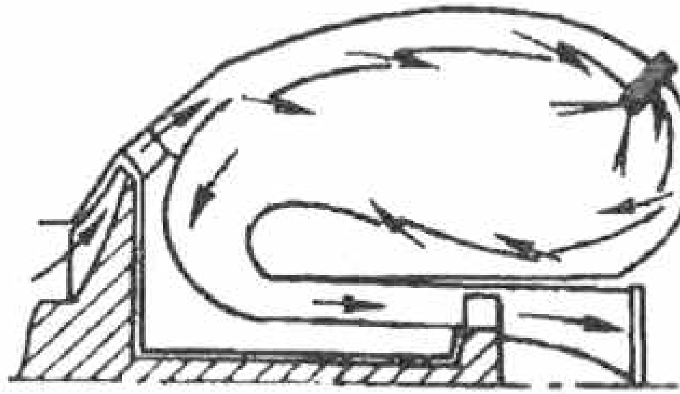


Figure 44 – Reverse flow combustion chamber

Spraying chambers are widely used in modern GTEs. Fuel is supplied streamwise to burning zone in atomized state (drop diameter is 40...100 microns). When combustion chambers were in their infancy and the combustion processes were not studied well enough, fuel was sprayed opposite to main flow to improve fuel atomization and its mixing with air. But fuel spray nozzles operated at extremely high temperatures which caused carbonization of their discharge.

Nowadays (e.g. D-36, D-136, NK-8, NK-22, NK-144 etc.) fuel, leaving centrifugal nozzle is not atomized directly to burning zone, but under a dome. There fuel is intensively mixed with hot air coming from compressor. About 75 % of fuel is vaporized in the under dome cavity. Then air/fuel and vapor/fuel mixture gets to primary zone. Such an approach to fuel supplying improves combustion efficiency and reduces combustion chamber emissions.

Tubular (individual) combustion chamber consists of flame tube, arranged inside individual casing. Number of flame tubes in the engine varies from 6 to 22. These tubes are equally spaced in circle, parallel or inclined to the axis. Flame tubes are joined together with interconnectors. They flip flame from pilot igniters or adjacent operating chambers to the rest chambers. These combustion chambers suite the most for joint operation with centrifugal compressors. Small volume of the single chamber eases its development during engine designing. Individual chambers provide high engine vitality and are very handle at overhaul.

The disadvantages of tubular combustion chamber are big mass (to 12…15 % from engine mass), necessity in mounting the nozzle box, big number of joints, all of which need to be leak-proof, increased hydraulic resistance. Moreover, tubular combustion chambers cannot be included to engine power scheme, which makes designers use extra structural elements in the structure of the casing.

Annular combustion chamber structurally consists of single annular flame tube, internal and external casings, which are usually included to engine power structure. These chambers are very compact and do not increase engine diametrical dimensions. The volume of the chamber can be used in the most efficient way. Their mass is less than 6…8 % from engine mass.

Gas temperature, velocity and pressure fields at annular combustion chamber discharge are the most uniform. Annular chamber has low hydraulic losses, and less problems with sealing the link up points.

The disadvantages of annular combustion chambers are:

– problems in the development of the big chambers (high mass flow rates at high pressures);

– problems to provide stable combustion, rigidity and strength;

– these chambers are complex to be examined and repaired in maintenance.

Tube-annular combustion chambers unite some advantages of tubular and annular chambers. Structurally, they consist of separate flame tubes (from 9 to 14) placed inside annular cavity, formed by internal and external chamber casings. Internal and external chamber casings are usually included to engine power structure. The diametrical size of the external casing usually does not exceed compressor or turbine.

These chambers take the intermediate position (mass, characteristics, complexity in development, maintenance and overhaul) between annular and tubular chambers.

6.3 Structural elements of combustion chamber

Regardless of combustion chamber scheme, the next structural elements are common for all:

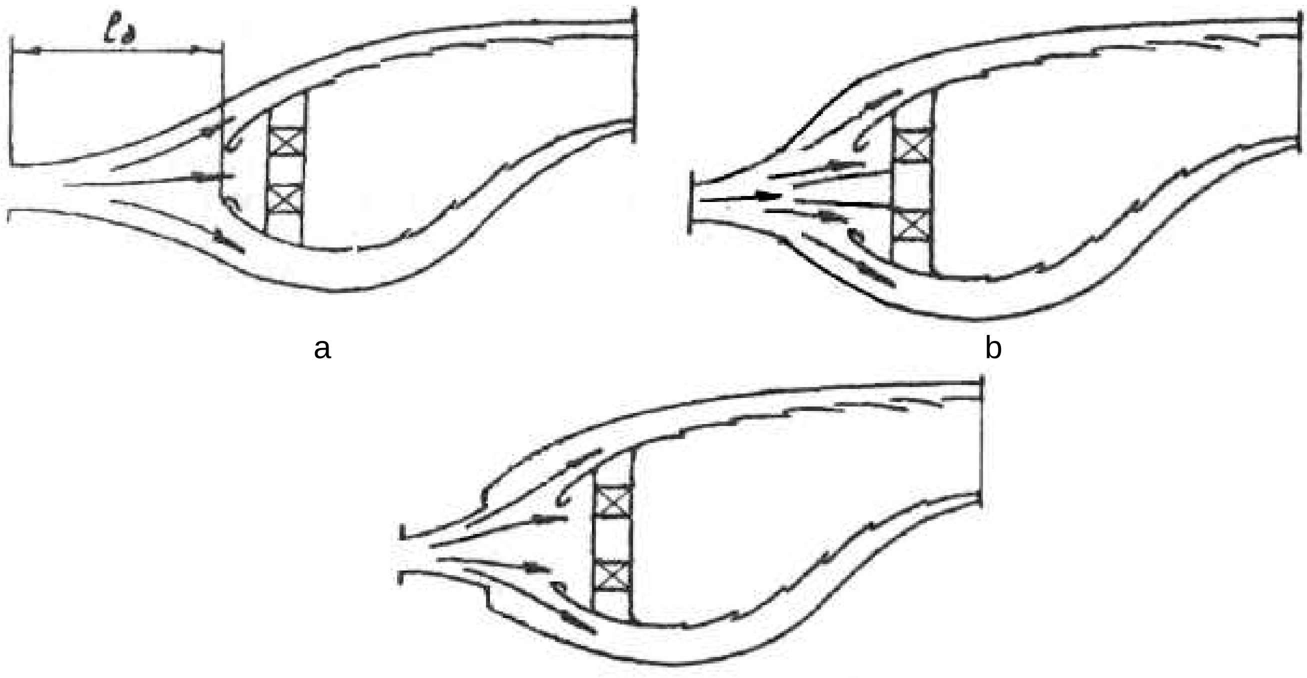
– diffuser;
– flame tube;
– flame-holders (swirl vanes);
– combustor dome;
– pilot igniters;
– drain valves;
– fuel manifolds with fuel nozzles;
– interconnectors (tubular or tube-annular combustion chambers only);
– nozzle boxes (tubular or tube-annular combustion chambers).

Diffuser is a combustion chamber component in its front part. Diffuser decelerates the airflow from velocities $120...180\ \frac{m}{s}$ (at diffuser inlet) to $30...50\ \frac{m}{s}$ (at diffuser discharge) before it gets to primary zone. The velocity must be decreased for stable fuel combustion. Hydraulic loses in diffuser constitute the main part of hydraulic loses in combustion chamber, that is why, their profiling is paid special attention. All diffusers may be classified into smooth-contoured, with flow splitting or dump.

Smooth-countered diffuser (Fig. 45, a) is a smooth duct with apex angle 18…25º. Smooth-contoured diffuser is the most hydraulically efficient due to smooth air flowing inside the duct. But the drawback of this diffuser is its big axial size, which increases the distance between rotor supports and engine length.

To decrease axial size of the diffuser, it can be constructed to be ended by the abrupt increase in cross-sectional area of the duct. This diffuser is termed dump (AL-21, TV3-117, R-29). There are special collars in the place of abrupt expansion. They catalyze flow disruption (Fig. 45, c).

In some instances, there may appear a strong need in diffuser with big apex angle (to 30…45º). To ensure smooth flow in diffuser, it must be divided in two or three ducts (Fig. 45, b) with small apex angles (AI-25).



a b

c

Figure 45 – Diffusers: a - smooth-contoured diffuser; b – with flow splitting into three ducts diffuser; c – dump (with abrupt increase of cross-sectional area) diffuser

Flame tube forms the volume for air/fuel mixture combustion. Modern chambers are manufactured by thing-wall rings rolling and welding, which decreases temperature stresses in the construction. The walls of the flame tube are cooled by the secondary air from the outside of the flame tube. Air forms the film of cool air flowing close to flame tube walls and prevents the walls from being in contact with hot gas inside the tube.

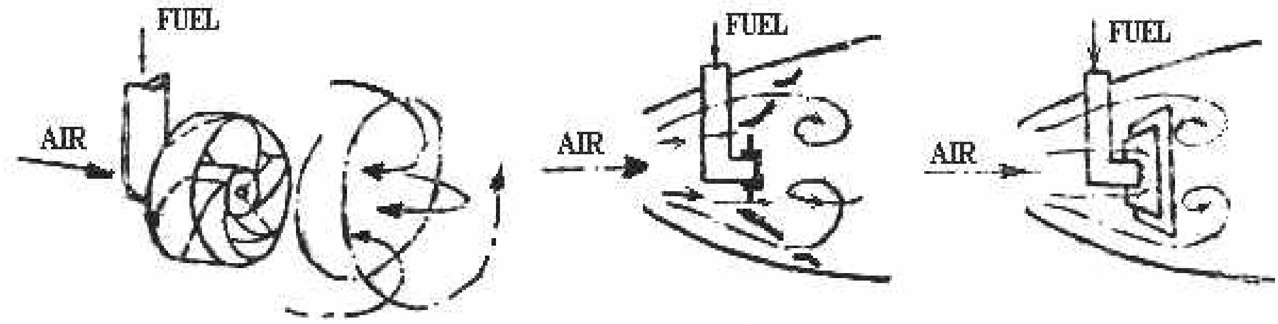
Fastening of flame tube must provide its free thermal strains. Flame tube is fastened same to two-support beam: in one cross-section –rigidly fixed, and in the other one – it has freedom of displacements.

