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**ПРОЕКТУВАННЯ СИСТЕМ СИЛОВИХ  
УСТАНОВОК ЛІТАКІВ**

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**V. F. Shmyrov, R. U. Tsukanov, A. I. Ryzhenko, V. D. Pehterev**

# **AIRPLANE POWER PLANTS SYSTEMS DESIGNING**

Synopsis

**В. Ф. Шмирьов, Р. Ю. Цуканов, О. І. Риженко, В. Д. Пехтерєв**

**ПРОЕКТУВАННЯ СИСТЕМ СИЛОВИХ УСТАНОВОК ЛІТАКІВ**

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Aircraft power plant systems design methods, such as engine mount, shock absorption, fuel system, lubrication system design and analysis are stated. Necessary information about power plant internal aerodynamics, subsonic and supersonic air intakes and exhaust units is given. Wide outlook of aircraft power plant development is given. This information is minimally required for passing exam on the course «Airplane Power Plant System Designing». The book can be useful for students to carry out the course and diploma projects. Data and information appearing in this book are for educational purposes only.

This synopsis is intended for students studying for major subject «Aerospace Engineering».

Fig. 206. Tab. 6. Bibliography: 56 names

The reviewers: ScD, Prof. Aleksey B. Leontiev,  
ScD, Prof. Boris V. Lupkin

Проектування систем силових установок літака: консп. лекцій / В. Ф. Шмирьов, Р. Ю. Цуканов, О. І. Риженко, В. Д. Пехтерев. — Х.: Нац. аерокосм. ун-т «ХАІ», 2010. — 220 с.

Викладено методи проектування систем силових установок літаків, а саме: конструювання й розрахунок систем кріплення двигунів та їх амортизації, паливних і масляних систем. Наведено необхідні відомості про внутрішню аеродинаміку силових установок (СУ), дозвукові й надзвукові повітрязабірники, а також вихідні пристрої СУ. Дано широкий огляд перспектив розвитку силових установок найрізноманітніших літальних апаратів. Дані та інформація, наведені у книзі, призначено виключно для цілей навчання.

Для студентів, що навчаються за напрямком «Авіа- та ракетобудування», при підготовці до складання іспиту з дисципліни «Проектування систем силових установок літаків», а також при виконанні курсових і дипломних проектів.

Іл. 206. Табл. 6. Бібліогр.: 56 назв

Рецензенти: д-р техн. наук, проф. О. Б. Леонтьєв,  
д-р техн. наук, проф. Б. В. Лупкін

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«Харківський авіаційний інститут», 2010 р.

## PREFACE

The task of up-to-date aircraft designing includes extensive complex of related problems. It is necessary to satisfy the requirements of aerodynamics and flight dynamics, strength and stiffness, reliability and survivability, repairability and, the main, effectiveness. Recently, different versions of economic efficiency criterion are frequently used as the main optimization criterion both for civil and military aircraft. A number of factors directly effecting on economic efficiency of created or operated aircraft are related to power plant designing solutions. Rational choice of type and number of engines, their model and arrangement on aircraft significantly effects on following all main requirements to the designed aircraft and often even to realizability of new aircraft (e. g. vertical/short take-off and landing (VTOL/STOL) aircraft, stealth aircraft and micro aviation aircraft). Engine mount scheme selection rationality and its parameters choice correctness impact not only to strength and stiffness, but also allow to eliminate resonance, discomfort condition for passengers onboard, increased crew fatigability, fast equipment wear; provide maintainability and repairability; and as a result impact to profit brought by aircraft and its competitiveness. Fuel system designing quality directly impacts to reliability and fire danger of power plant, direct and indirect aircraft maintenance expenses, for example, due to greater or less mass of unusable fuel, required fuelling time etc. Air intake designing significantly impacts to aircraft aerodynamic performance, engine efficiency, unprepared airfield basing capability, aircraft maneuverability... Therefore, great attention is paid to studying the course «Airplane Power Plant System Designing», its full and up-to-date methodical support.

This publication contains the synopsis for the lecture course «Airplane Power Plant System Designing», which is taught for the fourth course students, being studying for the major subject «Aerospace Engineering». The synopsis includes 11 themes, which are intended to power plant arrangement and classification, engine mount and its shock-absorption, fuel and lubricant systems, internal aerodynamics, subsonic and supersonic air intakes designing, exhaust units and power plants outlook.

With the brevity purpose, we use two techniques within this synopsis: when the difference is insufficient, instead «turbojet and bypass turbojet» we write «(bypass) turbojet»; and we use «[+]» sign to mark advantage and «[-]» sign to mark disadvantage correspondingly.

This synopsis can be useful for the fullest understanding of information given at lectures for students independent studying of some questions, when preparing to laboratory work passing, when doing course and diploma projects and also bachelor papers.

## **Theme 1. INTRODUCTION TO COURSE «AIRCRAFT POWER PLANT SYSTEMS DESIGNING»**

### **1.1. Subject of Discipline. Definition of Concept Power Plant. Purpose of Aircraft Power Plant**

Aircraft power plants (ENG) **are intended** for creation of thrust force, provision of energy for some systems, and also for solution of special problems.

Power plants **are** set of engines and systems providing their operation on all operational modes (Mach flight number M, altitude H, temperature T etc.), allowable for an aircraft.

**Subject** of the discipline «Aircraft power plant systems designing» are functionality, design features, arrangement, analysis of operation, methods of designing of systems and equipment of aircraft power plants.

Power plant **consists of**: main and auxiliary **engines** and **systems** providing their operation:

- propeller and propeller spinner installation (onto piston engines, turboprop engines and prop-fan engines);
- fuel system (to feed engine by fuel and fuel storage on an aircraft);
- lubrication system (to provide of friction surface lubrication and heat rejection from units of power plant);
- engine mounts (for reliable attachment of engine with the units installed on it (i.e. pumps, generators, cowlings) to attachment fittings of an aircraft structure);
- air ingestion system (air intakes with its ducts);
- exhaust system with a device of thrust reverse, with noise suppression devices;
- cooling system for engines and their power plant units;
- control system of engines and other power plant systems;
- starting system;
- fire protection system;
- deicing system;
- cowling system (engine nacelles and cowlings);
- special systems;
- auxiliary power plant (APU).

### **1.2. Over-all Technical Requirements to Aircraft Power Plants**

A number of ways are used for increase of speed, range and flight altitude, load-carrying capacity and also for improvement of characteristics of aircraft take-off and landing.

1. Required **thrust force  $P$**  measured in [N], [kg], [daN],  
or **power  $N$**  measured in [W], [h.p.], [kw] of power plant.

As it is known, the maximum flight speed is determined by formulas:

for (bypass) turbojets:  $V_{MAX} = \sqrt{2P/C_{xa}S\rho_H}$ , for piston engines and turbo-props:  
 $V_{MAX} = \sqrt[3]{2N\eta_p/C_{xa}S\rho_H}$ .

Now, the highest achievements for various engine types are the following:

for GE90-115B turbojet engine —thrust is 513950 N;

for NK-12 turboprop engine —power is 11000 kw;

for VD-4K piston engine — power is 3160 kw.

2. Minimal **specific gravity** of power plant (ratio of the engine weight to its thrust force or to its power)

for turbojet:  $\gamma_{en} = m_{en} \cdot g/P$

or for piston and turboprop:  $\gamma_{en} = m_{en} \cdot g/N$  [daN/kw], [kg/h.p.].

Now<sup>1</sup>, the highest achievements for various engine types are the following:

for piston engines is  $\gamma_{en} = 0.67...1.3$  [daN/kw];

for turboprop engines is  $\gamma_{en} = 0.27...0.33$  [daN/kw];

for turbojets is  $\gamma_{en} = 0.2...0.25$ ;

for turbojets with afterburner is  $\gamma_{en} = 0.15...0.2$ ;

for bypass turbojets<sup>2</sup> is  $\gamma_{en} = 0.165...0.22$ ;

for bypass turbojets with afterburner is  $\gamma_{en} = 0.1...0.15$ ;

for VTOL lifting engines  $\gamma_{en} = 0.1$ .

3. Minimal **relative mass of power plant and fuel** (ratio of mass of power plant (fuel) to take-off mass of the airplane)

$$\bar{m}_{P.P} = \frac{m_{P.P}}{m_0} = n_{en} k_{P.P} \frac{m_{en}}{m_0} = n_{en} k_{P.P} \left( \frac{m_{en} g}{P_0} \right) \left( \frac{P_0}{m_0 g} \right) = n_{en} k_{P.P} \gamma_{en} t_0;$$

$$\bar{m}_F = \frac{m_F}{m_0},$$

Type of aircraft	$\bar{m}_{P.P}$	$\bar{m}_F$
Subsonic passenger and transport	0.08...0.14	0.18...0.40
Maneuverable	0.18...0.22	0.25...0.30

where  $t_0 = P_0/m_0g$  — thrust/weight ratio of airplane,  $n_{en}$  — number of engines;  $k_{P.P}$  — ratio of weight of power plant to weight of engines ( $k_{P.P} = 1.2...2.2$ ).

<sup>1</sup> For comparison, Mozhaisky airplane steam engines had  $\gamma_{en} = 10.7$  [daN/kw].

<sup>2</sup> The most powerful turbojet in the world GE90-115B has  $\gamma_{en} = 0.167$ .

4. Minimal **specific fuel consumption** (the mass of fuel necessary for creation a unit of thrust (power) at one hour). For the most powerful engines it makes

$C_p, \left[ \frac{kg}{N \cdot hour} \right]$	$C_e, \left[ \frac{kg}{kw \cdot hour} \right]$	Engine
0.033	—	GE90-115B
—	0.224	NK-12
—	0.251	VD-4K
0.0662	—	D-36

5. Good aerodynamic characteristics:

– Minimal aerodynamic drag, created by power plant and minimal impact pressure losses due to optimal engine air intakes, radiators, cooling systems designing;

– Optimal propeller shape and propeller control;

– Optimal nozzle shape and nozzle control (for supersonic aircraft and for aircraft with thrust vectoring).

6. Provision of desired **flight performance** of an aircraft ( $L_{max}, V_{max}, H_{ceil}, H_{zoom}, t_{max}, m_{p.l} \dots$ ).

7. Provision of **reliability, safety and survivability of power plant**. (That is, reliability, safety and survivability must be provided at all operational modes of power plant in all conditions allowable for an aircraft).

8. Provision of necessary **strength** and vibration damping of attachment of the engine, units of power plant, nacelles, cowlings.

9. **Maintainability**, testability, repairability. (That is, convenience of maintenance, of ground test and in-flight test, a little time required for replacement of a faulty unit or engine).

10. Minimal negative effect to environment. (**Noise level and engine emission** must match to standards [3]. Noise level and vibrations in cabin should be minimal.)

11. **Protection of power plant against birds, dust** and foreign matter from ground (dust filters on air intakes of piston engines, cut-off air streams, grids, upper air intakes on MiG-29).

12. Provision of high **service life of power plant systems**.

13. **Power plant efficiency**. It is characterized by a specific fuel consumption of engine and aerodynamic perfection of power plant (minimal external drag, minimal pressure losses in an air intake, maximum efficiency of a propeller and optimal control of engine in flight).

14. General requirements to all parts of an aircraft — **minimal mass and overall dimensions**.

15. **Minimal cost of power plant life cycle** (designing + manufacturing + testing + storage + operation + maintenance + repair + salvaging).

## Theme 2. CLASSIFICATION OF POWER PLANTS. TYPES OF AIRCRAFT ENGINES AND THEIR ARRANGEMENT ON AIRCRAFT

Aircraft power plants can be classified by: engines type; number of engines, their arrangement on the aircraft etc.

### 2.1. General Analysis of Aircraft Engine Types and Areas of Their Application

#### 2.1.1. Piston Engines

Before the second half of 40th years the main aircraft engines were piston engines. Now after temporary liking for jet engines, piston engines find wide application again in general aviation (executive, trainer, agricultural, acrobatic aircraft) and also for unmanned air vehicles (UAV) of various masses and sizes (ref. Theme 11).

Piston engines are divided into air-cooled engines and water-cooled engines. **Air-cooled engine** is a casing with radially arranged cylinders. Otherwise they are called radial engines (Fig. 2.1, a). For greater uniformity of rotation the number of cylinders is usually odd (3, 5, 7, 9...). When the number of cylinders required for desired power, is not arranged within one plane, they are arranged in some rows. There are one-, two- and four-row radial engines.

**Water-cooled engine** is a casing with cylinder blocks, arranged in a line (with 2...8 cylinders in a line). By arrangement of these blocks on the front view, they are distinguished, V-shaped (Fig. 2.1, b), Y-shaped, X-shaped, H-shaped and opposed piston engines. For cooling these engines, liquid is used. This liquid is usually water. But at negative temperature special antifreeze is used. This liquid is then cooled in the special air-water cooler. Such engines are also called in-line engines.

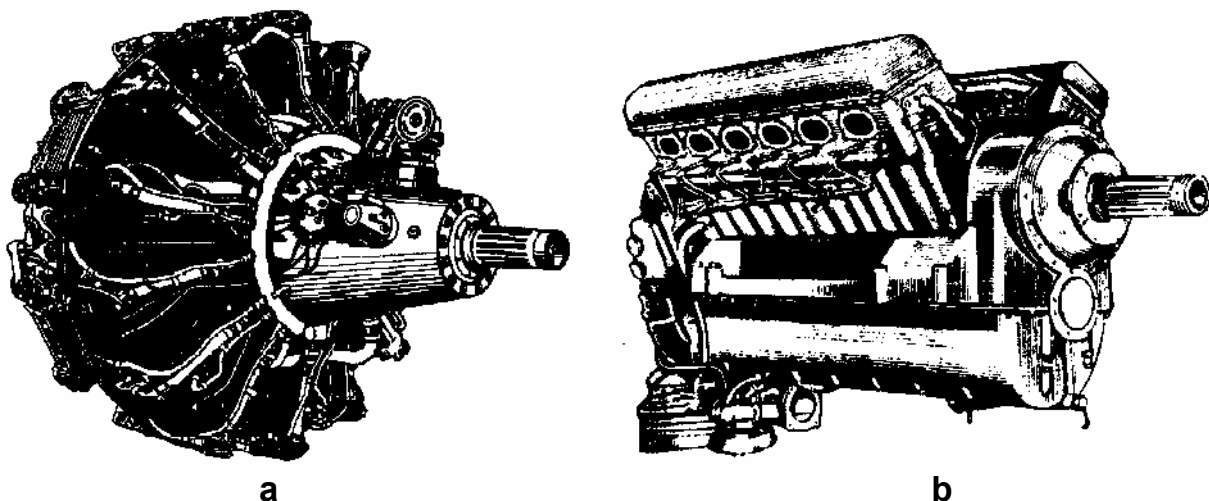


Fig. 2.1. Piston Engines (a — Radial, b — In-Line)



It is typical for piston engines:

[+] High efficiency  $C_e = 0.25...0.3$  [kg/kw·hour];

[+] High reliability, simplicity in maintenance, refined design;

[+] Compactness of power plant arrangement;

[-] High specific gravity  $\gamma_{en} = 0.67...1.3$  [kg/kw];

[-] Impossibility to achieve high speed.