Flame-holders (swirl vanes) provide stable air/fuel mixture combustion by creating the reverse flows zone, intensifying mixing processes and increasing the turbulence of the flow. All existing combustor domes may be classified into:

- bladed (swirl vanes) – R-11 (Fig. 46,a);
- «grater» type (jet) – D-25V, D-30 (Fig. 46,b);
- conical (based on effect of a flow stall) – AI-20, AI-25 (Fig. 46,c);
- combination of the first three.

Mixers supply secondary air inside the flame tube for gas temperature reduction to given value before delivering it to the turbine. Secondary air is supplied gradually through holes or slotted brunch pipes of variable section. Cooling air must be avoided from getting in the reverse flow zone and violating fuel burning process because of local gases cooling. Secondary air streams must get deep in the core of the primary hot flow to reduce gas temperature not only close to walls, but in the flow core.

During engine starting, not only rotor acceleration, but reliable air/fuel mixture ignition must be provided. Mixture ignition depends on many factors, main of which are fuel type, the quality of air/fuel mixture preparation, power of igniters and their position in the combustion chamber and operation time.



a b c

Figure 46 – Flame-holders:
a - bladed (swirl vanes); b - «grater» type (jet); c - conical

Pilot igniters provide initial air/fuel mixture ignition during engine starting. The air/fuel mixture can be ignited by an electric plug (mostly the engines, which operate at low altitudes (D-25V, TV3-117) or the engines with small combustion chambers (RD-33)), or electric plug combined with pilot fuel injector (AL-7, R-11). Low voltage (1500…2500V) igniter plugs the most used (semiconducting, surface discharge igniter plugs). Cooling the pilot igniter is capacitive at starting mode, by heating of its own mass. Oxygen can be supplied from onboard oxygen tanks to igniter (R-25) to facilitate flight relight or engine starting in winter. Combustion chamber starting is done by the ignition system, which consists of ignition device, electric plugs, solenoid valves, and starting fuel nozzles. Plug and starting fuel nozzle are usually placed in common casing. Such unit is intended for air/fuel mixture initial ignition during engine starting and is termed starting igniter. Their number in the engine depends on combustion chamber construction, its sizes and engine application purposes. Generally, engines have two igniters, but sometimes, in big combustion chambers, their number can be increased to five.

Starting igniter consists of casted casing, screen-deflector guiding air to starting nozzle, spherical bush with igniter nozzle rolled inside and flame tube delivering flame in the combustion chamber (Fig. 47). Air gets in starting igniter from combustion chamber through holes and is twisted by the deflector.

Interconnectors distribute flame from one flame tube to another in tubular and tube-annular combustion chambers and equalize pressure in combustor domes.

Some combustion chambers do not have starting igniters as separate units, but only electric plugs.

Nozzle box is aimed to provide smooth transfer from circular cross-section (discharge of tubular or tube-annular combustion chamber) to annular cross-section (nozzle vanes inlet).

Nowadays, modernization of combustion chambers pursues the next goals:

– shortening the combustion chambers;
– reducing the emissions;
– lowering fuming level.

As it was mentioned above, one of the problems in designing combustion chambers is reducing the fuming level. So the most prospective method to reduce fuming level is to improve the construction of the chamber in order to prevent zones with rich mixtures. Such improvement is substantial for the primary zone. To prevent fuming it is necessary to supply additional air through special apertures and pipes to the core flow in the primary zone. Perhaps, constructions with preliminary fuel evaporation (D-36, D-136) and air-blast atomizer (R-29) can be used.

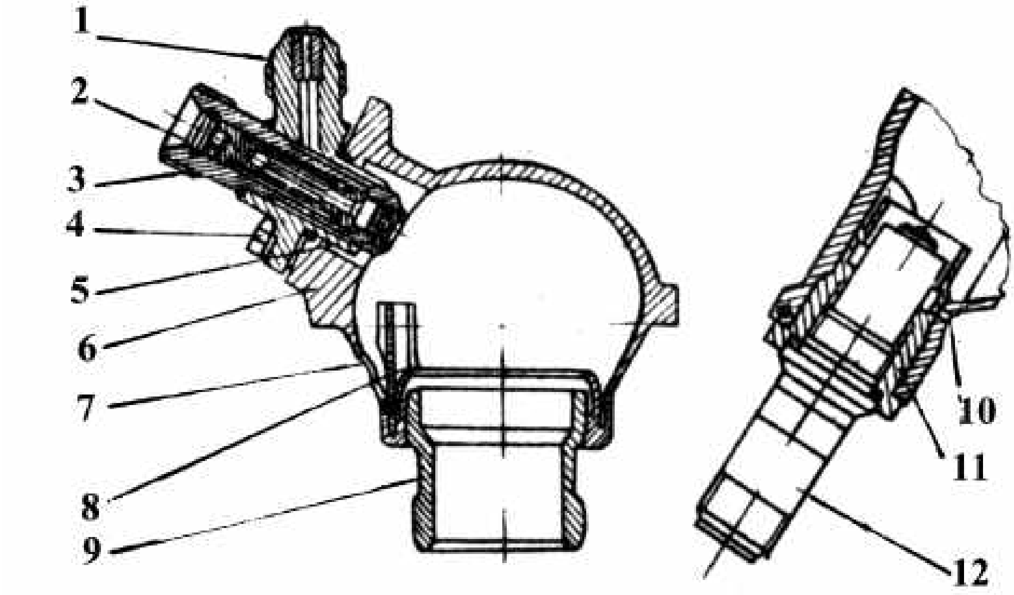


Figure 47 – Starting igniter:
1 – adapter with metering oxygen nozzle; 2 – fuel filter; 3 – starting nozzle;
4 – bolt for nozzle attachment; 5 – atomizer; 6 – casing; 7 – aperture for air supply;
8 – screen-deflector; 9 – the nozzle for the igniter; 10 – aperture for cooling air; 11 – bush;
12 – ignition plug

Nowadays, low fuming chambers are technically mastered, but chamber development, aimed fuming level lowering, should not make other technical characteristics (ignition reliability at starting, combustion stability, high efficiency, smooth temperature field, lifetime, etc.) worse.

Combustion chamber can be shortened by improving the mixing process (e.g. supplying the preliminary mixed/vaporized fuel or implementing the dual-dome combustion chambers).

Dual dome construction lets one-third times annular combustion chamber shortening, comparing to single-dome annular. Dual dome diffusers are usually used jointly with diffusers that have small apex angles. Air enters flame tube through numerous apertures at high velocity that is advantageous for air and fuel mixing in primary zone.

7 AFTERBURNING

To break the sonic barrier and continue the flight at supersonic velocity, to shorten the takeoff run distance and acceleration time, to improve the rate to climb, supersonic turbojet engines of jet airplanes should provide significant engine thrust augmentation (to 45…60 % during takeoff, and to 130…170 % for supersonic flight). Thrust augmentation is obtained by reheating the working substance in the specially added engine component termed afterburning combustion chamber (afterburner).

Afterburner is the most profitable way to increase the engine thrust, because moderate increase in weight and overall sizes lets vast augmentation of the specific thrust. Afterburning mode is fuel inefficient, but increase in specific fuel consumption is compensated by the improvement of airplane operation parameters.

7.1 Afterburner application purposes

Afterburner is mounted after last turbine stage and is used for extra fuel burning. Supplied energy increases gas enthalpy and correspondingly – gas velocity at nozzle discharge and engine thrust.

Mixture consisting of combustion products from the turbine, air from engine bypass duct and air from turbine cooling system, blown back to gas path in the turbine, is guided to afterburner inlet. Mixture in afterburner is close to stoichiometric (excess air/fuel ratio equals to $\alpha_{\Sigma}=1,1...1,3$), that is why temperature in burning zone is high (to 2050…2200 K).

Structural elements of the afterburner must:

– be structurally suitable for operation at high temperature conditions in chemically active gases environment;

– provide stable fuel combustion at all operating modes of an aircraft (needed operating range according to mixture composition is $\alpha_{\Sigma\,min}=0,7...0,9$ to $\alpha_{\Sigma\,max}=2,0...2,5$);

– ensure reliable chamber starting in the whole range of flight altitudes and velocities, which are permitted to turn on afterburner;

– make no effect on gas generator operation;

– prevent turbine blades from being overheated while afterburning turning on and off, and during its operation;

– create minimum hydraulic and heat loses;

– minimum mass.

7.2 The construction of afterburner main elements

Afterburner consists of combustor dome and combustor itself. Combustor in its turn consists of diffuser, flame stabilizers, fuel supplying system and ignition units (Fig. 48).

Diffuser decelerates the gas flow to provide conditions suitable for stable afterburner fuel combustion. It is mounted right after turbine discharge. Diffuser discharge area and its length are designed to ensure flow deceleration from $V=300...400\frac{m}{s}$ at diffuser inlet to $V=250...200\frac{m}{s}$ at its discharge.

The construction of the afterburner must be optimized to provide minimum hydraulic losses at minimum overall sizes and mass. Optimal apex angle is 8…12º. The optimal ratio between discharge cross-section area (F_2) to inlet one (F_1) must lay in the range between $\frac{F_2}{F_1}=1,3...2,3$. Diffuser annular duct is formed by turbine cowl and outer casing, joined together by struts or articulated rods (Fig. 49, 50). Joint must provide free temperature strains. Diffuser walls are profiled in the way to ensure minimum hydraulic loses at minimal length. Sometimes, internal wall is truncated (dump diffuser).

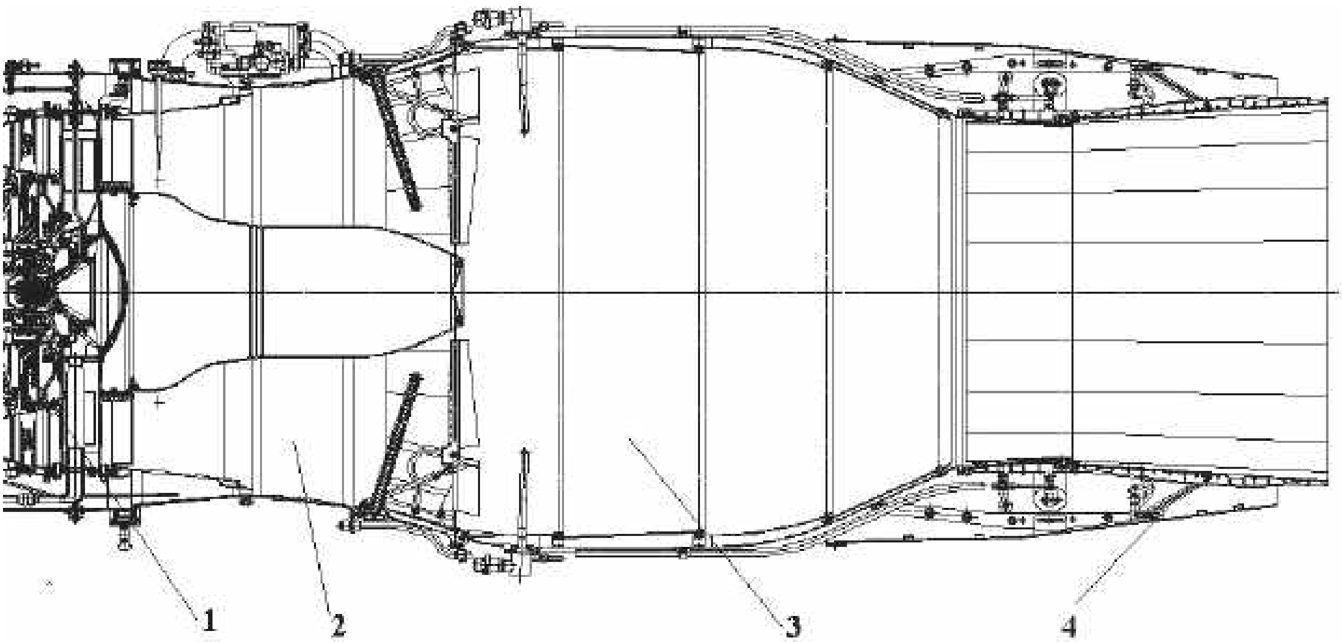


Figure 48 – ATFE afterburner and controllable nozzle:
1 – turbine; 2 – combustor dome; 3 – afterburner; 4 – adjustable nozzle

The velocity of flame distribution in turbulent flow is $10...15\frac{m}{s}$, and flow velocity at diffuser discharge, as it was mentioned above, is $150...200\frac{m}{s}$. That is why, stable fuel combustion is impossible without special devices, termed flame stabilizers. They provide invariable flame front position in afterburner, preventing it from being blown away by gas flow. Bluff body shape stabilizers are the most widely used. Usually they are V-shape chutes manufactured from sheet material with apex angle $30°...60°$, turned streamwise.

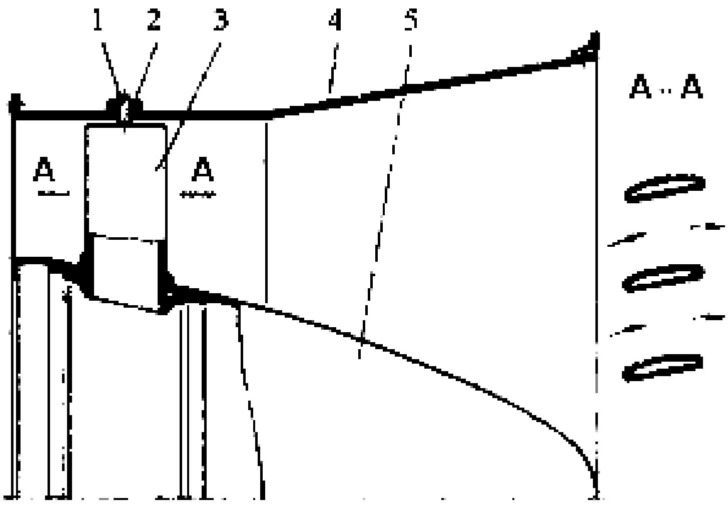


Figure 49 – Diffuser with turbine cowl fastening by struts: A-A – straightening airfoils, formed by struts; 1 – pin; 2 – spherical bush; 3 – strut; 4 – casing; 5 – cowl

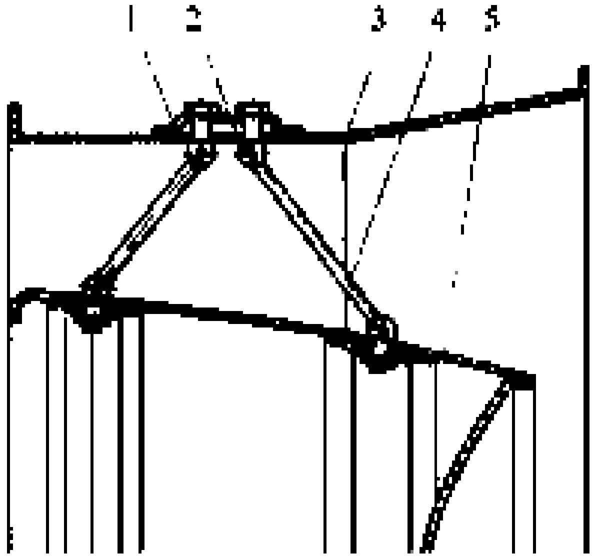


Figure 50 – Diffuser with turbine cowl fastening by articulated rods: 1 – shroud; 2 - rod fastening hinge; 3 – external casing; 4 – rod; 5 – turbine cowl (the internal wall of the duct)

Stabilizers form the reverse flow zone downstream. Combustion products at temperature 1500...2000°C circulate here. Hot combustion products circulating in the reverse flow zone are the heat source for continuous ignition of new air/fuel mixture portions supplied to afterburner. Stabilizer walls are cooled by incident flow of more cold air and by afterburning fuel.

Flame stabilizers can be annular, radial, and combined (radially-annular).

Sometimes, shape of stabilizers is determined not only by engine construction or its afterburner features, but also by the necessity to suppress dangerous vibratory combustion.

Stabilizers significantly obstruct gas-path (to 20…25 % of gas path area). To reduce the hydraulic losses, stabilizers are echeloned and shifted relative to each other upstream or downstream (Fig. 52). Some engines have easy faired stabilizers (niche and aerodynamic) to improve engine characteristics at modes without afterburning (Fig. 51).

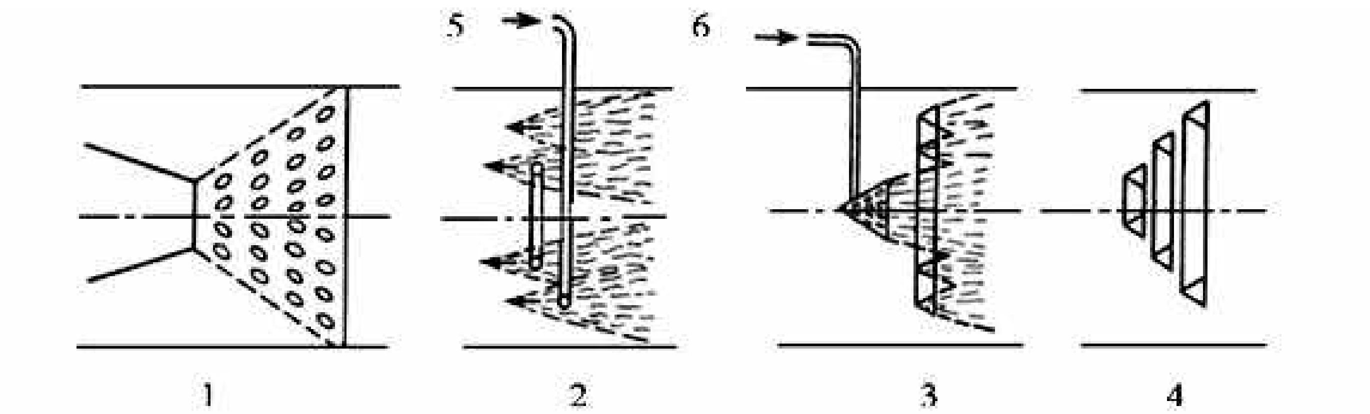


Figure 51 – Flame stabilizers:
1 – “basket” type stabilizer; 2 – aerodynamic stabilizer; 3 – stabilizer with “orderly” flame source; 4 – annular V-shape stabilizer; 5 – air supply; 6 – fuel supply to ignition source