The matter is that thrust developed by power plant with propeller, is equal to  $P_a = N_e \eta_p / V$ , where  $N_e$  — effective power of engine;  $\eta_p$  — efficiency of propeller;  $V$  — flight speed. As the effective power of piston engines does not depend on flight speed, and the efficiency of the propeller drops with approaching to sonic speed, then the thrust force developed by the power plant with a propeller decreases at the flight speed grows. At the same time, aerodynamic drag (required thrust) sharply increases.

### 2.1.2. Turboprop Engines

At the moderated subsonic flight speed ( $M < 0.6...0.8$ ) and moderated flight altitude (6000...10000 m) turboprop engines (Fig. 2.2) have broad applications.

Turboprop engines have the following properties:

[+] Capability to obtain a high-power in one unit (NK-12 — 11000 kw);

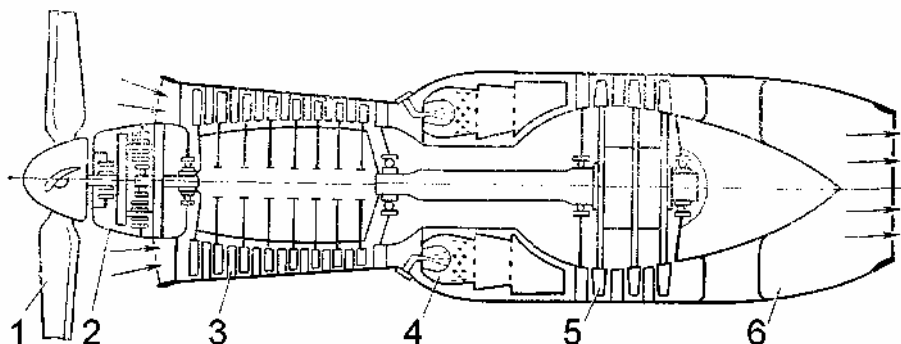
[+] Small specific gravity ( $0.27...0.33$  [daN/kw]);

[+] Small area of a mid-section, in comparison with piston engines;

[+] Specific fuel consumption is less, than at turbojet ( $C_e = 0.33...0.4$  [kg/kw·hour]);

[-] Specific fuel consumption is more, than at piston engines;

[-] Speed limitation because of a propeller;



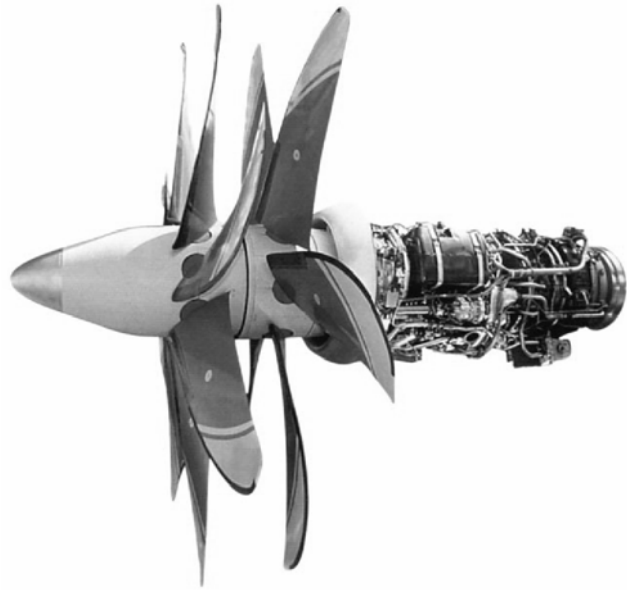
**Fig. 2.2. Scheme of Turboprop Engine:**

**1 — Propeller; 2 — Reduction Gear; 3 — Compressor; 4 — Combustion Chamber; 5 — Turbine; 6 — Jet Nozzle**

[-] Specific gravity and the cost of maintenance is more, than at turbojet;

[-] Limitations of arrangement, because of a propeller.

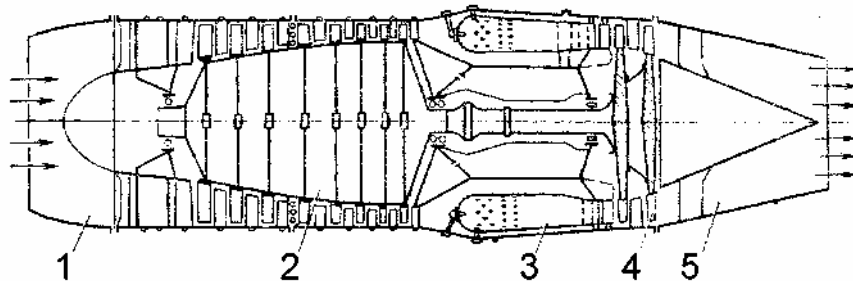
**Prop-fan engines** differ from turboprop engines only by propeller (Fig. 2.3). The propeller of a prop-fan engine is optimized for high subsonic flight speed ( $M = 0.8...0.95$ ). It has high efficiency at such flight speed. Such propellers usually have broad scimitar-type blades. It reduces aerodynamic drag of rotated blade when the supersonic speed is reached on the blade tip. Usage of such propellers allows to decrease specific fuel consumption by 15-20% as well as noise and vibration into cabin.



**Fig. 2.3. Prop-Fan Engine**

### 2.1.3. Turbojets

To reach higher (including supersonic) speed is possible by means of turbojet (Fig. 2.4). It is considered, that specific thrust (the ratio of engine thrust to air consumption through it) of turbojet becomes more, and the specific fuel consumption — less, than turboprop has at speed over 800 km/h. Turbojet is a direct reaction engine, that integrates an engine and a propulsor (Engine is a machine transferring some kind of energy to mechanical work. Propulsor is a devise providing movement.) in one unit (at piston engine and turboprop the propulsor is a propeller). Now, turbojets are practically excluded by more efficient by-pass turbojet engines from application at both civil and combat aircraft.



**Fig. 2.4. Scheme of Turbojet:**

**1 — Air Intake; 2 — Compressor; 3 — Combustion Chamber;  
4 — Turbine; 5 — Jet Nozzle**

- Turbojet has:
- [+] Small specific gravity (0.2...0.25);
  - [+] Small overall dimensions;
  - [+] High thrust (60...51395 [daN]);
  - [+] Convenience of arrangement (because of absence of a propeller);
  - [–] Big specific fuel consumption ( $C_p = 0.7...0.9$  [kg/daN·hour]).

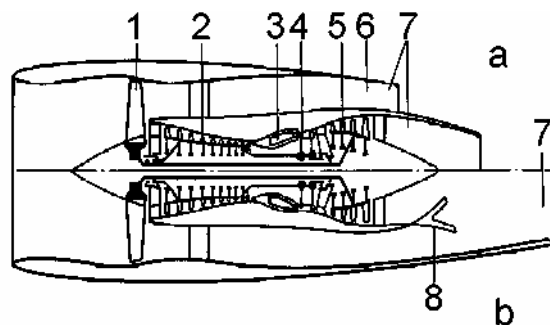
#### 2.1.4. Bypass Turbojets

At transonic and moderated supersonic flight speed the bypass turbojets (Fig. 2.5) are the most efficient. The bypass engines differ from turbojets by presence of the second (external) contour, air flows around of the combustion chamber and turbines by which. Then this air mixes with combustion products of the first contour. Thus the air consumption increases and speed of jet stream decreases.

Turboprop accelerates the big air consumption up to a low speed. Turbojet accelerates the small air consumption up to a high speed. Bypass turbojet takes an intermediate place between them. A fan of the bypass turbojet is a version of propeller in a ring channel.

Bypass turbojets are characterized by bypass ratio. It is a ratio of an air consumption through an external (air, cool or second) contour to an air consumption through an internal (gas, hot or first) contour

$$m = \frac{m_2}{m_1}$$



**Fig. 2.5. Scheme of Bypass Turbojet:**

- a — Bypass Turbojet Without Mixing; B — Bypass Turbojet With Mixing;**  
**1 — Fan; 2 — High Pressure Compressor; 3 — Combustion Chamber;**  
**4 — Shaft With High Pressure Turbine; 5 — Low Pressure Turbine; 6 — Jet**  
**Nozzle Of Air Contour; 7 — Jet Nozzle; 8 — Mixer**

Bypass turbojets have the following advantages.

1. When bypass ratio is low ( $m < 2$ ) the specific fuel consumption is usually 15...20 % less, than at turbojets. Such bypass turbojets are applied at supersonic flight speed. When  $m = 5...8$  the specific fuel con-

sumption is usually 40...45 % less, than at turbojets. Such bypass turbojets are used at subsonic flight speed.

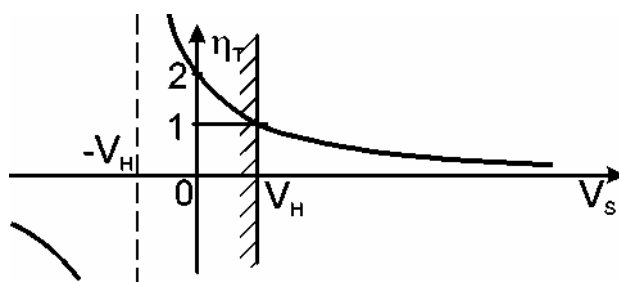
2. Bypass turbojets have lower noise levels (in comparison with turbojets). Because noise generated by gas stream sharply depends on speed difference between gas stream and surrounding air. And the stream from the first contour (fast and noisy) is surrounded with a stream from the second contour (slow and less noisy).

Specific fuel consumption of bypass turbojets is low because of high thrust efficiency. Thrust efficiency is a ratio of effective thrust work (which is made by gas), to available kinetic energy of the gas stream

$$\eta_T = \frac{2}{1 + \frac{V_s}{V_H}}$$

where  $V_s$  — speed of a jet stream;  $V_H$  — flight speed.

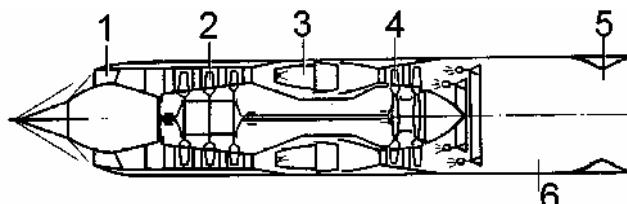
Thrust efficiency tends to one, when speed of the jet stream tends to flight speed from above. It happens in bypass turbojets with the big bypass ratio (Fig. 2.6). Such engines are sometimes called **turbofan**.



**Fig. 2.6. Relation between Thrust Efficiency and Speed of Jet Stream**

### 2.1.5. Turbojet and Bypass Turbojets with Afterburner

At supersonic flight speed the aerodynamic drag grows very sharply. Hence, bigger engine thrust is required respectively. It can be reached by use of the afterburner behind of the engine (Fig. 2.7). Such engines are called (bypass) turbojets with afterburner. By the scheme such engines are an intermediate link between (bypass) turbojets and ramjet engine.



**Fig. 2.7. Turbojet with Afterburner:**

**1 — Air Intake; 2 — Compressor; 3 — Combustion Chamber; 4 — Turbine; 5 — Jet Nozzle; 6 — Afterburner**

A disadvantage of such engines is essential increase of the specific fuel consumption when afterburner is on.

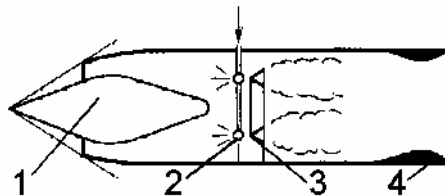
Such engines are used:

1) for airplanes with a subsonic main flight mode, which restrictedly uses a supersonic speed, bypass turbojets ( $m = 0.3...1.3$ ) with afterburner;

2) for airplanes with supersonic cruising speed:  
(bypass) turbojets with afterburner, at  $M \leq 2.5$  ;  
turbojets with afterburner, at  $M \leq 3.5$  .

### 2.1.6. Ramjet Engines

For flight speed, corresponding  $M \geq 3.5$  the compressor is not needed. It is possible to provide compression in the air intake. In such cases ramjet engines (Fig. 2.8) are appropriately used.



**Fig. 2.8. Scheme of Ramjet Engine:**

**1 — Air Intake; 2 — Fuel Nozzles; 3 — Flame Stabilizer; 4 — Jet Nozzle**

They distinguish ramjet engines with subsonic combustion of fuel which use at  $M < 8$ , and ramjet engines with supersonic combustion of fuel, which are used at higher speed.

Ramjet engines have:

[+] Low specific gravity (approximately two times lower, than turbojets have);

[+] Simple design;

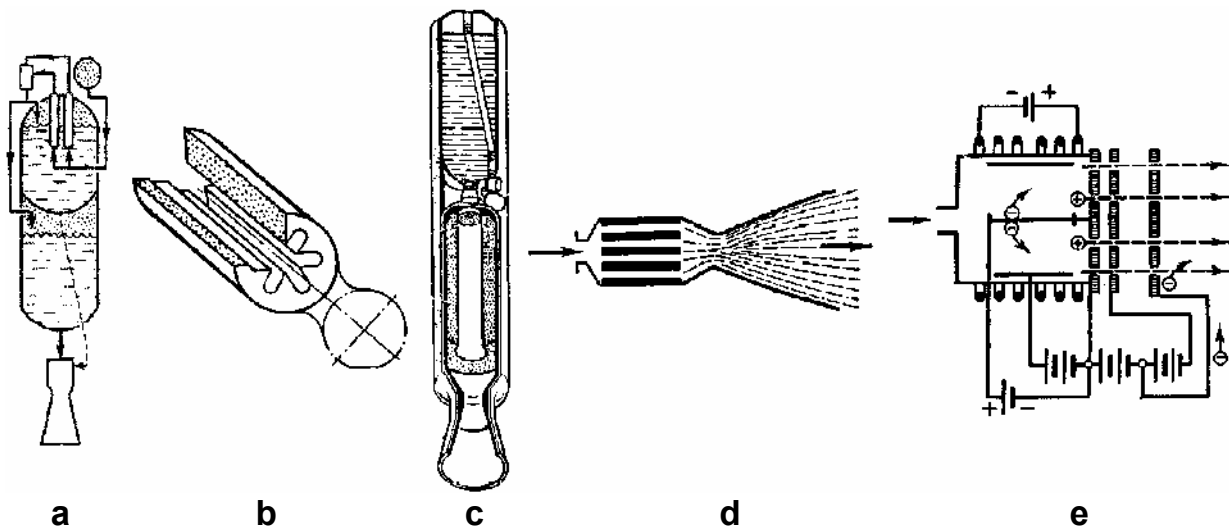
[+] High efficiency at  $M > 1.5$  ;

[-] Need of acceleration up to  $M = 1.5$  by engines of another type.

On airplanes and helicopters, ramjet engines are not practically used, except for experimental vehicles. Ramjet engines have found a broad application in supersonic cruise missiles.

### 2.1.7. Rocket Engines

Rocket engine as a first approximation represents the combustion chamber with a jet nozzle (Fig. 2.9). They have not air intake. Fuel for rocket engines is onboard a vehicle. In this case the fuel is combustible and oxidizing agent. Rocket engines are divided into: liquid-propellant rocket engines (Fig. 2.9, a); solid-propellant rocket engines (Fig. 2.9, b);



**Fig. 2.9. Rocket Engines**

hybrid rocket engines (Fig. 2.9, c); nuclear rocket engines (Fig. 2.9, d); electro-rocket engines (Fig. 2.9, e). In aviation the first two types find restricted application.

**Liquid-propellant rocket engines** have:

- [+] Small engine weight (20...30 times less, than piston engine);
- [+] Small overall dimensions;
- [+] Independence of thrust on the flight altitude and speed;
- [+] Absence of air intakes;
- [-] Need of storage of oxidizing agent onboard the vehicle;
- [-] Higher specific fuel consumption (20...50 times more, than piston engine).

**Solid-propellant rocket engines** in addition to this have:

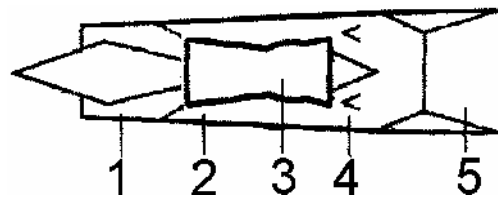
- [+] Little start-up preparation time;
- [+] Long time of high availability to start;
- [-] Complexity of thrust control;
- [-] Complexity of thrust cutoff;
- [-] Complexity of keeping constant thrust during operating time.

Rocket engines are extremely seldom used in aviation: 1) as a rocket booster during takeoff from a short runway; 2) as a rocket booster in fight. Last version of application of rocket engines is more likely property of a history, than practical use. Rocket engines have found the broadest application in rockets, including air and anti-aircraft.

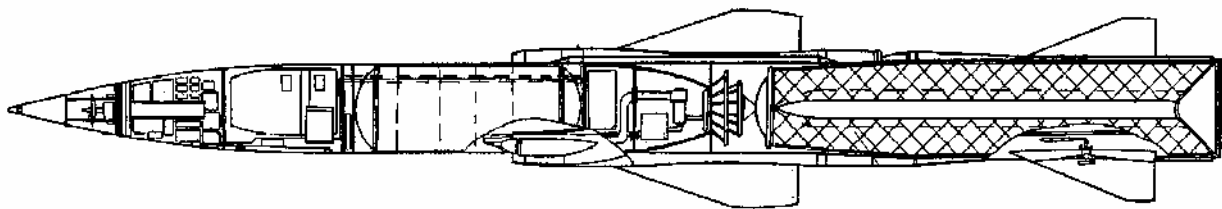
### 2.1.8. Combined Engines

To obtain a good engine performance in a broad range of altitudes and flight speeds, now it is considered appropriate, a combination of two types of engines in one unit. Such engines are called combined engines or variable cycle engines:

- turbo-ramjet (Fig. 2.10);
- turbo-rocket;
- rocket-ramjet (Fig. 2.11).



**Fig. 2.10. Combined Turbo-Ramjet Engine:**  
**1 — Common Air Intake; 2 — Bypass Contour For Ramjet Engine; 3 — Turbojet Engine; 4 — Fuel Nozzle Manifold; 5 — Common Jet Nozzle**



**Fig. 2.11. Combined Rocket-Ramjet Engine**

Such engines allow to take-off and land in turbojet or solid-propellant engine mode, and cruise flight — in ramjet or liquid-propellant engine mode. This fact should increase flight range and provide acceptable characteristics of take-off and landing (if it is required).

Now work is underway on creation of such engines for spaceplanes in the world. Another area of their application is rockets.

### **2.1.9. Combined Power Plants**

Combined power plants are power plants in which engines of different types are used. They were used in days of transition from piston engines to turbojets. Now they are property of a history. But, it is known, the history is repeated.

For instances it is possible to remind:

(piston engines + turbojets): Convair B-36; Ryan FR-1 Fireball; Su-7 (first with this name);

(turbojets + liquid-propellant engines): SM-50 (version of MiG-19); E-50 (version of MiG-21);

(piston engines + solid-propellant engines): Me-323 Gigant.

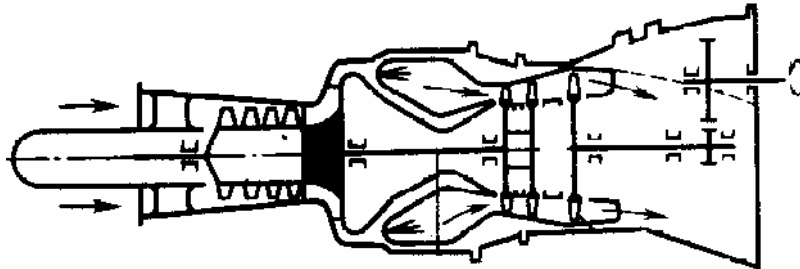
(piston engines + ramjet engine): Yak-7 (prototype version).

### **2.1.10. Turboshaft Engines**

Turboshaft engines are very similar to turboprop. Only difference is a free turbine behind the main turbine (Fig. 2.12). This free turbine is not

linked mechanically to the compressor. There is only gas-dynamical linkage. The effective power of the engine is transmitted by the free turbine.

Such engines are widely used in helicopters. They have greater stability of operation under specific conditions of helicopters, in comparison with turboprop engines.



**Fig. 2.12. Scheme of Turboshaft Engine**

## **2.2. Auxiliary Power Unit**

Auxiliary power unit (APU) is intended generally:

1. To start the main engines on ground and in flight.
2. To supply an airplane with the electric power on ground and in flight in emergency situation.
3. To maintain activity of air conditioning system on ground.

Sometimes, in addition to these it is used for:

1. Emergency supply of a hydro system.
2. Creation of additional thrust (An-24/26; An-71).

APU consists of the same systems as the main power plant consists:

- Small gas-turbine engine;
- Engine mounts;
- Fuel system;
- Lubrication system;
- Starting system;
- Fire protection system;
- Air-supply system and an exhaust system.

It is necessary to note, that some of the systems can be a part of systems of main power plant (for example: fuel and fire protection systems).

APU are usually located onboard of an aircraft in a sponson (Il-76), or in an engine nacelle (An-24/26) or in a tail part of a fuselage (An-140). The main requirements to APU location are:

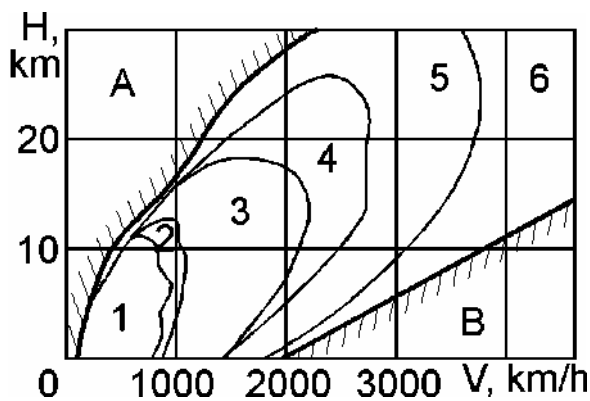
1. Fire safety.
2. Not to disturb airframe of the assembly, which is located inside.
3. Convenience of maintenance.



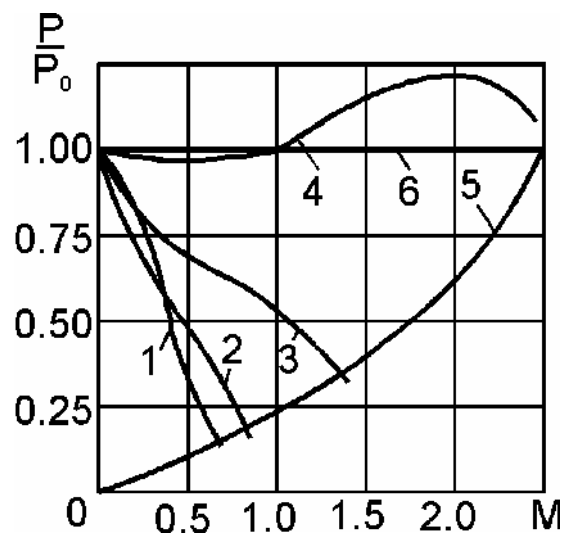
## 2.3. Engine Selection

The engine type selection is carried out depending on technical requirements to a designed aircraft. The areas of rational application of engines of various types are shown in Fig. 2.13. Here are designated areas of effective use for: 1 — piston engines; 2 — turboprops; 3 — bypass turbojets; 4 — turbojets; 5 — ramjet engines; 6 — rocket engines. A — area of insufficient wing lift. B — area of an inadmissible aerodynamic heating.

In Fig. 2.14 relative engine thrust dependences on Mach flight number are shown. From the figure it is obvious that, it is impossible to reach sonic speed by using piston engine or turboprop. It is seen too, that ramjet engine becomes effective at  $M > 1.5$ . As a whole, Fig. 2.14 good illustrates previously said.



**Fig. 2.13. Application Areas of Engines of Various Types**

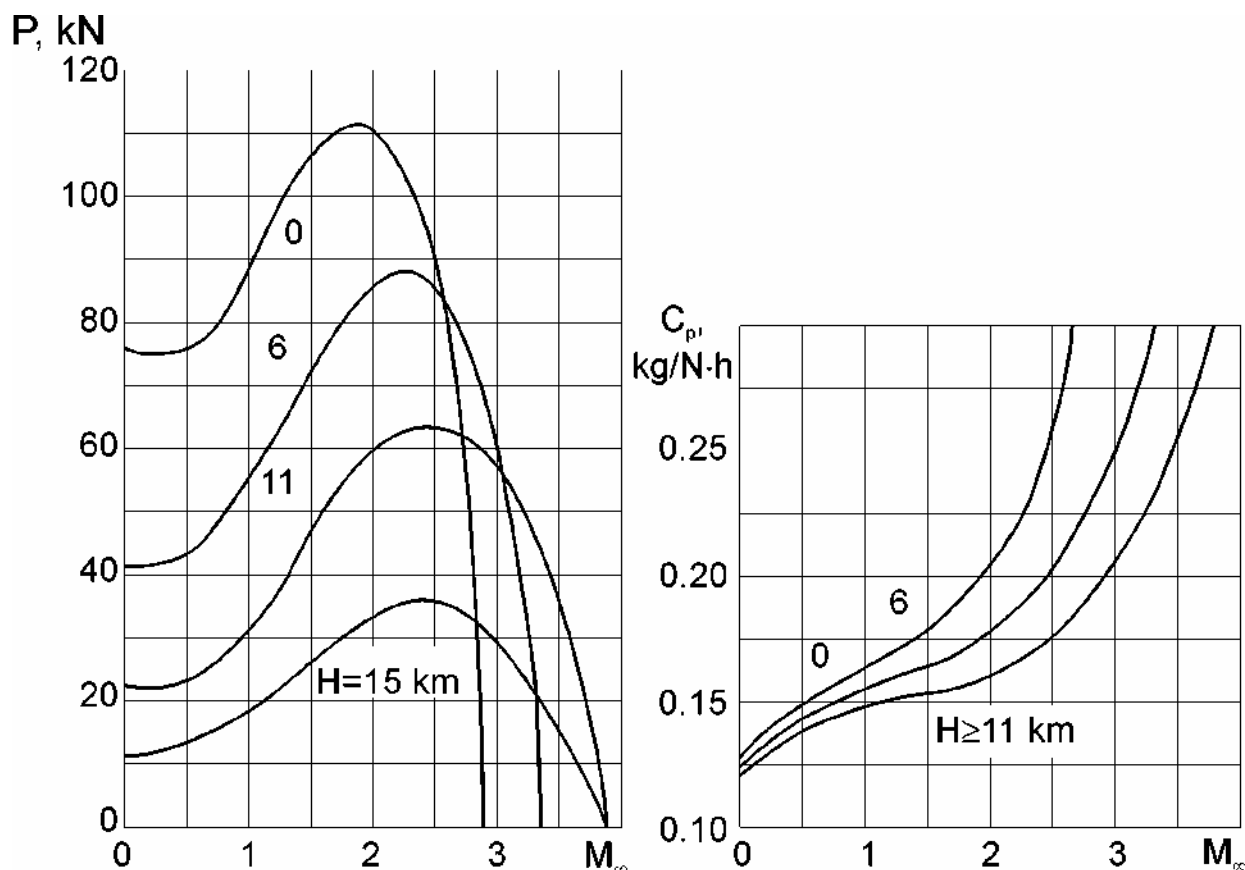


**Fig. 2.14. Dependences of Relative Engine Thrust to Mach Flight Number**

Engine model selection is carried out by comparative analysis of altitude–airspeed performance, overall dimensions, mass and air consumption of several engines (in real designers bureau condition, as a rule, the new engine or its modification optimized for a new aircraft is created). Fig. 2.15 shows typical altitude–airspeed performance of bypass turbojet.

## 2.4. Number of Engines

One of the main requirements to up-to-date airplanes (with more than one engine) is ability to continue take-off and climb when one engine has failed. Thus for good safety the vertical speed of climb and the flight path angle must be more then or equal to the specified values. These values depend on the type of airplane and are specified by requirements to the designed airplane. These values can influence on the number of engines installed onboard the airplane.



**Fig. 2.15. Altitude–Airspeed Performance of By–pass Turbojet**

Equations for motion of an airplane at steady climb are known from a flight dynamics:

$$\begin{cases} P = X_a + mg \sin \Theta; \\ Y_a = mg \cos \Theta, \end{cases}$$

where  $P$  — thrust;  $X_a$  — aerodynamic drag;  $Y_a$  — lift;  $mg$  — weight;  $\Theta$  — flight path angle.

Whence we obtain 
$$P = mg \left( \frac{\cos \Theta}{K} + \sin \Theta \right),$$

where  $K$  — lift–drag ratio of the airplane.

Thus, total thrust required for the climb at  $\Theta = \Theta_{\min}$ :

$$P_{req} = mg \left( \frac{\cos \Theta_{\min}}{K_{t/o}} + \sin \Theta_{\min} \right).$$

Then the available thrust of all engines, selected by the condition of one engine is failed during takeoff, makes:

$$P_{av} = \frac{n_{en}}{n_{en} - 1} mg \left( \frac{\cos \Theta_{\min}}{K_{t/o}} + \sin \Theta_{\min} \right).$$

Knowing the thrust, required for flight with a cruising speed at a cruise altitude, and also thrust, required to maintain the desired take-off run, we compare them with available thrust from the previous expression. Thus we obtain the expression to calculate the number of engines:

$$n_{en} = \frac{P_{av}}{P_{av} - mg \left( \frac{\cos \Theta_{\min}}{K_{t/o}} + \sin \Theta_{\min} \right)}.$$

For medium-range and short-range airliners (An-140, An-148, Tu-334, Il-114) the installation of two engines is typical. The installation of three engines on up-to-date airliners happens very seldom. On long-range and intercontinental airliners of previous generation four engines are usually installed (B-747, A-340, Il-96-300). It was forced by ICAO requirements for passenger airliners, flying above a sea during more than 4 hours. Now, this requirement is not presented because of engines reliability increase. And for new generation airliners, installation of two engines is typical. Only super heavy airplanes, such as A-380, are equipped with four engines (ref. Theme 11).