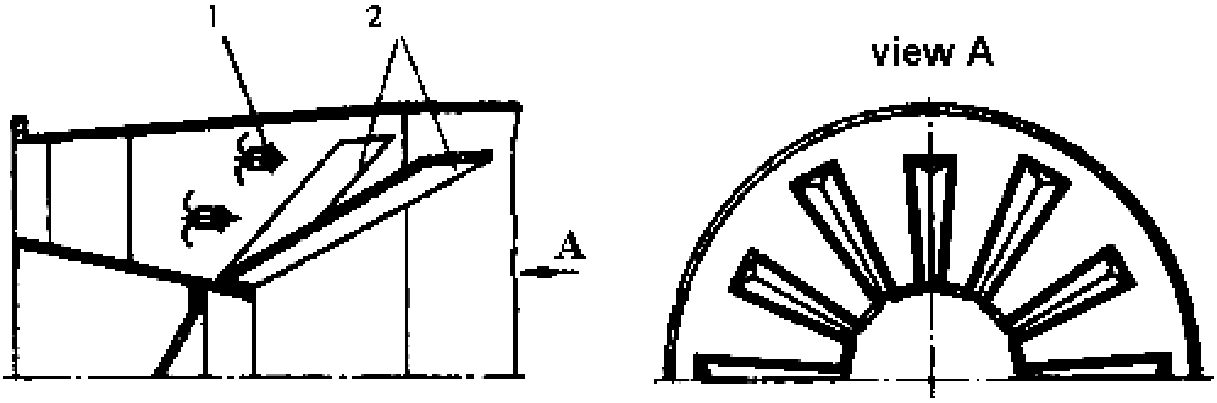


Figure 52 – Combustor dome with radial flame stabilizers:
1 – fuel manifolds with fuel nozzles; 2 – radial flame stabilizers

Generally, stable fuel ignition and its combustion can be obtained by simple stabilizers. But if fuel is hard to be ignited, then pre-burner may be applied (Fig. 53). Pre-burner is a miniature combustion chamber, where airflow is delivered at low velocity (about $10...20\frac{m}{s}$). Hot gases from pre-burner are guided to zones, where ignition and fuel combustion is hampered.

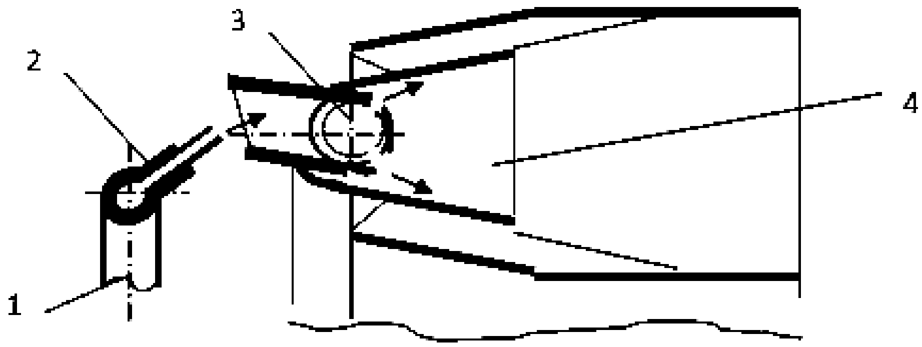


Figure 53 – Flame stabilizer with pre-burner:
1 – fuel manifold; 2 – fuel nozzle; 3 – carburetor; 4 – pre-burner

Fuel supply system and carburetion system provide fuel supplying to afterburner, its atomization, partial evaporation and its mixing with the gas flow. These systems must also ensure needed fuel distribution between stabilizers and in the cross-section of the afterburner. Carburetion system consists of fuel supplying pipelines and manifolds, nozzles for fuel atomization (atomizers) and arrangements for fuel evaporation (carburetors). Liquid fuel is atomized by centrifugal or spraying nozzles, which are welded to fuel manifolds. Centrifugal nozzles are mounted in opposite to gas flow motion, and spray nozzles – across the flow or at some angle to the flow. Final atomization on little drops happens under gas flow action.

Fuel manifolds with fuel nozzles are mounted upstream stabilizers at the distance $100...150$ mm. They are fastened in the way to provide free temperature strains(Fig. 54) .

If fuel nozzles are mounted as it was considered, then the sprayed fuel and gas flow are mixed at the highest efficiency, the vast majority of the liquid drops evaporate before they get to stabilizer trailing edge, the flame torch has the maximum cross-sectional size. Unevaporated drops form liquid film on the surface of stabilizer. By traveling to stabilizer trailing edges fuel drops cool the stabilizer surface and, flowing down from the trailing edges, enrich the reverse flow zone. These arrangements broaden the stable combustion range of the afterburner. Similar to main combustion chamber, afterburner is complex for the analytical analysis, hence sizes of stabilizers and their relative positions, number of fuel nozzles and their positions are experimentally determined during afterburner development. Nowadays application of special units termed carburetors, which are mounted inside the stabilizer to evaporate the liquid fuel, became common. They are designed to provide stable afterburner operation on the lean mixtures ($\alpha_{\Sigma} \geq 3$).

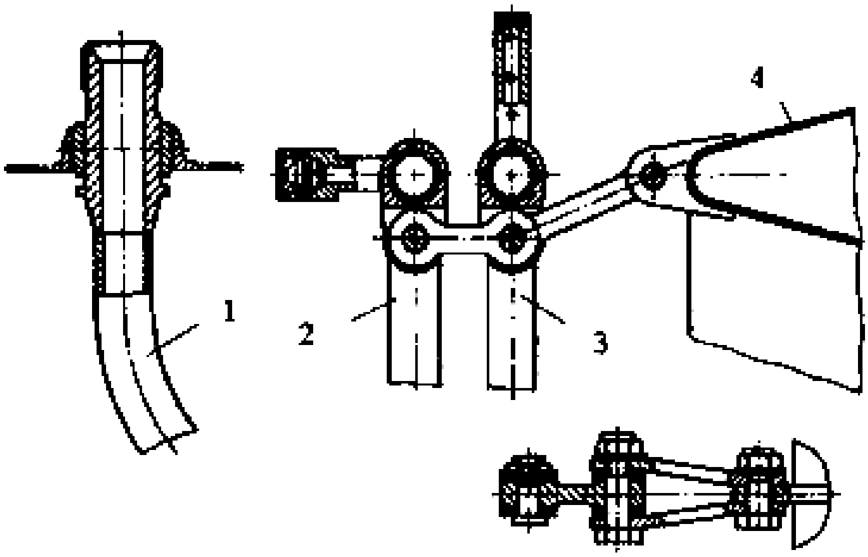


Figure 54 – Flame stabilizer with two fuel manifolds:
1 – pipe for fuel supplying to manifold; 2 – 1st manifold; 3 – 2nd manifold;
4 – flame stabilizers

Air/fuel mixture is ignited by ignition device that is coaxial to combustor dome, or by special pilot igniter, which is fastened to external casing of the combustor dome (Fig. 55).

Fuel is ignited by the electric plug inside the ignition device. To simplify ignition, air from compressor discharge or oxygen from onboard balloons is supplied to ignition device.

Pilot torch from ignition device is directed through special branch pipe to zone with the most intense turbulence. Here, the energy needed for the ignition is minimal. Modern ATFEs are equipped with simple system, which is able to form a high thermal power torch. The system is known as “fire path”. Pilot spray nozzle is mounted at main combustion chamber discharge facing the turbine. When passing through the turbine, pilot fuel is evaporated and next ignited being acted by gases at high temperature. This forms the powerful flame torch inside and after the turbine (Fig. 56). To initiate the combustion process, “fire path” is turned on for a short period of time (usually 0,2…0,5 s) to avoid great heat influence on turbine blades.

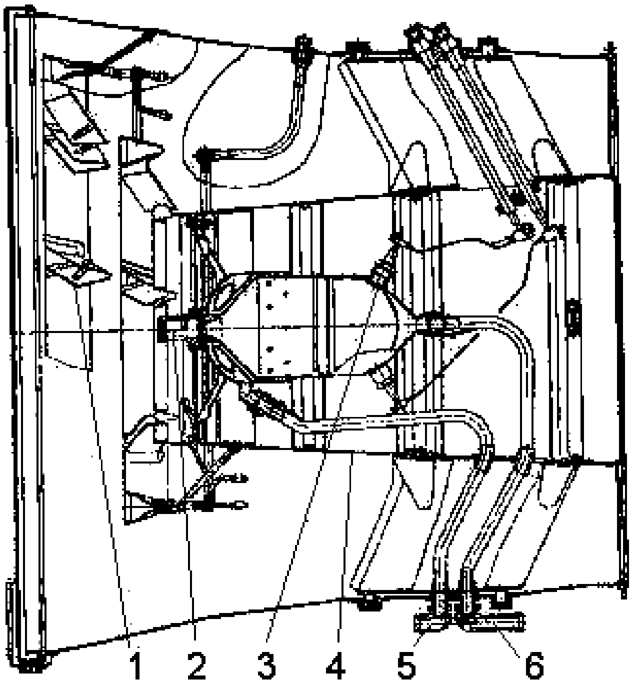


Figure 55 – Ignition device:
1 – flame stabilizers; 2 – flame dissector; 3 – electric plug; 4 – turbine cowl;
5 – air supplying pipeline; 6 – carbureted air/fuel mixture supplying pipeline

Sometimes there appears a strong need to start afterburner at low operating modes (modes with low gas temperature at afterburner inlet). Hence, catalytic ignition devices or nozzles with starting fuel ultrasonic atomization must be used to start the afterburning.