On up-to-date combat aircraft two engines are usually installed. It is called by requirements of combat survivability. Only on long-range heavy bombers (B-1B, B-2, Tu-95, Tu-160) four engines are used. The big rarity is to use greater number of engines (B-52 — eight engines).

## 2.5. Analysis of Scheme of Engine Arrangement on Aircraft

### 2.5.1. Piston Engines

**Piston engine is in nose part of fuselage** (Fig. 2.16).

In such scheme:

[+] The power plant is good inscribed in an airplane; hence it has good aerodynamic characteristics.

[+] The propeller operates in undisturbed air flow.

[+] The power plant is very compact.

[+] It is convenient to maintain the power plant.

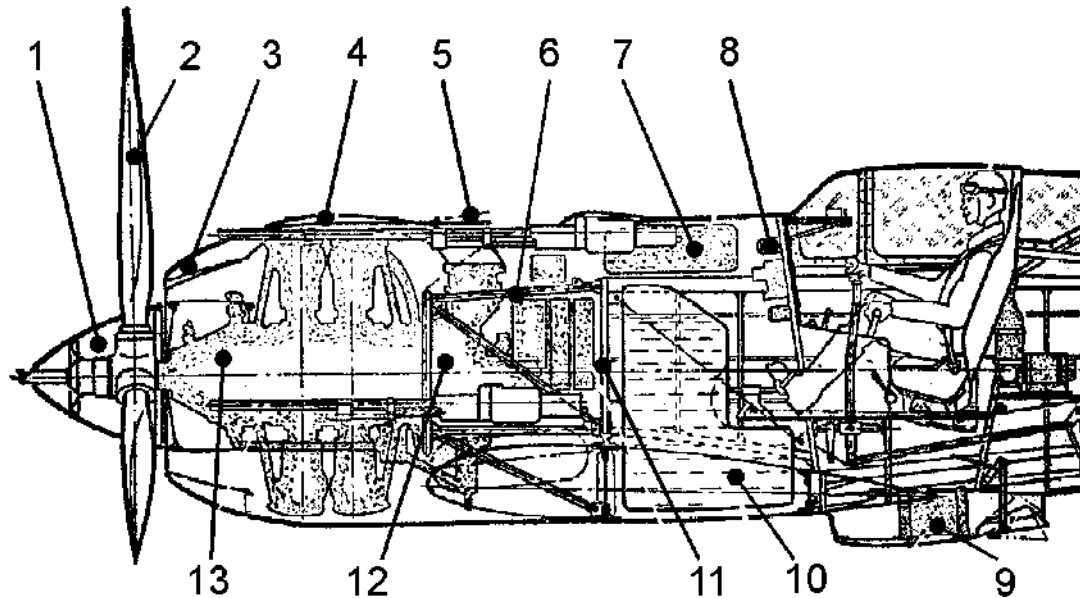
[+] The engine defends a pilot when firing from a forward hemisphere.

[+] The fuselage is free to arrange payload.

[-] It is difficult to arrange an armament, a radar, a nose landing gear strut.

[-] Bad forward view.

[-] When landing gear is failed to extend at landing the propeller is damaged. The off-load engine can be damaged too.



**Fig. 2.16. Arrangement of Piston Engine in Nose Part of Fuselage:**  
 1 — Propeller Spinner; 2 — Propeller; 3 — Cowling; 4 — Air Intake of the Carburettor; 5 — Cowl Gills (Cowl Flaps); 6 — Engine Mounts; 7 — Oil Tank; 8 — Power Plant Control Instruments; 9 — Oil Cooler; 10 — Fuel Tank; 11 — Load-Carrying Frame; 12 — Exhaust System; 13 — Engine

Practically all single-engine piston-engined airplanes were built by this scheme.

**Piston engine in mean part of fuselage (Fig. 2.17).**

[+] Good capabilities to arrange armament, equipment, a nose landing gear strut in a nose of a fuselage.

[-] Transmission increases weight and reduces reliability of the power plant.

[-] Problems with cooling of engine.

Bell P-39 Aircobra and P-63 Kingcobra fighters were serially built by such scheme.

**Piston engine with a pusher propeller in a tail part of a fuselage (Fig. 2.18).**

[+] The wing and whole airplane is streamlined by an undisturbed air flow.

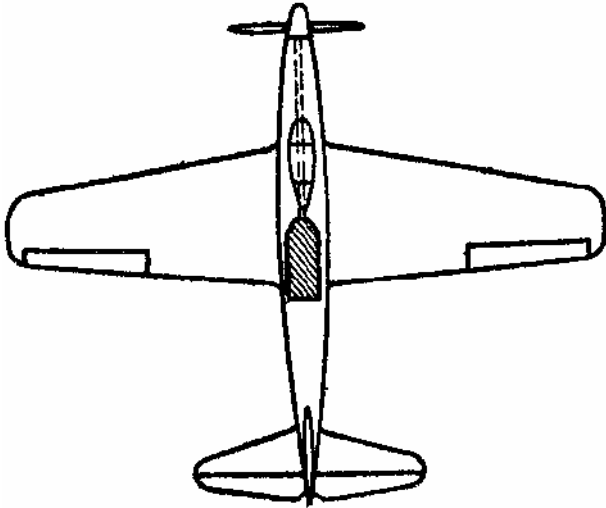
[+] It is simple to arrange an armament, equipment, a nose landing gear strut in a nose part of a fuselage.

[+] Good forward view.

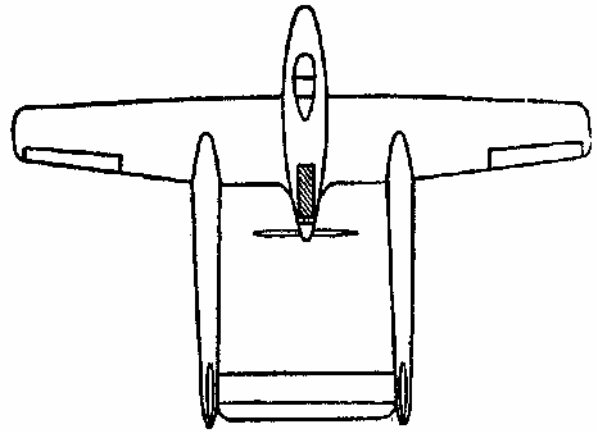
[-] The propeller operates in the disturbed air flow.

[-] It is difficult to maintain take-off and landing angles of attack because the propeller can contact a runway.

[-] Emergency escape of an airplane is dangerous because of a propeller.



**Fig. 2.17. Arrangement of Piston Engine in Middle Part of Fuselage**



**Fig. 2.18. Arrangement of Piston Engine in Tail Part of Fuselage**

Such scheme was used for some experimental fighters, such as Northrop XP-56 and Kusu Hikoky J7W Sinden. Sometimes this scheme of engine arrangement was used in combination with twin-boom fuselage scheme, e. g. SAAB J-21.

**Piston engines with tractor propellers on a wing (Fig. 2.19).**

[+] It is simple to arrange an armament, equipment, a nose landing gear strut in a nose of a fuselage.

[+] Good forward view.

[+] If the airplane can fly with one failed engine, this scheme has higher survivability, than single-engine one has, and vice versa.

[+] The engines unload a wing in flight. Hence the wing has smaller weight.

[+] Main landing gear struts are good retracted into engine nacelles.

[+] Wing propeller slipstream improves stall performance of the wing.

[-] There is a big turning moment when one (especially outboard) engine fault.

[-] There is an additional interference drag of a wing and an engine nacelle.

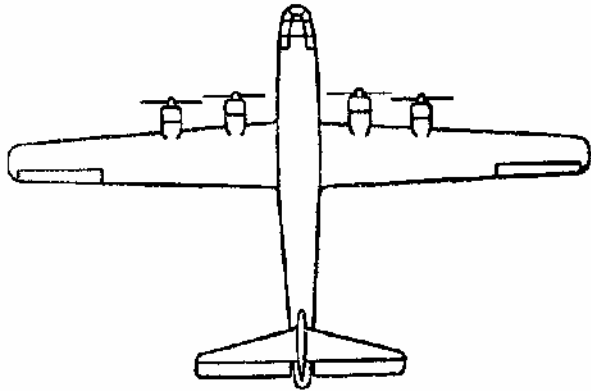
[-] The wing is streamlined by a disturbed flow.

Practically all double-engine and four-engine piston-engined airplanes were built according such scheme.

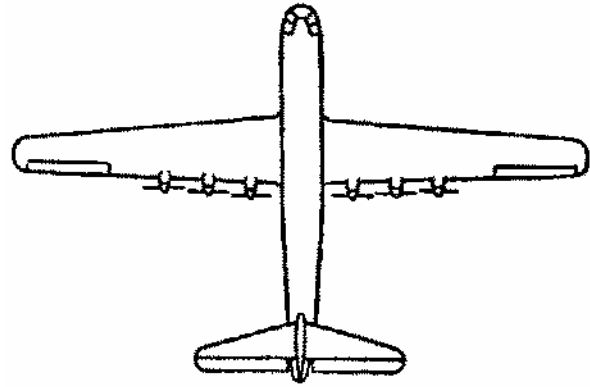
**Piston engines with pusher propellers on a wing (Fig. 2.20).**

[+] The wing is streamlined by undisturbed flow. When using laminar profile, it allows significantly to lower drag of an airplane (hence to increase range and to reduce fuel weight).

[-] As the wing trailing edge is held by engines, it is difficult to arrange the high-lift devices.



**Fig. 2.19. Arrangement of Piston Engines with Tractor Propellers on Wing**



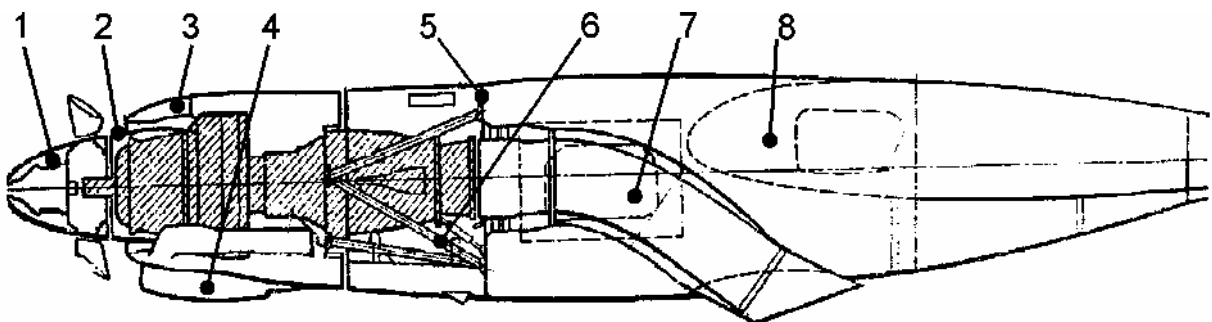
**Fig. 2.20. Arrangement of Piston Engines with Pusher Propellers on Wing**

Such scheme was used at the Convair B-36 production bomber, at a number of prototype airplanes built by the «fling wing» aerodynamic configuration by Northrop and Horten.

### 2.5.2. Turboprop Engines

Schemes of arrangement of turboprop engines, their advantages and disadvantages are similar to those of piston engines. The widest application has the scheme with two or four turboprops on a wing. Here there are two versions of relative positions of an engine nacelle and a wing.

**Top of engine nacelle is in line with upper surface of a wing** (Fig. 2.21).



**Fig. 2.21. Arrangement of Turboprop Engine under Wing:**  
 1 — Spinner; 2 — Cowl of the Propeller Gear; 3 — Cowling; 4 — Oil Radiator; 5 — Load-Carrying Frame (Firewall); 6 — Engine Mounts;  
 7 — Exhaust System; 8 — Load-Carrying Rib; 9 — Firewall

[+] The engine nacelle is short. Hence, it has smaller weight and drag.

[+] Maximum thickness of the engine nacelle is brought forward in relation to maximum thickness of a wing. It increases a critical Mach number of such scheme.

[+] High-lift devices can be arranged on trailing edge.

Turboprops are usually installed by such scheme at high-wing monoplanes (An-140, An-70, Tu-95).

**Lower surface of engine nacelle is in line with the lower wing surface** (Fig. 2.22).

[+] The main landing gear is good arranged in an engine nacelle.

[-] The extension pipe is required for exhaust gas and heat protection (firewall) under it. It increases mass and sizes of the engine nacelle.

[-] All advantages of the previous scheme become disadvantages here.

Turboprops are usually located by such scheme at low-wing monoplanes (Il-18, Il-114).

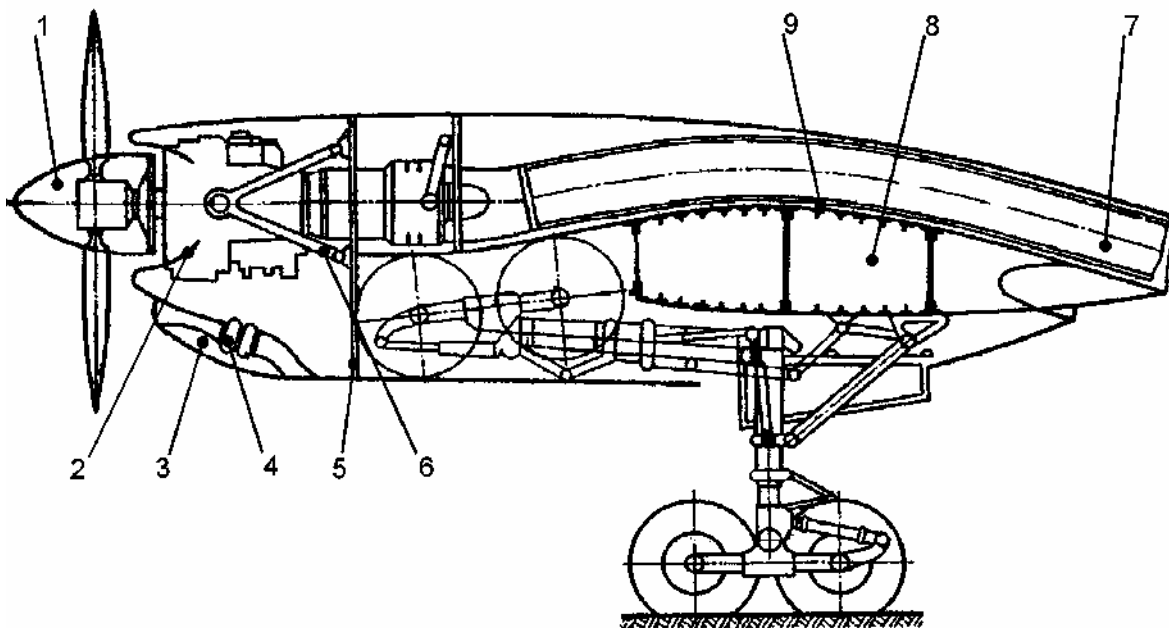


Fig. 2.22. Arrangement of Turboprop above Wing

### 2.5.3. Single-Engine Turbojet Airplanes

«Redan» scheme (Fig. 2.23).

[+] The air intake is short; hence the pressure losses are small.

[-] It is difficult to arrange an armament, radar, and a nose landing gear strut.

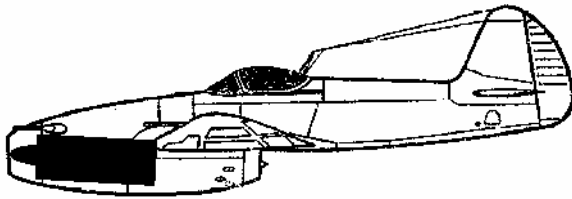
[-] The pilot has a bad in-flight view.

[-] There are some problems with taking aside of jet stream.

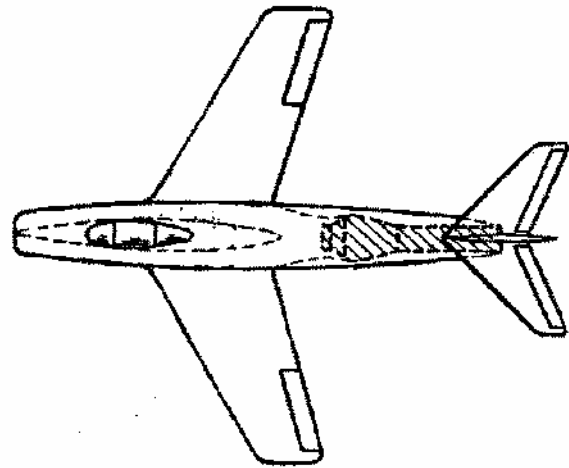
This scheme has been widely used in days of transition from piston engines to jet (MiG-9, Yak-15, Yak-17, Yak-23).

**The turbojet in a fuselage with a nose air intake** (Fig. 2.24).

[+] The air intake is in undisturbed flow.



**Fig. 2.23. Arrangement of Turbojet by «Redan» Scheme**

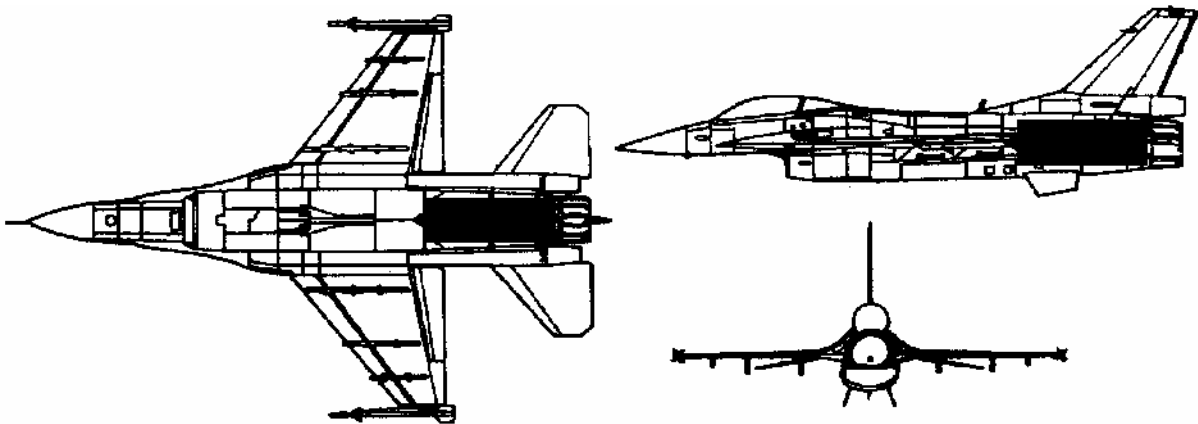


**Fig. 2.24. Arrangement of Turbojet in Fuselage with Nose Air Intake**

- [+] The air intake is good arranged in a fuselage.
- [-] The air intake duct is long; hence, the pressure losses are big.
- [-] The air intake duct takes a payload volume in a fuselage.
- [-] It is difficult to arrange an armament, radar, and a nose landing gear strut.

Many old jet fighters (MiG-15, MiG-17, La-15) were built by such scheme.

**(By-pass) turbojet is in a fuselage, the air intake is under nose of a fuselage (Fig. 2.25).**



**Fig. 2.25. Arrangement of Turbojet in Fuselage, Air Intake Is under Nose of Fuselage**

- [+] It is simple to arrange radar in a nose of a fuselage.
  - [+] It is possible to use the subsonic air intake on the supersonic airplane (F-16, A-7, F-8).
  - [+] The air intake is shorter than nose one.
  - [-] It is possible to ingest some foreign matter from ground.
- The production fighters MiG-17PFU, MiG-19PM, F-8, F-16 were built



by such scheme.

**The turbojet is in a fuselage, air intakes are in root sections of a wing** (Fig. 2.26).

[+] It is simple to arrange an armament, equipment, a nose landing gear strut in a nose of a fuselage.

[+] Good forward view.

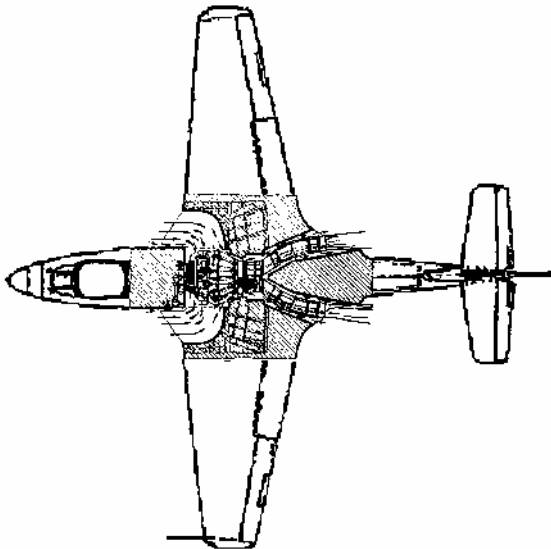
[+] The air intakes are short; hence the friction pressure losses are small.

[−] The curved air intake duct increases pressure losses.

Many shipborne USA fighters, for example, MacDonnell FH-1 Phantom, Grumman F9 Panther, Hawker Hunter were built by such scheme.

#### **2.5.4. Multiengined Airplanes with (Bypass) Turbojets**

**Turbojets Are in Fuselage by a Vertical Packet** (Fig. 2.27).



**Fig. 2.26. Arrangement of Turbojet in Fuselage, Air Intakes Are in Root Sections of Wing**



**Fig. 2.27. Arrangement of Turbojets in Fuselage by Vertical Packet**

[+] The mid-section of a fuselage is small; hence, its drag is small.

[+] There are no turning moment when one engine is failed (the pitching moment is easier to counteract).

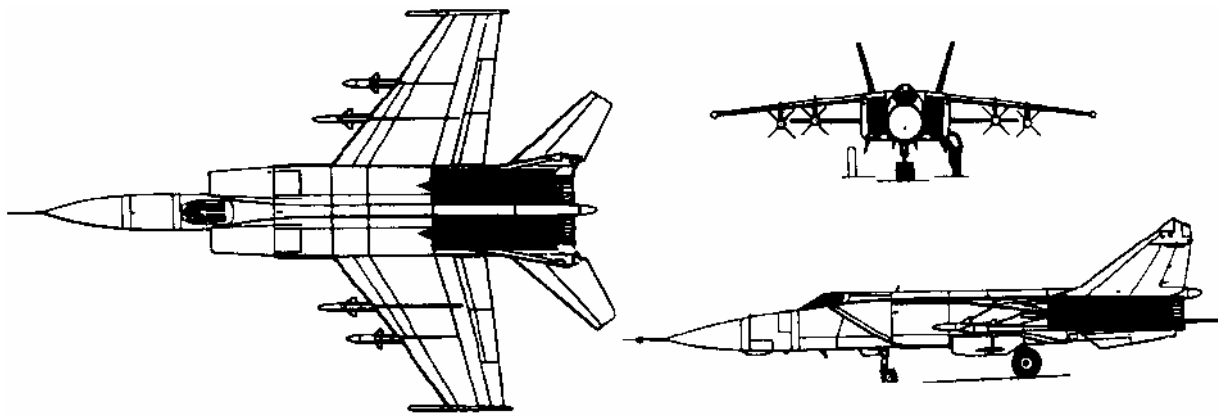
[−] It is difficult to arrange air intake duct in this case.

[−] There are big pressure losses in such air intake duct.

The fighters BAC Lightning and Su-15P are built by such scheme.

**(By-pass) turbojets are in a fuselage by a horizontal packet** (Fig. 2.28).

[+] The mid-section is a little increased in comparison with single-engine scheme. It menaces the small drag of a fuselage.

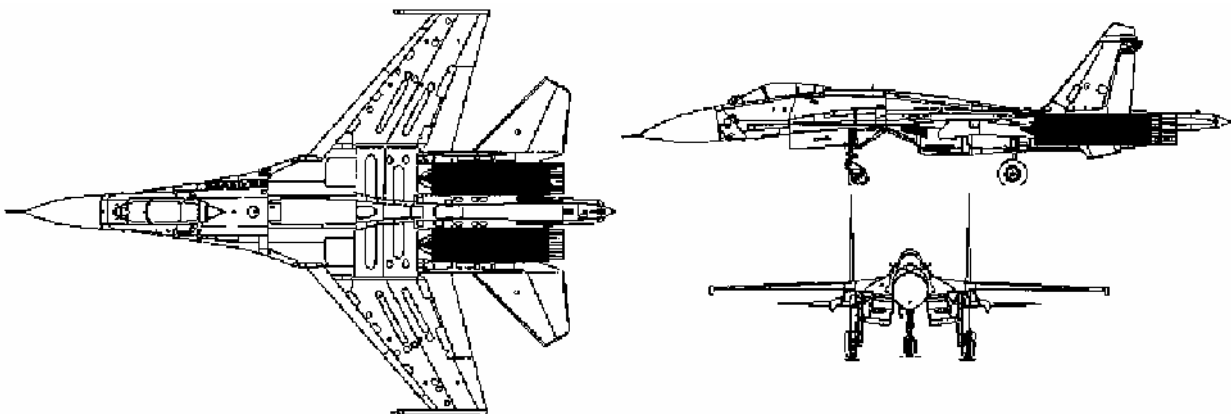


**Fig. 2.28. Arrangement of (By-pass) Turbojets in Fuselage by Horizontal Packet**

- [+] The turning moment under one engine is failed is small.
- [-] Under fire or turbine breakage of one engine, the other is usually damaged.
- [-] The air intake duct takes a payload volume inside a fuselage.
- [-] When a frontal air intake is installed, it has a big length. It causes big pressure losses.

This scheme is one of the most widespread in the world. The fighters MiG-19, MiG-25, SEPECAT Jaguar, and MacDonnell Phantom-II, bombers Tu-22M were built by such scheme.

**(By-pass) turbojets in nacelles under a lifting body (Fig. 2.29).**



**Fig. 2.29. Arrangement of (By-pass) Turbojets in Nacelles under Lifting Body**

- [+] The air intake ducts are short and straight. Hence pressure losses in them are small.
- [+] When one engine is failed, the second usually remains operational.
- [+] It is possible to control direction by difference of thrusts of engines.
- [+] The access at maintenance on ground is convenient.

[+] The body is free to arrange some equipment.

[-] When one engine is failed, the turning moment is more, than at the previous schemes.

[-] It is possible to ingest some foreign matter from ground.

Such arrangement is applied on F-14, MiG-29, Su-27 fighters, Su-34 bomber.

**(By-pass) turbojets are in a root section of a wing** (Fig. 2.30).

[+] The air intake ducts are short and straight. Hence pressure losses are small in them.

[+] External drag is small.

[+] The turning moment when one engine is failed is small.

[+] There is the positive interference of the wing and the fuselage.

[-] For passenger airplanes, there is a strong noise in the passenger compartment.

[-] The jet stream flows near to the fuselage that causes a fatigue failure of the skin.

[-] When breaking the turbine, its blades damage the fuselage.

[-] It is difficult to arrange the high-lift devices in the root section of a wing.

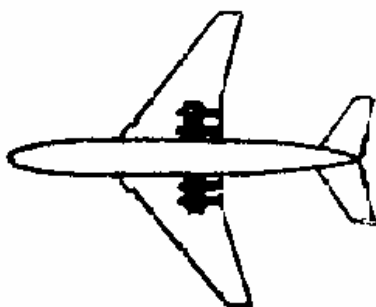
[-] The payload volume for fuel storage in a wing decreases.

[-] It is difficult to arrange the caps and walls of the spars of the wing.

[-] It is difficult to arrange bypass turbojets with high by-pass ratio.

The engines are arranged by such scheme at Comet, Tu-104, Tu-124 passenger airplanes, at Tu-16 bomber, Nimrod anti-submarine aircraft, Su-25, Su-39 attack aircraft.

**Turbojets are under wing without pylons** (Fig. 2.31).



**Fig. 2.30. Arrangement of (By-pass) Turbojets in Root Section of Wing**



**Fig. 2.31. Arrangement of Turbojets under Wing without Pylons**

[+] The engines unload the wing in flight.

[+] The main landing gear can be retracted in an engine nacelle, thus total drag of an airplane is decreased.

[+] The engines are mounted directly to a wing without a pylon. Thus

the mass of engine mounts decreases.

[−] The interference of the wing and the engine nacelle results in reduction of the critical Mach number. Therefore such scheme is not applied on transonic and supersonic airplanes.

[−] The turning moment when one engine is failed is more, than in the previous schemes.

Yak-25, Yak-28 fighters, Il-28, Il-46 bombers were built by such scheme.

**(By-pass) turbojets are in a tail part of a fuselage with pylons** (Fig. 2.32).

[+] The wing is aerodynamically clean.

[+] The noise in the cabin is small.

[+] The turning moment, when one engine is failed, is small.

[+] There is a good access to engine at maintenance.

[+] The wing shields engines from ingestion of foreign matter from ground.

[+] It is simple to arrange the devices of thrust reverse.

[−] Engines do not unload the wing; hence, the wing mass increases.

[−] Engines load the tail part of the fuselage, hence, the fuselage mass increases.

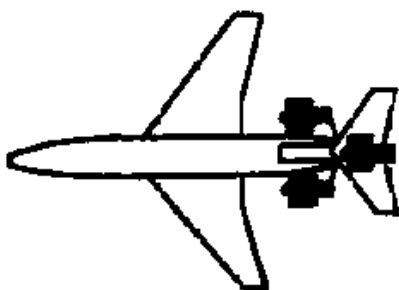
[−] Fuel pipelines pass through the fuselage; thus fire danger increases.

[−] It is difficult to center-of-gravity of an empty airplane. Sometimes the fourth landing gear strut in tail part is required (Il-62).

[−] The extended nose part impairs performance of longitudinal and directional stability of an airplane.

Now the most of passenger low-wing monoplanes for small number of passengers (executive airplane) with (bypass) turbojets is designed by such scheme. In past, some mid passenger airplanes (Caravel, Tu-134, B-727) were built by such scheme. The A-10 attack aircraft is built by such scheme too.