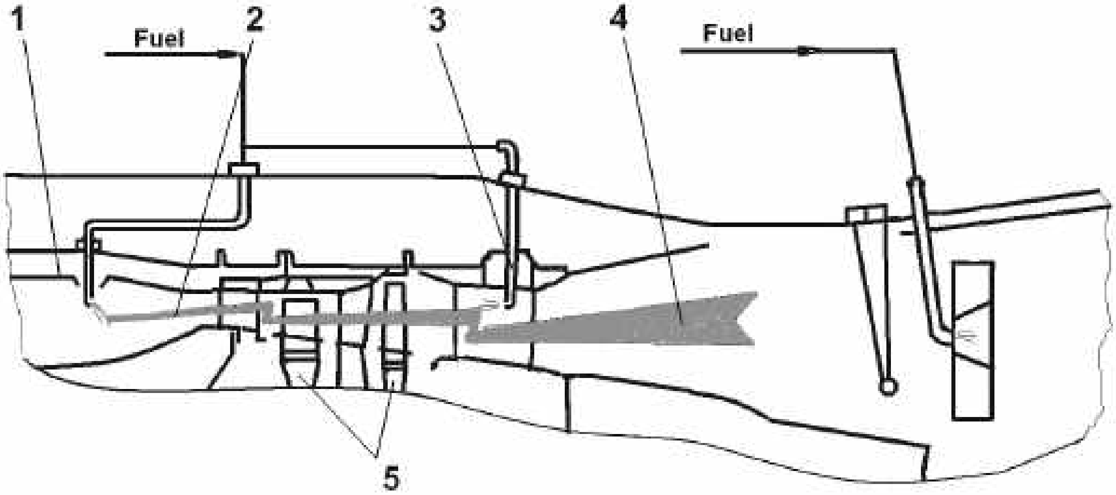


Figure 56 – The "fire path":
1 – flame tube of main combustion chamber; 2 – fuel trickle; 3 – pilot fuel nozzle;
4 – flame torch; 5 – turbine stages

Combustor itself is a cylindrical, conical or spherical shell, welded from heat-resistant sheet material. Combustor itself starts from stabilizers discharge and ends by jet nozzle fastening flange. Sizes of the combustor itself are chosen to provide most full fuel combustion.

To increase static and vibration strength of the structure, the combustor walls are welded spirally. Combustor is fastened to combustor dome telescopically or via flange-bolt coupling in the way to ensure free temperature strains in case they are non-uniformly heated. The external surface of combustor wall is cooled by air, which passes between combustor wall and the casing.

During afterburner operation, there can appear an off-design mode, when gas oscillates with pressure amplitude $\Delta p \leq 0{,}05 MPa$ and frequencies $f = 50...5000 Hz$. This mode is termed “resonant burning”. At this mode there appear longitudinal and transversal (radial and tangential) acoustic oscillations in the combustor. The presence of resonant burning is detected by the relevant "squeal" and quick destruction of afterburner elements. To suppress the high-frequency oscillations designers equip the afterburner with an anti-vibration screen. It is mounted from inside the combustor along its wall. It looks like corrugated punched structure (Fig. 57).

Screen is acoustic-resonant absorber that suppresses oscillations of definite frequencies. If oscillation frequencies of resonant burning are low, then screen efficiency is also low. Hence the suppression of low-frequency oscillations is performed experimentally during engine development. To prevent resonant burning the following arrangements can be done: change of afterburner acoustic volume (by changing the shape of turbine cowl and its sizes), displacement of the maximum heat release zone, variation of the fuel distribution, variation of stabilizers form and echelon positions, changing the gas flow velocity, etc.

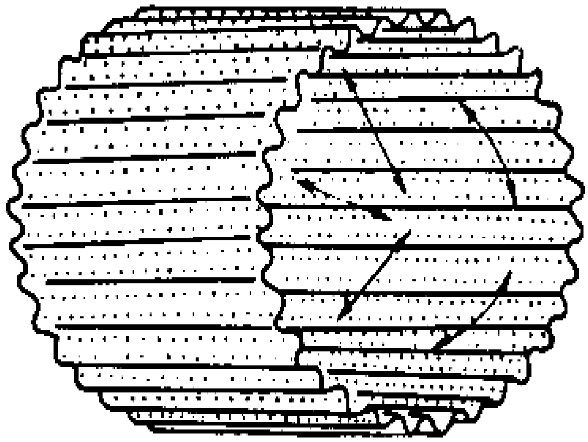


Figure 57 – Anti-vibration screen

The internal surface of combustor wall is cooled by gas from the turbine of, in case of ATFE – by the secondary flow. At the same time, screen operates as a heat insulator. It reduces the wall heating by radiation and provides convective and film cooling.

Screen must be mounted inside the casing to provide the compensation of heat expansion of its elements relative to the casing. Radial compensation is provided by longitudinal corrugations, which are deformed within the material elastic limits. Axial compensation is provided by oval holes in the screen for bolt connection or by telescopic leaning on the adjacent screen section.

8 EXHAUST SYSTEMS

Exhaust system plays some roles that are very important for engine proper operation. Exhaust system:

– transforms the energy of gas into the energy of jet thrust;
– provides the thrust vectoring;
– assists in keeping the corresponding operating mode of gas generator;
– transports gas in the fuselage or the nacelle;

– reduces noise level of the power plant as a whole;
– screens the direct infrared radiation of the engine, etc.

When designing a new engine, one always faces some challenges, associated with designing and development of the engine exhaust system, its coordination with aircraft.

8.1 The construction of exhaust system elements

The simplest TJE exhaust system consists of turbine cowl and fixed-geometry convergent duct. Turbine cowl prevents abrupt flow expansion and vortex formation at turbine discharge and also protects turbine disk from being heated by hot gases (Fig. 58).

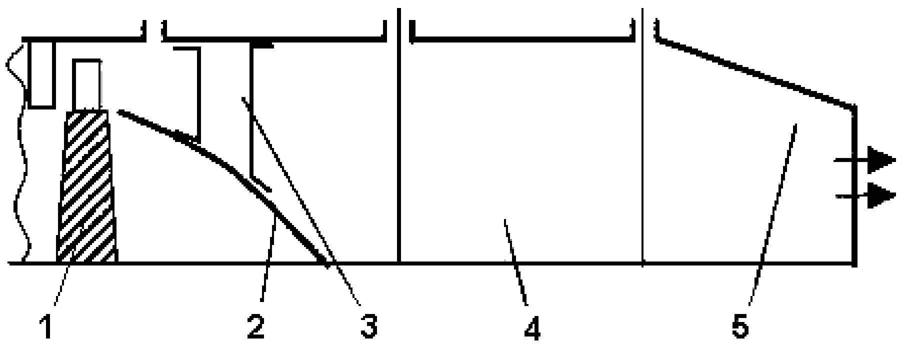


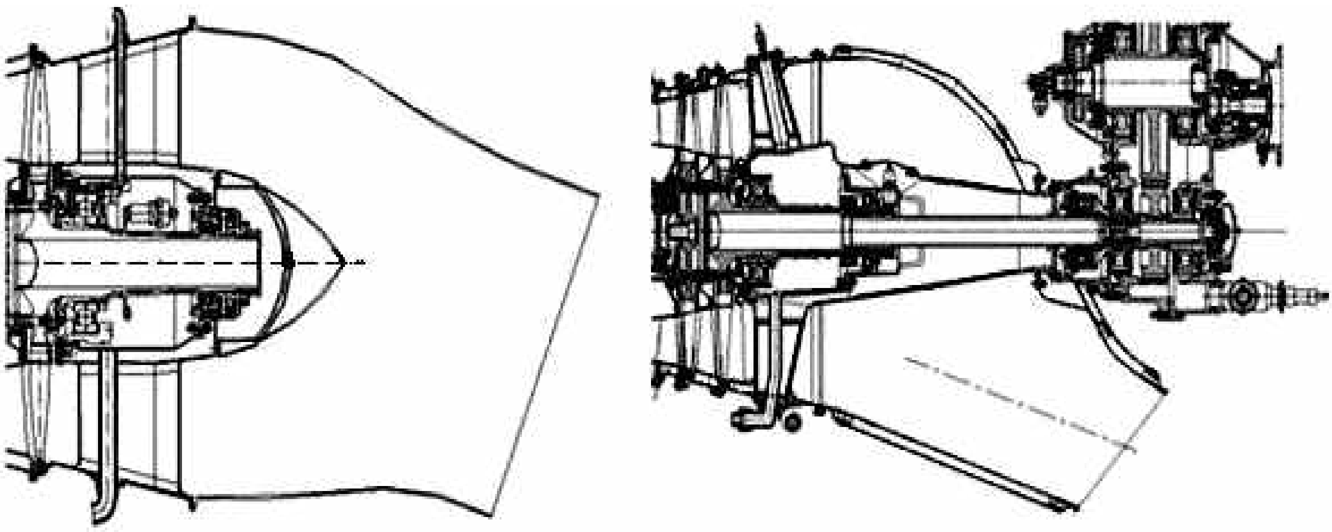
Figure 58 –Fixed exhaust nozzle:
1 –disk; 2 – turbine cowl; 3 – strut; 4 – lengthening part of the casing; 5 – nozzle

The apex angle of the turbine cowl is 35…50°. This cowl is joined to external casing by radial struts or rods, coated by domes to prevent their overheat. The struts are cooled by air. Their construction provides free thermal expansion of adjacent elements relative to each other. If nozzle is profiled to provide the off-axis discharge, then domes are twisted so to provide gas flow straightening at minimum hydraulic losses in the rest parts of the exhaust system. In some instances that are mostly caused by the engine position on the aircraft there appears a need in the lengthening part of the casing. It is welded from heat resistant sheet steel. The diameter of lengthening part of the casing is chosen to provide gas flow velocity lower than $\mathbf{V = 150...200\ \frac{m}{s}}$. This velocity range is the most hydraulically efficient. Lengthening casing must be fastened to make small displacements of one part relative to another possible. Such fastening compensates the production inaccuracies and also reduces the stresses that appear when fuselage and engine nacelle are deformed.

Lengthening part of the casing leans on the engine nacelle by its rear mounting belt. Rollers with eccentrics move through guiding grooves and compensate the strains. Lengthening part of the casing is coated with the heat insulator and the screen to prevent the construction elements of the glider from being heated by hot gases. Lengthening part of the casing is also blown over by the cooling air.

The major part of the TPE thrust is provided by the propeller, and jet component constitutes only 5…15 %. That is why gas flow parameters are not significantly changed in the exhaust system. So, gas is exhausted without being significantly expanded. Velocity of the jet stream at nozzle discharge is significantly lower comparing to TJEs or TFEs.

That is why in case of the need, gas can be exhausted at some small angle to airplane trajectory (to 20º) (Fig. 59).



a b

Figure 59 – Exhaust systems:
a – single-shaft TPE; b – TPE with free turbine

The exhaust systems of TShEs guide gas the flow aside (Fig. 60).

Free turbine of TShE extracts maximum amount of work from gas flow making gas pressure at free turbine discharge lower than atmospheric. So, TShEs have divergent duct instead of commonly used convergent. Gas stream leaving the divergent nozzle produces no jet thrust.

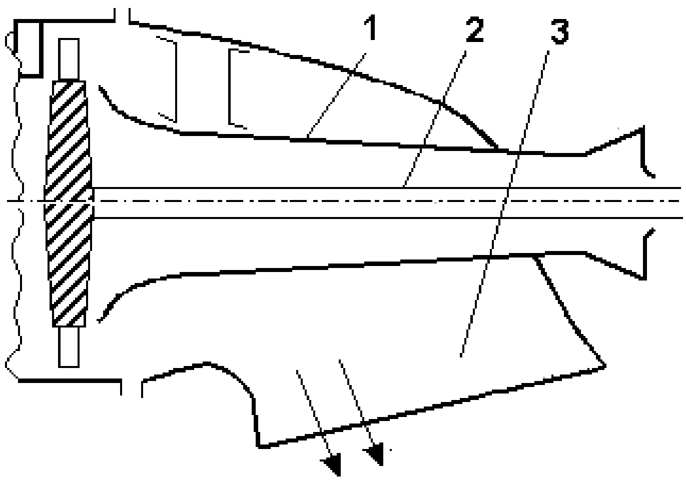


Figure 60 – The exhaust system of TShE:
1 – shaft coating; 2 – shaft of free turbine; 3 – divergent nozzle

Mixer is an turbofan component where primary and secondary flows are mixed. Mixer is arranged after the turbine.

Mixing device, which is mounted at mixer inlet, divides air and gas flows into small separate streams. Small streams can be mixed over a smaller region than big ones (Fig. 61).

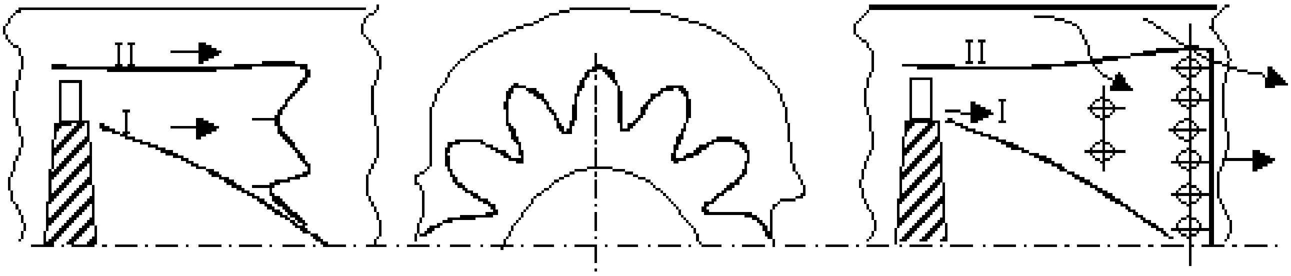


Figure 61 – Mixer

8.2 Jet nozzles of TJE and TFE

Nozzles can be classified according to aircraft flight speed and gases exhaust speed into convergent subsonic nozzles and supersonic nozzles. These nozzles can be adjustable and fixed.

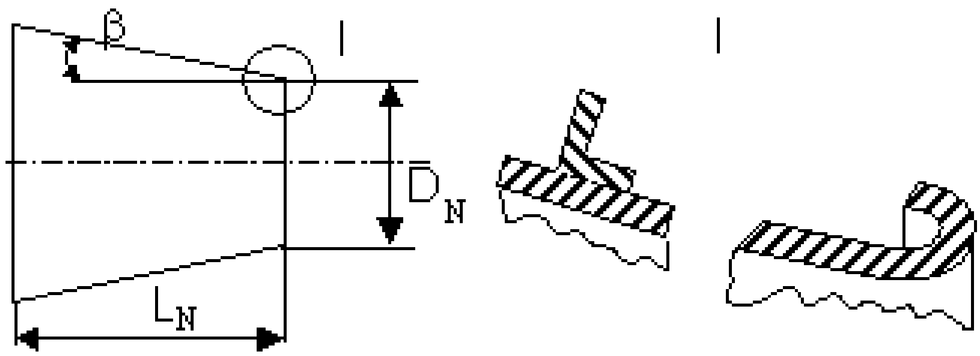


Figure 62 – Engine nozzle flanging

Fixed subsonic nozzles have apex angle β less than $10...12°$ and ratio of nozzle length to its diameter $\frac{l_N}{D_N} \approx 0,15...0,4$.

Nozzle discharge usually is round, but sometimes may be elliptical. Usually to make nozzles more rigid, designers apply special constructive means, like profiled rings, welding to nozzle trailing edge or trailing edge flanging (Fig. 62).

8.3 Adjustable nozzles

It is generally known that gas needs to be expanded to transform the potential energy of gas (pressure) into kinetic (velocity). The maximum thrust is obtained at full expansion condition, i.e. gas must leave the nozzle at atmospheric pressure. Hence, the nozzle has an appropriate form.

Nozzle operation at off-design modes deteriorates engine performances. It is clear that subsonic and supersonic nozzles must be adjustable to provide engine efficient operation at all modes and in all flight conditions. Adjustable nozzles reduce starting time and facilitate engine starting as a whole, increase compressor stability, minimize the fuel consumption at cruise mode, make engine gas generator operation invariable at afterburning and non-afterburning modes.

Nozzle control is performed by varying the area of its discharge station (for supersonic nozzles – two stations: critical and discharge). There are two methods to control the adjustable nozzles:

– by turning special plates (flaps) round hinges, which are attached to the casing; these flaps form round rim (Fig. 63). Flat nozzle, which is a sort of adjustable flap ejector nozzle, is formed by two horizontal and two vertical plates;

– by displacing specially profiled turbine cowl (central body) in axial direction;

– by pneumatic (compressed air) varying the nozzle cross-section area; air is supplied to the discharge station, where it creates the “liquid” component of the nozzle.

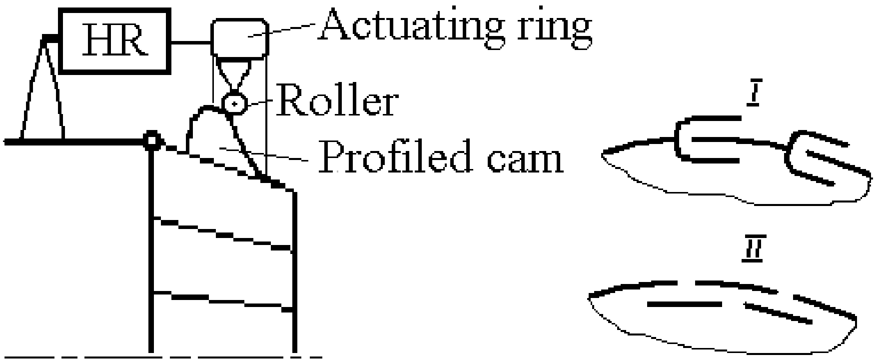


Figure 63 - Adjustable nozzle

Controlling the nozzle area at discharge station optimizes engine operation at all modes, facilitates engine starting, makes engine parameters the best possible for this mode. Nozzle is manufactured from separate flaps that are hinged at nozzle inlet station. The hydraulic ram (HR) is usually applied to vary the nozzle discharge area via actuating ring.

The law, discharge area varies, depends on cam profile (see Fig. 63). The leakproofness of the nozzle is provided by slot joining of the flaps or by above flaps. Number of flaps in already existing constructions is $z = 6...36$. The bigger flaps number nozzle has, the closer its form is to round. Flaps are cooled by the air.

Ejector nozzle is a single-row, adjustable, flap nozzle with ejector flaps mounted above. In this case, metal divergent section can be replaced with gaseous stream of the secondary air from under the ejector flaps (Fig. 64).

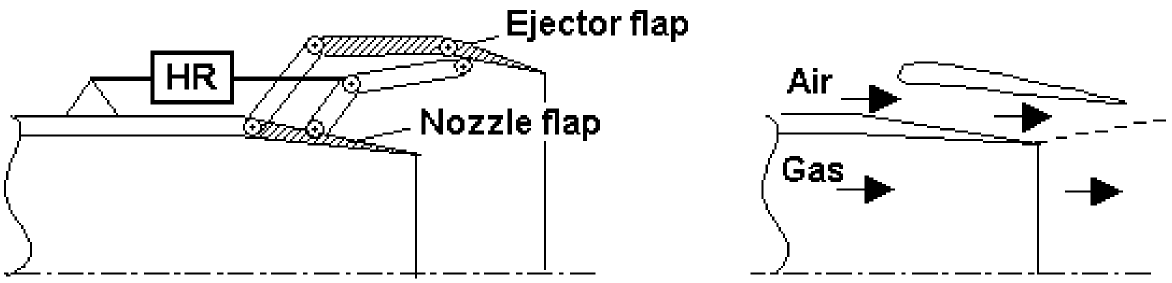


Figure 64 – Ejector nozzle

The first ejector nozzle was proposed by Russian engineer Geshvend in 1887. Big amounts of cold air available in afterburning engines and simple control make ejector nozzles reasonable to be applied for afterburning engines (ATJE R15B for airplane MiG-25). Ejector flaps can be feathered (without mechanical drive). In this case, they will self-align under pressure difference acting them (p_N and p_{amb}) from the gas stream (two-shaft ATJE R29-F-300 for airplane MiG-23).