**(Bypass) turbojets are under the wing with pylons** (Fig. 2.33).



**Fig. 2.32. Arrangement of Turbojets in Tail Part of Fuselage with Pylons**

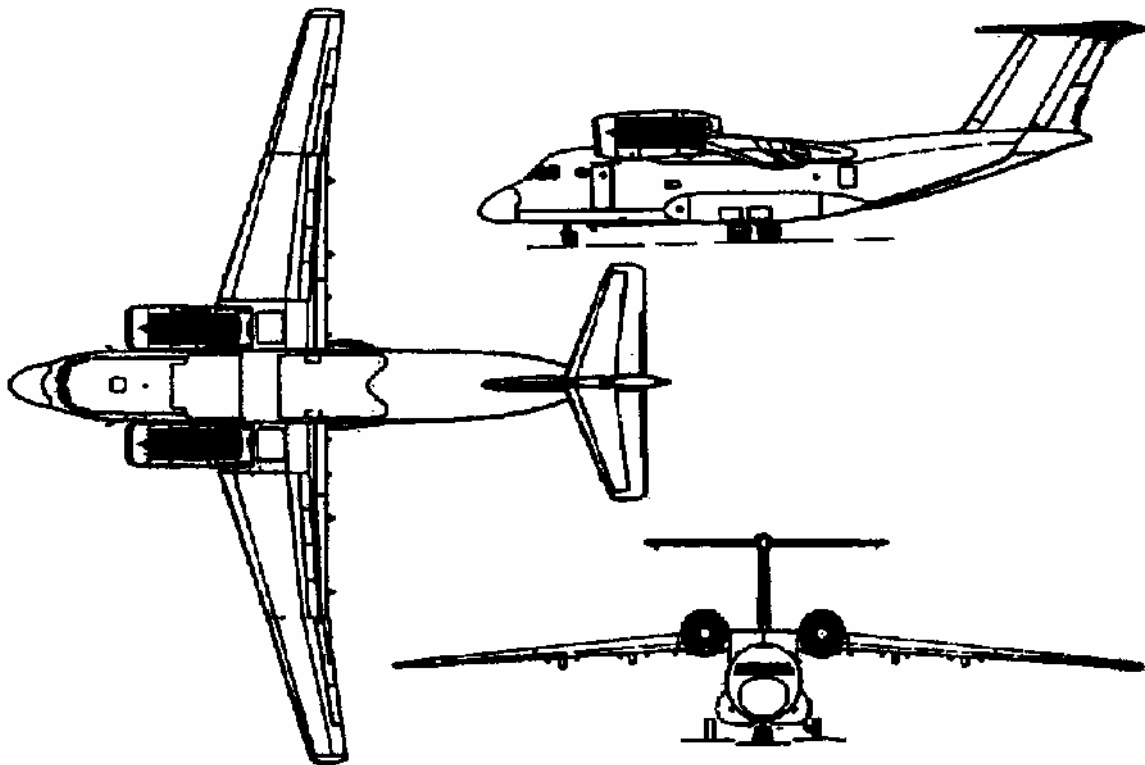


**Fig. 2.33. Arrangement of (Bypass) Turbojets under Wing with Pylons**

- [+] The engines unload a wing in flight.
  - [+] The engines serve as anti-flutter mass.
  - [+] If a height above ground is small, than access for engine maintenance is good.
  - [+] There is low fire danger.
  - [+] It is simple to arrange the device of thrust reverse.
  - [-] The turning moment, when one engine is failed, is big.
  - [-] For low-wing monoplanes, it is possible to ingest foreign matter into the engines.
  - [-] The drag of an airplane increases due to engine nacelles.
  - [-] The mass of an airplane increases due to engine nacelles.
- Now practically all passenger and transports airplanes with (bypass) turbojets are built by such scheme.

**(Bypass) turbojets are above the wing** (Fig. 2.34).

- [+] The engines unload the wing in flight.
- [+] The engines serve as anti-flutter mass.
- [+] The Coand effect (supercirculation) can be realized here. It allows significantly improve take-off and landing characteristics of an airplane.
- [+] The jet stream is shielded from the observation from below by the wing. Thus contact of missiles with thermal target seeker is made difficult.



**Fig. 2.34. Arrangement of (Bypass) Turbojets above Wing**

[+] The engines are arranged highly above a runway. It prevents from ingestion of foreign matter into them.

[-] Access to engines at maintenance is difficult.

[-] The turning moment when one engine is failed is big.

The An-74 STOL airplane is serially built by such scheme.

### 2.5.5. Heavy supersonic airplanes

At heavy supersonic airplanes the engines are usually arranged in nacelles underneath the wing or a lifting body. There are three versions of their arrangement (Fig. 2.35).

**All engines are in one packet** (Fig. 2.35, a).

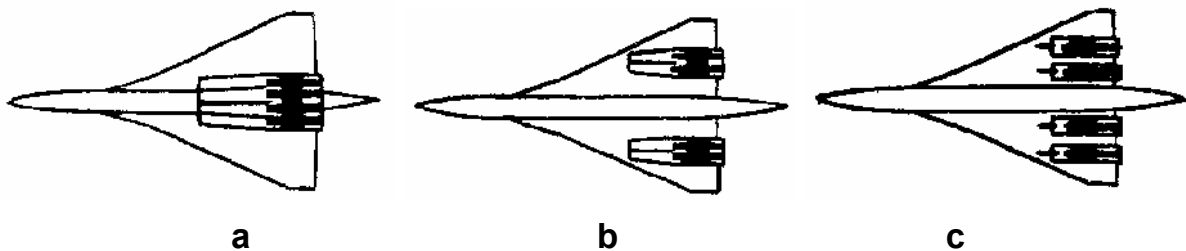
[+] Effect from compression under a fuselage is the greatest.

[+] Turning moment, when one engine is failed, is the smallest.

[-] Air intake is long; hence the pressure losses are big.

[-] Wing is not unloaded, hence, the mass of the wing increases.

The majority of heavy supersonic aircraft (Tu-144 experimental, T-4, XB-70) had been built and designed by such scheme.



**Fig. 2.35. Versions of Arrangement of Engines at Heavy Supersonic Airplanes**

**The engines are in packets by pairs** (Fig. 2.35, b).

[+] Fire danger is smaller.

[+] Length of the air intakes is smaller.

[-] Positive effect from compression is less.

[-] Turning moment is more.

Both supersonic airliners Tu-144 and Concorde were serially built by such scheme.

**The engines are in personal nacelles** (Fig. 2.35, c).

[+] Fire danger is minimal.

[+] Length of air intakes is minimal.

[+] Engines unload the wing in flight. Hence the wing has smaller weight.

[-] Turning moment, when one engine is failed, is the biggest.

[-] Positive effect from compression is minimal.

## Theme 3. NACELLES AND COWLINGS OF ENGINES. ENGINE MOUNTS

### 3.1. Nacelles and Cowlings of Engines

#### 3.1.1. Purpose and Components of Nacelles and Cowlings

The nacelle is **intended** to reduce aerodynamic drag and to organize airflow ensuring cooling of engine.

**Requirements** to engine nacelles are:

1. Minimal aerodynamic drag in a system of the aircraft (at working engine or inoperative one; in view of an aerodynamic interference).
2. Rational organization of airflow for cooling an engine (propeller gear, cooler installations).
3. Good access to power plant at maintenance on ground.
4. Provision of fire safety (firewalls for fire isolation).

**Components** of a nacelle. Cowlings.

The nacelle usually consists of a body and cowlings. Airframe of cowlings can be frame and panel.

In the **frame** schemes, strength and stiffness are formed by a framework. Removable covers with thin skin, reinforced by load-carrying structure, are mounted to this framework.

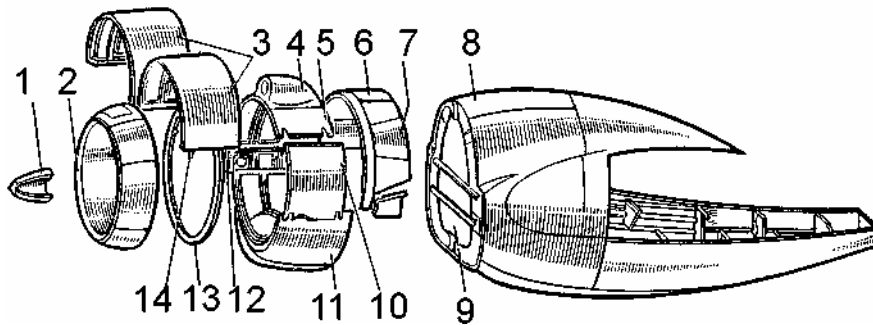
The **panel** cowling consists of stiff panels connected among themselves by locks and forming the closed load-carrying shell.

Loads from a cowling are transmitted to an engine, and through engine mounts to an aircraft. Simultaneous attachment of the cowling to the engine and to the aircraft is prohibited. Stiff covers of panels are made not removable, but hinging — with attachment fittings. In an open position, special braces support the cover. For the best fit of the covers in the closed position there are pressure-sealing parts on joints of panels.

#### 3.1.2. Airframe of Nacelles for Various Types of Engines

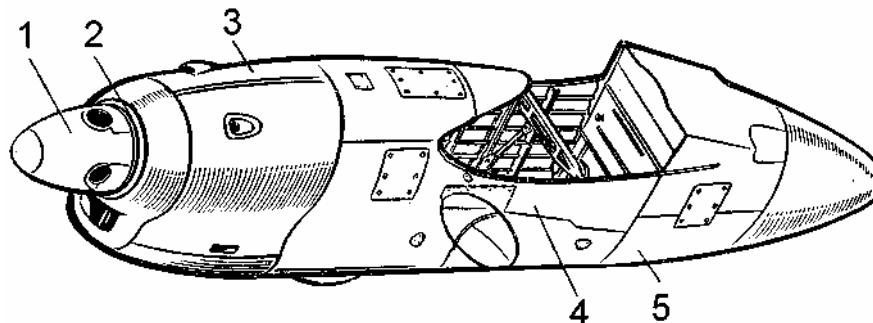
Nacelle of **piston engines** (Fig. 3.1) consists of spinner, cowling and casing. The cowling is mounted to engine and consists of air intake and hinging parts. Load-carrying elements in the casing of the nacelle carry loadings from engine mounts and transmit them to an aircraft.

The nacelle of a **turboprop** (Fig. 3.2) consists of spinner 1, cowling 3, casing 4 and tail part 5. The spinner serves for decreasing aerodynamic drag, shaping of the air intake duct and protection of the propeller bush from external action.



**Fig. 3.1. Nacelle of Piston Engine:**

1 — Propeller Spinner; 2 — Forward Ring of Cowling; 3, 5, 10, 11 — Cowl Panel of the Nacelle; 4 — Air Intake of Carburetor; 6, 7 — Cowl Flaps; 8 — Casing; 9 — Firewall; 12 — Side Panel; 13 — Frame of the Nacelle; 14 — Lock

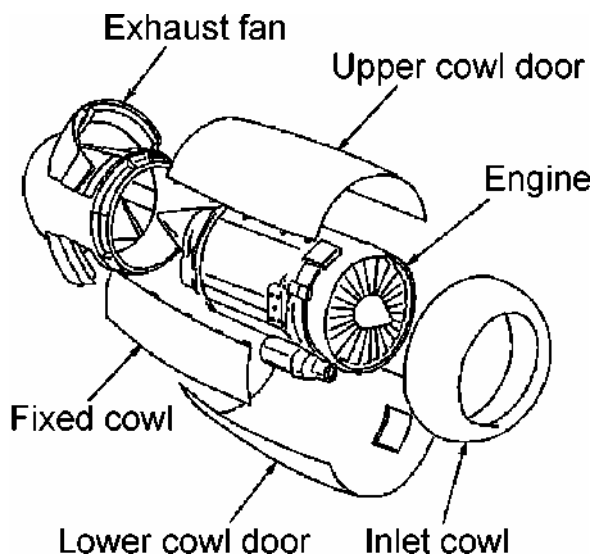


**Fig. 3.2. Nacelle of Turboprop:**

1 — Spinner; 2 — Fairing Of Propeller Gear; 3 — Cowling; 4 — Casing; 5 — Tail Part

The internal part of the air intake forms external outline of the air passage. The skin of the air intake lip is usually designed double. The annular slot between these skins forms the chamber of the deicing system of air intake. Hot air is supplied to this chamber from the compressor of engine.

Nacelles of **turbojets** (Fig. 3.3) of up-to-date airplanes as a rule are attached to the airframe through a pylon (under wing or in the tail part of a fuselage). Actually the nacelle usually consists of one-piece nose inlet cowl with the air intake, tail part with devices of thrust reverse and hinging cowling panels in the middle part.



**Fig. 3.3. Nacelle of Turbojet**



## 3.2. Engine Mounts

### 3.2.1. Engine Mounts. General Information

The engine mounts **is intended** to attach an engine with the units and equipment installed on it to attachment fittings of an airframe.

From the point of view of structural mechanics engine mounts are subdivided into truss, beam, truss-girder and frame. We should note that, in many cases truss engine mounts are very original. For example (bypass) turbojets are frequently attached to airframe by separate rods. Such system provides geometrical stability only as a whole.

Engine mounts are usually designed as statically indeterminate systems. It promotes reliability and survivability of engine mounts. However rigging of engine mounts becomes complicated.

It is necessary to keep in mind that engine is attached to engine mounts by brackets or trunnions. They are arranged on the engine in the strongest and the most stiff places.

(Bypass) turbojets and turboprops are usually attached to engine mounts at two (seldom at three) supporting locations. Afterburners and extension pipes are movably attached by the additional supporting locations.

When designing the engine mounts, it is necessary to provide access to units of the engine (and to the equipment installed on it) for maintenance. It is also required to provide a fast engine replacement.

**Requirements** to engine mounts are the following.

1. Carry all loadings from engine with the units and the equipment installed on it under any allowable operation conditions. Thus casing of the engine must not include in airframe of aircraft.
2. Provide specified strength and stiffness at minimal mass.
3. Absorb the vibration from engine and propeller that they were not transmitted to an airframe of aircraft.
4. Compensate for temperature deformations of engine parts without any additional load in these parts and in aircraft airframe.
5. Maintain high survivability and specified service life.
6. Aerodynamic drag should be minimal.
7. Maintainability (little duration of installation and removal of engine; ease of rigging; good access at maintenance.)

### 3.2.2. Loadings

Let's consider loadings which load engine mounts while in service. They are mass and aerodynamic forces, thrust force, reaction torque from a propeller, loadings from a device of thrust reverse. Moreover in case of

changing of aircraft motion direction, gyroscopic moment loads the engine mounts.

The mass forces are calculated by **power plant mass**. The power plant mass includes: mass of engine and mass of all equipment, cowlings, propellers and other units attached to the engine.

Approximately power plant mass can be determined as follows:

for airplanes with (bypass) turbojets  $m_{p,p} = (1.2...1.6)m_{en}$ ;

for airplanes with turboprops  $m_{p,p} = (1.9...2.2)m_{en}$ ,

where  $m_{en}$  — mass of the engine.

The **gyroscopic moment** is determined by formula

$$M_G = J_x \omega_x \omega_i \sin(\overline{\omega}_x, \overline{\omega}_i),$$

where  $J_x$  — mass moment of inertia of rotated parts of the engine,  $\text{kg}\cdot\text{m}^2$ ;  $\omega_x$  — angular velocity of rotated parts of the engine,  $\text{s}^{-1}$ ;  $\omega_i$  — angular velocity of rotation of the aircraft around of the  $i$  axis,  $\text{s}^{-1}$ . For turboprops, by-pass turbojets and geared piston engines, the adduced moment of inertia and adduced angular velocity of rotating parts of the engine are used.