Twin eyelid nozzle is an adjustable convergent-divergent nozzle (de Laval nozzle). It consists of two flap sets: subsonic, which are hinge fastened at nozzle inlet station, and supersonic, which are hinge fastened to subsonic ones (Fig. 65).

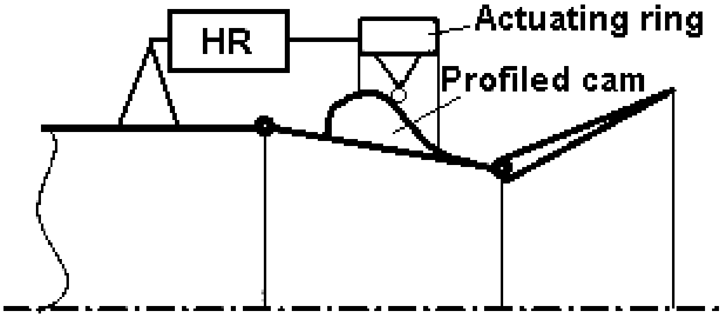


Figure 65 – Twin eyelid nozzle

Flaps position can be changed by the force applied from special rods (ATJE Al-21F-3A for airplanes Su-17 and Su-24), or switching them to feathered mode (in this case, position of flaps is determined under pressure difference, e.g. ATFE RD33 for airplane MiG-29). More complex, but more universal scheme is ejected twin eyelid nozzle termed **nozzle with gas dynamic contour break in the critical station**.

Nozzle operates similar to ejector nozzle at low flight speeds. But when flight speed becomes higher then supersonic section (second row of flaps) is attached to the rear edge of the subsonic section (Fig. 66).

Adjustable nozzle actuator consists of some hydraulic rams and flaps rearranging mechanism. Actuators can be two-position, three-position or fully-variable. Actuators are attached as close as possible to flaps hinging ring to exclude axial thermal strains influence on the nozzle area.

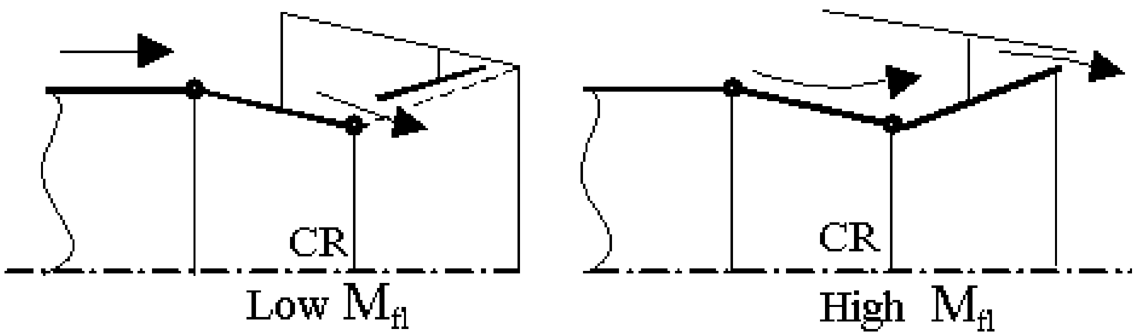


Figure 66 - Nozzle with gas dynamic contour break in the critical station

Kerosene is used as working substance in hydraulic rams at pressure 15…20 MPa. Cylinders are liquid-flow to prevent working substance (kerosene) from heating. They also can be covered with asbestos or protected by heat-reflecting screens.

The displacements of hydraulic rams are synchronized to prevent nozzle axis from being distorted. There exist hydraulic (by the constant flow valve that is placed in the exhaust pipe of the cylinder), electro-hydraulic (electrically controllable exhaust valve) or mechanical (by the rods joined by flexible shafts or worm-and-worm gears or scissors type mechanisms) synchronization. Control mechanism transmits forces from hydraulic rams to flaps. Sliding of the plunger rods is transformed to angular displacement of the flaps. Forces are transmitted from actuating ring via cam mechanism or multiple articulated joints.

Flat nozzles. Further development of all previously considered schemes was aimed to provide better coordination of nozzles with an aircraft. That was the reason flat nozzles appeared. The schematic circuit of flat nozzles is similar to axisymmetric one (Fig. 67).

The construction of flat nozzle is very advantageous for thrust vectoring, thrust reversal. Flat nozzles are also very easy to be integrated to the exhaust jet stream with the airflow leaving wing trailing part. Engines with flat nozzles have better takeoff/landing performances at the expense of vertical thrust component. Main disadvantages of such nozzles are bigger mass, comparing to analogical axisymmetric, higher losses, and problems with cooling of big flap panels.

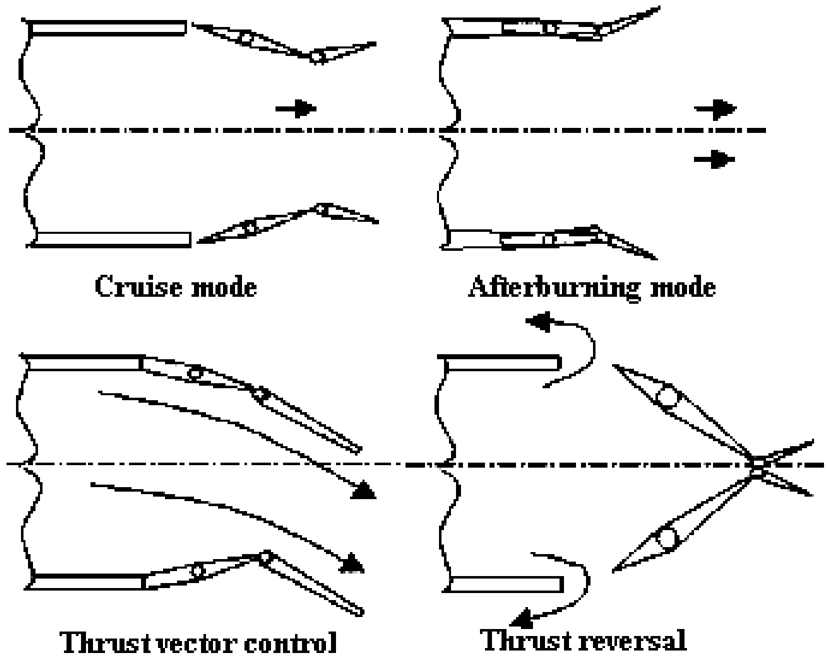


Figure 67 – The schematic circuit of flat nozzles

8.4 Thrust reversal and thrust vectoring

Thrust reversal is used for airplane deceleration during landing run and for maneuvers during taxiing. The reverser was firstly tested in 1948 (engines RD-45 and RD-10).

Base parameter of the reverser is the reversing factor, which is the ratio of engine reverse thrust to engine direct thrust at the same mode

$$-1 \le K_R = \frac{P_{REV}}{P} \le +1.$$

Nowadays, maximum obtained reversing factor at the test rig is $0,85\ldots0,9$. If turning angle of the gas flow is $120\ldots140°$, then $K_R = 0,5...0,6$.

The reverser is made the following requirements:

– the operation of the gas generator must not be affected by reverser turning on and off;

– thrust reverser at direct thrust mode must result in minimum losses of thrust;

– hot gases must be avoided from getting to engine inlet and on aircraft structure;

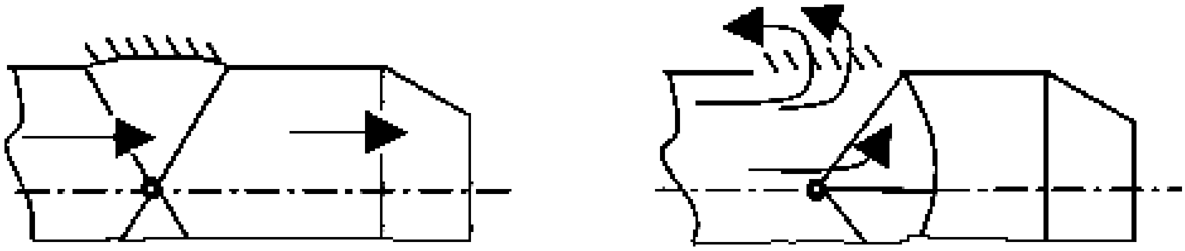
– quick thrust vectoring (for **1…2 s**);

– if aircraft has some engines, then their operation at reverse mode must be synchronized.

There are a lot of reverser schemes, but only two of them were implemented for the real engines:

– with flow reversing before propelling nozzle (“scoop” reverser) (Fig. 68);

– with flow reversing after propelling nozzle.



a b

Figure 68 – “Scoop” reverser:
a – direct thrust mode; b – reverse mode

First scheme is more complex. It needs vaned grids to reverse the flow. Grids must be sealed between “scoops”. Grids block the gas-path and hence reduce the thrust at reverse mode. But actuating force (pneumatic rams) is relatively low. Blocking doors are engine cowling elements at the second scheme. This scheme has the increased thrust losses at the reverse mode due to the necessity of high-speed flow reversing. Blocking doors must be strong and rigid enough to stand loads that are much higher than in first scheme.