The gyroscopic moment is directed in such a way that the axis of rotating parts of the engine should be turned to the  $i$  axis in the shortest way.

The **reaction torque from a propeller** of piston engine or turboprop acts in the direction opposite to rotation of a propeller and is equal to, N·m:

$$M_p = \frac{N}{\omega_p} = \frac{60N}{2\pi \cdot n_p},$$

where  $N$  — shaft power of a propeller, W;  $\omega_p$  — angular velocity of rotation of a propeller,  $\text{s}^{-1}$ ;  $n_p$  — number of revolutions of propeller per minute.

In case of installation of coaxial contra-rotating propellers their reaction torques are subtracted. Taking into account nonuniformity of power distribution between propellers, it is usually assumed:

$$M_p = (0.1...0.2) \frac{60N}{2\pi \cdot n_p}.$$

**Thrust force of piston engines or turboprops** is determined by formula, N:

$$T = \frac{N_e \eta_p}{V_H},$$

where  $N_e$  — effective power of the engine, W;  $\eta_e$  — efficiency of the propeller;  $V_H$  — flight speed, m/s.

Efficiency of the propeller depends on the flight speed in a complicated manner. Therefore within the course project, it is possible to use the

approximate relation for take off thrust:

$$T \approx (17.4...20.4)N_e.$$

The **aerodynamic forces**, acting to a cowling of engine, are determined by wind-tunnel tests.

The **mass forces** are applied in a center of mass of engine. These mass forces are gravity force and inertial forces. These inertial forces appear in case of changes of flight direction and aircraft attitude. At strength analysis, they usually use the coordinate system fixed in aircraft (the body axes). Origin of this coordinate system is fixed to center of mass of aircraft. The **0x** axis is directed forward along the axis of the aircraft. The **0y** axis lies in a plane of symmetry of aircraft perpendicularly to **0x**. The **0z** axis is directed by right-hand triple (as a vector product of units of **0x** and **0y** axes).

### 3.2.3. Critical Load Conditions

Forces and moments loading to engine mounts in run of aircraft, can assume various values. Therefore there are normative papers, such as NLGS-3, AP, FAR, JAR. A number of positions of aircraft, corresponding to the heaviest loading conditions are specified in them. These cases are called **critical load conditions**. Selection of the normative paper is determined by the technical requirements to the designed aircraft.

These critical load conditions are studied in detail in the course «Strength analyses of aircraft». Here, we will consider only supplementary conditions for engine mounts according to AP-25.

#### **Engine and APU torque.**

(a) Each engine mount, APU and its supporting structure must be designed for the effects of

(1) A limit engine or APU torque corresponding to takeoff power and propeller speed, acting simultaneously with 75 percent of the limit loads from flight condition 1;

(2) A limit torque corresponding to the maximum continuous power and propeller speed, acting simultaneously with the limit loads from flight condition 1;

(3) For turboprop installations, in addition, a limit engine torque corresponding to takeoff power and propeller speed, multiplied by a factor accounting for propeller control system malfunction, including quick feathering, acting simultaneously with 1g level flight loads. In the absence of a rational analysis, a factor of 1.6 must be used.

(b) For turbine engine and APU installations, the engine mounts and supporting structure must be designed to withstand each of the following:

(1) A limit engine torque load (considering as limit), imposed by:

(i) sudden engine or APU stoppage due to malfunction, which can be a temporary loss of power or thrust and which can cause stoppage as a result of vibrations;

(ii) limit angular acceleration of engine or APU.

(2) A limit engine torque load (considering as ultimate), imposed by stoppage of engine or APU due to structural failure (such as compressor jamming).

(c) This limit engine torque must be obtained by multiplying mean torque for the specified power and speed by a factor of:

(1) 1.25 for turboprop installations;

(2) 1.33 for piston engines with five or more cylinders;

(3) 2, 3, or 4, for engines with four, three, or two cylinders, respectively.

(A) Applying (a) to turbojets, limit torque must be equal to the torque appearing at limit angular acceleration of engine rotating parts.

#### **Side load on engine mount.**

(a) Each engine mount and its supporting structure must be designed for a limit load factor in a lateral direction, for the side load on the engine mount, at least equal to the maximum load factor obtained in the yawing conditions but not less than 1.33.

(b) This side load may be assumed to be independent of other flight conditions.

(A) When engine is mounted to a wing, side load directed from axis of airplane must be not less than

$$P_z = \omega_x^2 r \cdot m_{p,p},$$

where  $m_{p,p}$  — power plant mass;  $\omega_x$  — limit roll angular velocity;  $r$  — distance from engine center of gravity to longitudinal axis of airplane in plane view.

(B) Simultaneous action of above specified side load and weight of engine must be considered too.

**Critical condition 1** corresponds to curved flight with limit lift coefficient and limit load factor.

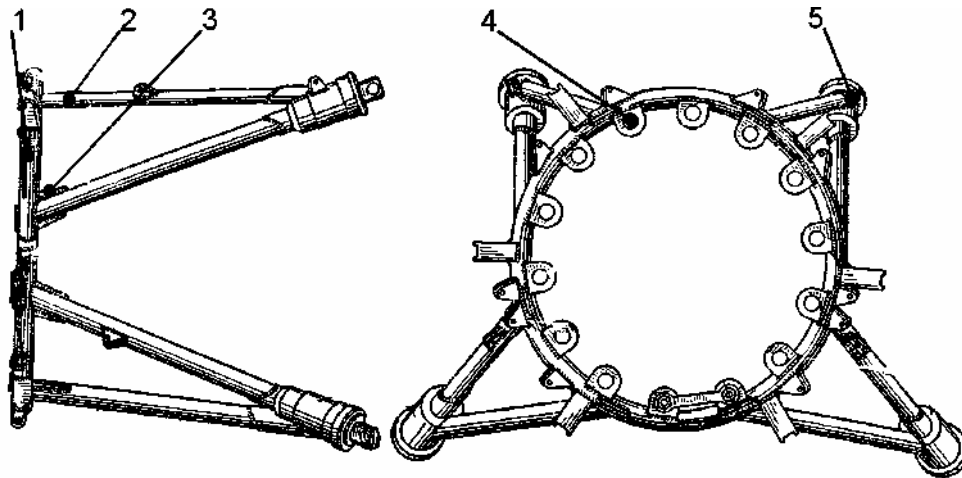
### **3.3. Airframes of Engine Mounts**

#### **3.3.1. Engine Mounts for Piston Engines**

**Attachment of radial piston engine** is usually performed by frame (Fig. 3.4). This frame consists of 6...8 beams and a load-carrying ring with attachment fittings for the engine. Shock absorbers are installed in these fittings. In addition the other shock absorbers can be installed in attach-

ment fittings to the airplane.

In designing calculations, for simplification, this frame is considered as a truss. That is the beams with anchorage at its ends are considered as rods with hinges at its ends.

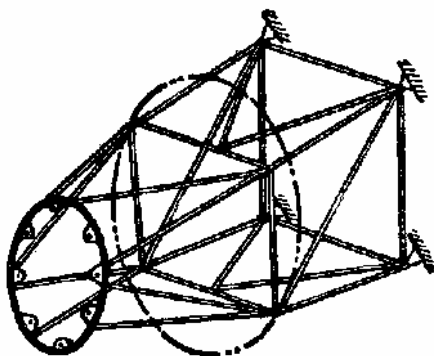


**Fig. 3.4. Engine mounts of radial piston engine**

The engine mount of radial piston engine is shown in Fig. 3.4. It consists of the load-carrying ring 1 with the attachment fittings 4 of the engine. In these attachment fittings shock absorbers are installed. The rods 2 of the frame are attached to the load-carrying ring and among themselves by welding with gusset plates 3. In the attachment fittings 5 of the frame to the airframe, the eyes are connected to rods. This system is twice statically indeterminate.

The engine mounts of piston engine is usually made of high-strength alloyed steels. To provide the fatigue life, they use the moderated hardening ( $\sigma_b = 1100...1200$  [MPa]).

Rigging of such system is carried out in attachment fittings to the airframe. Temperature deformations are compensated by shock absorbers. Strength analysis is performed by known methods of structural mechanics.



**Fig. 3.5. Scheme of Double-span Engine Mounts**

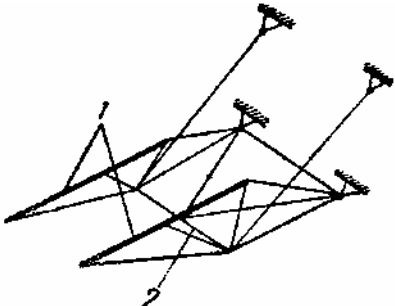
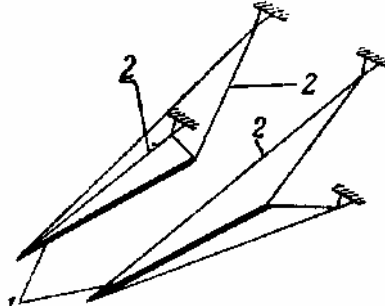


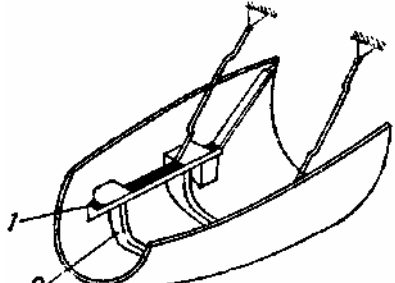
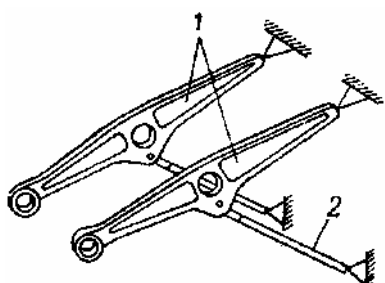
In practice, multispan engine mounts are met (Fig. 3.5). This is due to the fact that conditions of engine arrangement, minimization of the aerodynamic drag and optimality of angles of the truss  $\alpha \approx (30...45)^\circ$ .

**Engine mounts of in-line piston engines** (Fig. 3.6) are classified according to:

- 1) Airframe: truss, truss-girder, beam and frame;

2) Transmission of side loading: with a crosstie (the casing of engine does not transmit any side loading) and without a crosstie (the casing of engine transmits side loading);

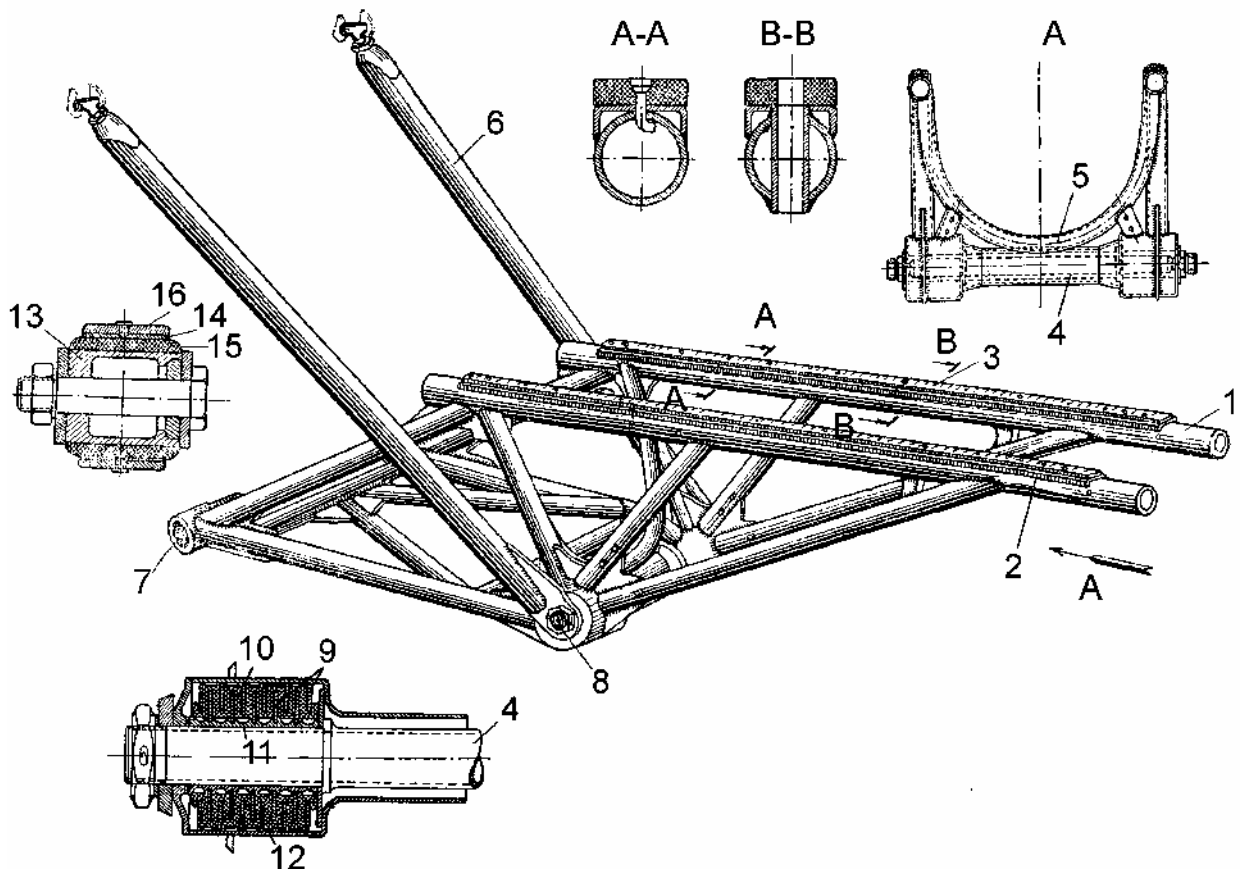
3) Type of attachment: concentrated attachment and distributed attachment.

Airframe	Airframes with a crosstie	Airframes without any crosstie
truss		
beam		
truss-girder		

**Fig. 3.6. Classification of Engine Mounts of In-line Piston Engines:  
1 — Engine-Support Beam; 2 — Rod**

In-line piston engines are attached to the engine-support beams 1, 2, Fig. 3.7. The fiber plates 3 are arranged between them and the engine. They serve for a fit of engine to all surface of the engine-support beam; vibration damping; decrease of effect of thermal stresses. The crosstie 4, 5 prevents the engine from side loads. This structure is attached to airplane by two rods 6 and two hinges 7. Therefore it is truss-girder. There are two shock absorbers in the nodes 8. These shock absorbers consist of a set of steel rings 9 and rubber rings 10 vulcanized together. They are separated with the steel rings 11 and 12.

Bush 13, steel ring 14 and rubber ring 15 are vulcanized together and installed into holder 16. They serve for vibration damping too.