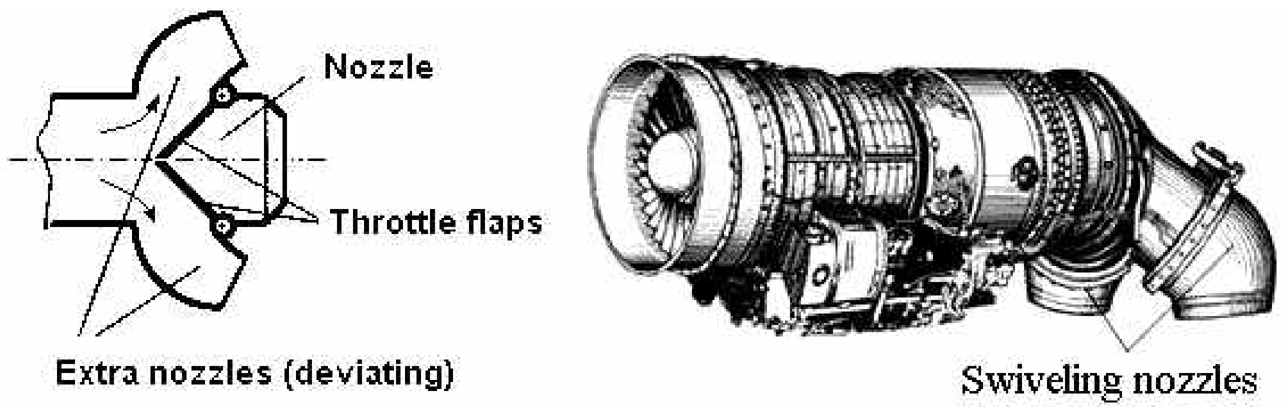


Figure 69 – Engine with swiveling nozzles

High bypass ratio turbofans without flows mixing have reverser in the bypass. The bypass is blocked by the throttle doors at reverse mode. The reversed airflow turns and, next, is guided outside through vaned grid.

Thrust vectoring is applied during vertical and shortened takeoff and landing, and for maneuverable airplanes. Thrust vectoring forms the vertical thrust component (Fig. 69). It is also applied during aircraft maneuvers.

Deviators may go into two baskets:
- nozzle independent deviators (extra nozzles or throttle flaps);
- nozzle integrated deviators (swiveling nozzles).

8.5 Noise and vibrations in power plant

Modern aircraft engine is complex heat machine with big number of systems, components and parts. From one hand, complex structure predisposes from the emergent individual vibrations in separate units and systems, and from other hand – vibration in certain systems influence other systems operation producing multi-frequency vibration of the engine. This vibration is transmitted to airplane through engine mounts, causing additional fluctuating stresses of its structure (additional stresses in their turn may lead to fatigue damages of the aircraft elements). Vibrations adversely affect crew work, passengers' convenience, and may cause failures of aircraft and automatic systems of the engine.

Power plant also generates broadband noise, which also may excite vibration as in engine elements, so in aircraft structure. These vibrations are also harmful both from strength, and from commercial points of view.

Aerodynamic disturbing factors are major noise sources of power plant. Rotating parts add harmonic components.

Acoustic noise in the engine has the aerodynamic origination. The major portion among them takes the propeller (fan), compressor, combustion chamber, turbine and jet stream noise.

Noise in exhaust systems appears when exhaust jet stream is mixed with ambient air (Fig. 70). Noise intensity depends on exhaust stream diameter (jet nozzle diameter) and its velocity.

Frequency range of aerodynamic noise is broad (to $50 \ldots 100$ kHz). Maximum vibration power lays within the frequency frame $1{,}5 \ldots 15$ kHz, depending on exhaust stream velocity. Combustion chamber generates broadband noise during normal operation, because of vortexes in reverse flow zone. Noise is complemented with additional components at off-design modes. The sources of gas vibrations appear in case of resonant combustion, self-excited oscillations, irregular fuel supply, etc.

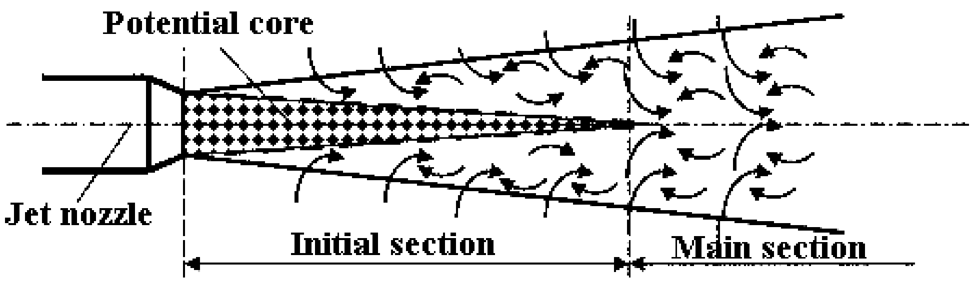


Figure 70 - Exhaust jet stream is mixed with ambient air.

Engine acoustic noise basically propagates to the front sector (upstream engine inlet) and to the rear sector (downstream jet nozzle). Noise distribution aside the engine is caused by the partial reflection and absorption of acoustic waves by structural elements (by engine casing, bypass duct and nacelle).

Maximum noise level is limited by the international standards or ICAO international standard. Limiting factors are takeoff mass of the airplane, number of engines, their thrust at takeoff mode, etc. Limitations, given in these standards are close, but it should be noted that CIS (Commonwealth of Independent States) standards are more strict than ICAO standard or, for example, American standard FAR.

Also, it should be noted that from year to year noise limitations for newly designed engines and airplanes become tighter (noise reduction is $\mathbf{4 \ldots 10\ dB}$).

Essentially important is ICAO resolution about the modernization of airplanes that are currently in use. Their noise level must also be reduced.

The problem of noise reduction can be solved only in complex "airplane – engine – airport – maintenance conditions". That is why, reducing noise level requires complex constructive and planning activities (optimization of airport runways, limiting the build-up territories near the airport, changing flight paths under localities, etc.).

As engine is the main noise source onboard the aircraft, then major activities for noise level reduction must relate its perfection.

The problem of noise level reduction during its airfield testing is easily solved by optimizing the ground location and orientation, applying the airfield suppressors and screening barriers, and improving the engine control. But main means to reduce the noise level are constructive (first of all, it is a proper choosing of working process parameters and control program).

Main parameters are bypass ratio, total compressor pressure ratio, turbine inlet temperature, total fan pressure ratio and circumferential velocity of the blade tip of the fan (LPC) first stage. Engines with low noise level have high bypass ratios $\mathbf{m = 4 \ldots 8}$ at moderate circumferential velocities ($\mathbf{v = 330 \ldots 350\ \frac{m}{s}}$). High bypass ratio turbofans are the most efficient when ratio between turbine inlet temperature and engine inlet temperature is big. But increase in TIT leads to the acceleration of the jet stream leaving the nozzle, i.e. to jet stream noise increasing (temperature rise per 100° results in jet stream noise increase on $\mathbf{3 \ldots 5\ dB}$). That is why, recommended TIT for low-noise engine is $\mathbf{T_G^*}$ **=1350…1400 K**.

The highest turbofan efficiency can be obtained at high pressure ratios (optimal compressor pressure ratios, at which fuel consumption is minimum, may reach $\mathbf{30 \ldots 40}$). The pressure ratio and noise level are indirectly related. The compressor pressure ratio affects exhaust stream velocity, gas density at nozzle discharge, circumferential velocity of blade tip of compressor first stage and compressor stages loading. The pressure ratio of low-noise engines is reasonable to choose within the limits $\boldsymbol{\pi^*_{C\Sigma} = 20...30}$.

As it is clear from previous considerations about engine main parameters, the problem of light-weight, efficient and low-noise engine cannot be solved definitely, but needs compromises.

To reduce the noise level designers can:

– profile the intake without flow disturbances;;

– enlarge axial clearances between rotating and fixed gas-path elements;

– choose optimal number of compressor guide vanes and turbine nozzle vanes;

– use single-stage fan without IGVs;

– profile exhaust duct in the way to get minimum noise.

Both, newly designed engines and existing engines with high-noise level, are exposed to special constructive arrangements to suppress the structural and jet stream noise of power plant (acoustic absorbent linings of nacelle elements and engine stator).

Operation principle of multi-lobe noise suppressor lies in the replacement of single big diameter nozzle by nozzles of smaller diameter but with equivalent area (Fig. 71).

Naturally, multi-lobe nozzle reduces engine noise during takeoff, but deteriorates its efficiency during the flight. That is why, nowadays, noise suppressors are removed from gas path at cruise modes.

Constructive measures for noise suppression make airplane and its maintenance much expensive. So, let`s consider an airplane with takeoff mass **78 tones**, which has noise level **100,4…101,2 dB** during takeoff and climbing, and **108,2…109,9 dB** during descent and landing as an example. To reduce noise level to **90,5…92,5 dB** during takeoff and **96,0…97,5 dB** during descent and landing results in increasing the airplane mass for **2000 kg**, shortening the flight distance for **510 km** (for **17,5 %**) and increasing the airplane price to **$800000** USD.

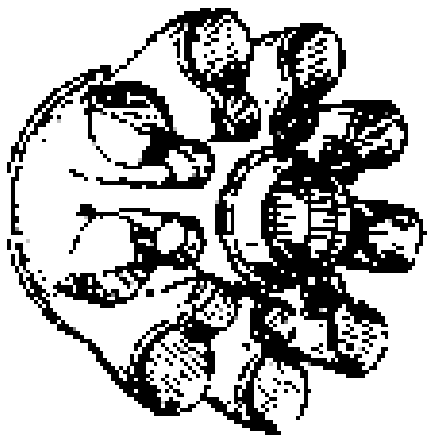


Figure 71 – Multi-lobe noise suppressor

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(Англійською мовою)

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