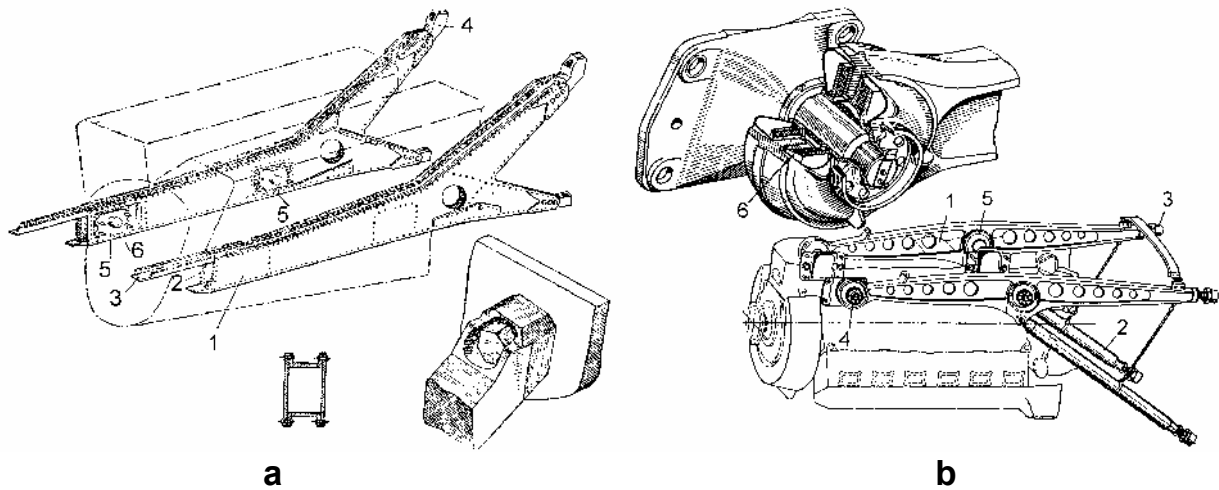


**Fig. 3.7. Truss-Girder Engine Mounts of In-line Piston Engine**

Engine mounts of in-line piston engines usually consist of more than six rods. Sometimes, by conditions of arrangement, some rods are designed with a breaking of its axis. These rods must be reinforced in places of breaking. Holes for attachment of engines in the engine-support beam are sometimes made oval shaped. It allows an engine to be extended when being heated. Rigging of such systems is provided by regulated struts 6.

In truss-girder and beam airframes (Fig. 3.8), the beams are usually not connected among themselves. Thus casing carries side loadings. Engine is attached at four points by trunnions. Due to little stiffness of the forward fittings, all forces (except of  $M_z$ ) are carried by the rear (main) fittings. This fitting is arranged near a center of mass of engine. The moment  $M_z$  is jointly carried by forward (auxiliary) and rear fittings.

The beam structure can be assembled or stamped. Beams 1 of assembled structure (Fig. 3.8, a) are made of aluminium alloys. They consist of caps 3, webs 2 and reinforcements 5. The reinforcements must be in places of trunnions 6. The stamped beams 1 (Fig. 3.8, b) can be made of a magnesium alloy. In truss-girder airframe (Fig. 3.8, b) in addition to the beam 1 there are rods 2. These rods should not prevent engine from thermal expansions.

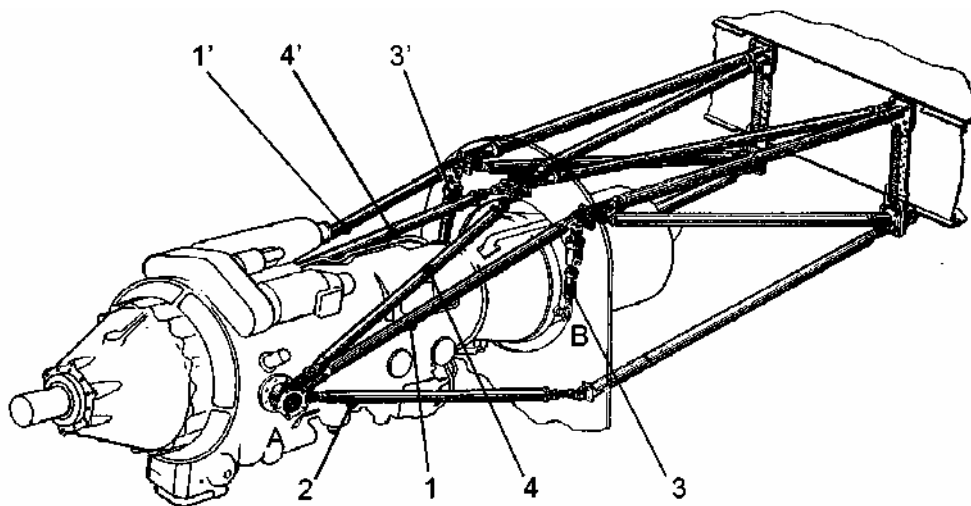


**Fig. 3.8. Beam and Truss-Girder Engine Mounts of In-Line Piston Engines**

### 3.3.2. Engine Mounts of Turboprops

Attachment of turboprops at airplanes is usually performed by three-dimensional frame structures. Its airframe can be truss (An-10, An-24) or truss-girder (Il-18).

The engine mounts of turboprops (Fig. 3.9) consists of eight rods. They are made of high-strength alloyed steel. The rods are provided with regulated ends and check holes. These check holes serve to control minimal required number of turn of thread. As  $8-6=2$ , the structure is twice statically indeterminate.



**Fig. 3.9. Truss Engine Mounts of Turboprop**

Let's consider loading of this engine mounts. Using symmetry and considering symmetrical and antisymmetrical loadings separately, it is possible to reduce the problem to once statically indeterminate. (When symmetrical loads act, the forces in symmetrical rods are equal, and, when an-



tisymmetrical loads act, the forces in symmetrical rods are equal in value and have opposite sign.)

Vertical loads are approximately distributed in inverse proportion to distances from the center of mass to the supporting locations. Further, considering equilibrium of nodes A and B, it is possible to see, that external loadings are carried as follows:

$F_x$	A	1, 1', 2, 2'	$M_x$	A+B	1, 1', 2, 2', 3, 3'
$F_y$	A+B	1, 1', 2, 2', 3, 3'	$M_y$	A	1, 1', 2, 2'
$F_z$	A	1, 1', 2, 2', 4, 4'	$M_z$	A+B	1, 1', 2, 2', 3, 3'

In truss-girder airframe (Fig. 3.10), elements 5 are designed as beams. It allows: first, to reduce length of rods 6. Second, a beam usually carries compressive stresses better than a rod. It is necessary to note, that in this case the beam carries bending stress only from action of side loads.

### 3.3.3. Engine Mounts of (Bypass) Turbojets

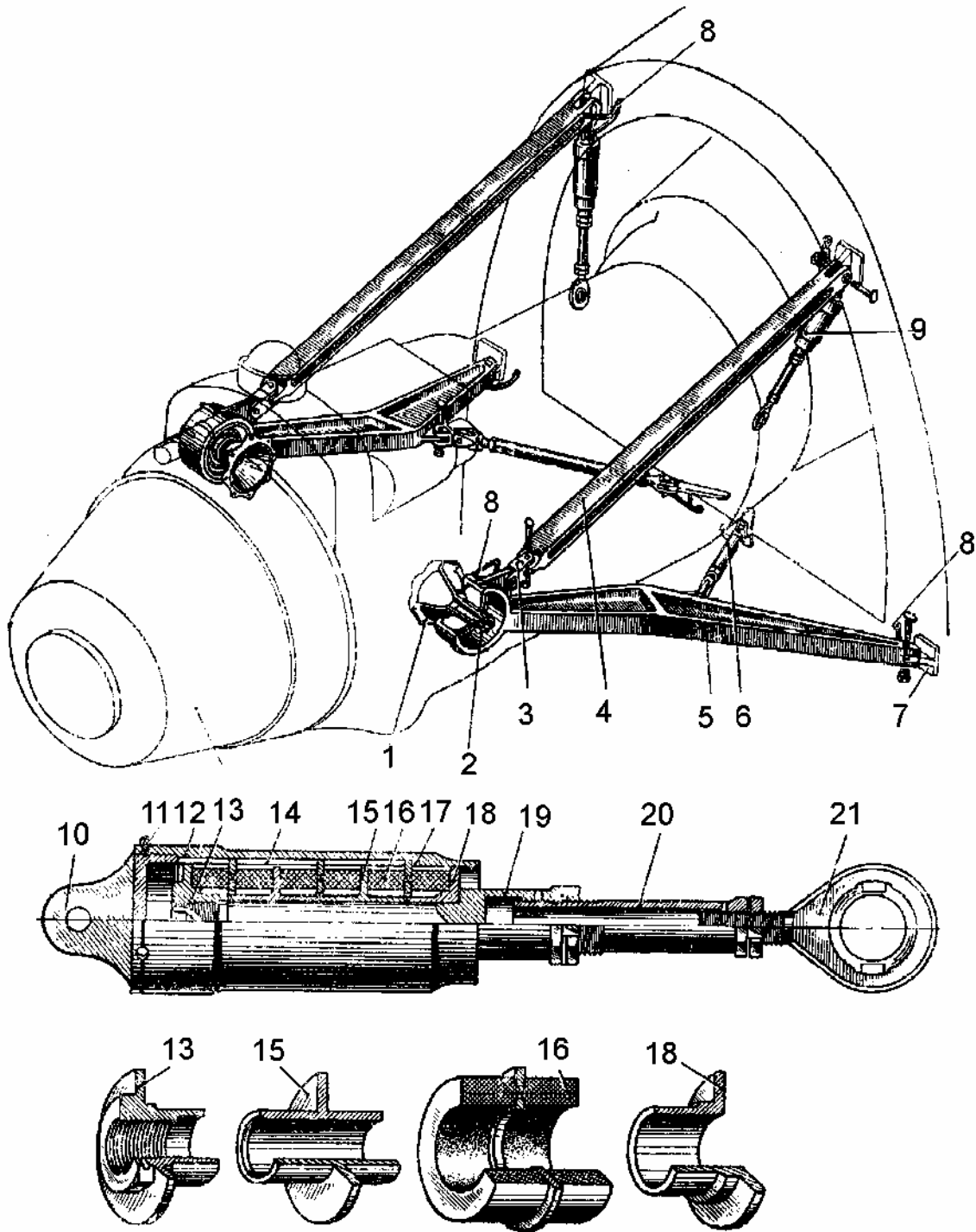
Depending on engine arrangement at an airplane, engine mounts can strongly differ. Engine mounts of (bypass) turbojets are usually designed by truss or truss-girder airframe. It is necessary to note, that trusses in this case usually represent a set of separate rods. As a whole they provide geometrical stability of the engine mounts.

**Engine mounts of (bypass) turbojets inside a fuselage.** In Fig. 3.11 the engine mounts of turbojet with centrifugal compressor, arranged inside a fuselage, is shown.

The engine is attached at two trunnions in nodes A and in two hinged nodes B. The structure consists of eight rods and has  $8 - 6 = 2$  redundant members. As the system is symmetric, the extent of static indefinability can be reduced up to one. Considering loading the structure, we see, that external loads are carried as follows:

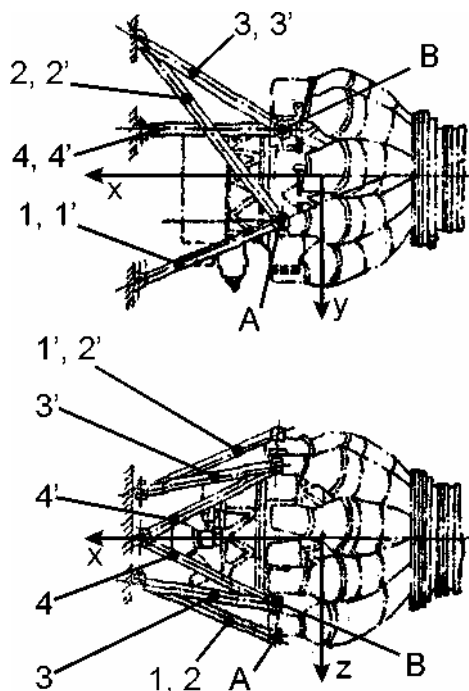
$F_x$	A	1, 1', 2, 2', 3, 3'	$M_x$	A	1, 1', 2, 2'
$F_y$	A	1, 1', 2, 2', 3, 3'	$M_y$	A	1, 1', 2, 2'
$F_z$	B	1, 1', 2, 2', 3, 3', 4, 4'	$M_z$	A, B	1, 1', 2, 2', 3, 3'

It is necessary to pay attention to design realization of the attachment of rods 4, 4' and nodes A. The attachment of rods 4, 4' (Fig. 3.12) transmits to fuselage the loadings, lying in a plane of the load-carrying frame ( $F_z$ ), and does not transmit loading normal to this plane. In the nodes A (Fig. 3.13) the eccentric split spherical bushings are located. They allow to execute rigging of engine at the airplane.

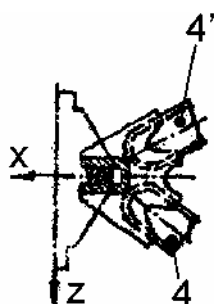


**Fig. 3.10. Truss-Girder Airframe Engine Mounts of Turboprop:**  
 1 — Forward Trunnion; 2 — Forward Shock Absorber; 3 — Link;  
 4 — Upper Supporting Strut (Rod); 5 — Beam; 6 — Internal Supporting  
 Strut (Rod); 7 — Bracket; 8 — Jumper; 9 — Rear Supporting Strut (Rod)  
 With Shock Absorber; 10 — Fork; 11 — Retaining Screw; 12 — Casing;  
 13 — Nut; 14, 15, 18 — Bushings; 16 — Disk Shock Absorber; 17 — Ring;  
 19 — Central Rod; 20 — Transitional Bushing; 21 — Eye

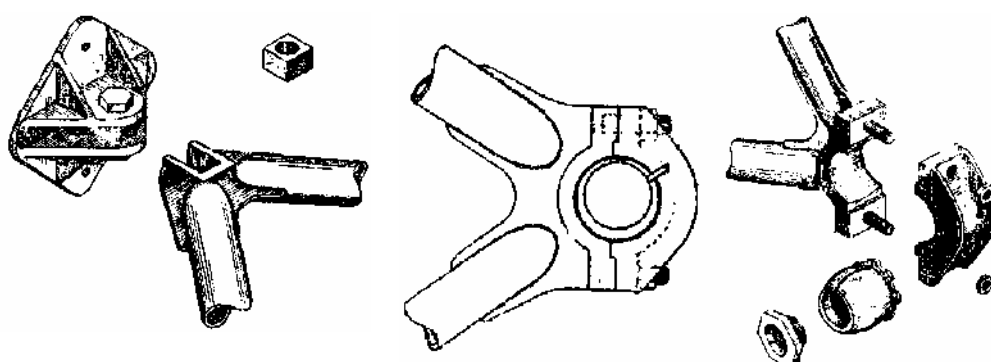
All these rods are made of high-strength alloyed steel as pipes. Installation and removal of the engine are performed when the tail part of fuselage is detached.



**Fig. 3.11. Engine Mounts of Turbojet with Centrifugal Compressor inside Fuselage**



**Fig. 3.12. Design of Attachment of Rods 4, 4' (Fig. 3.11)**



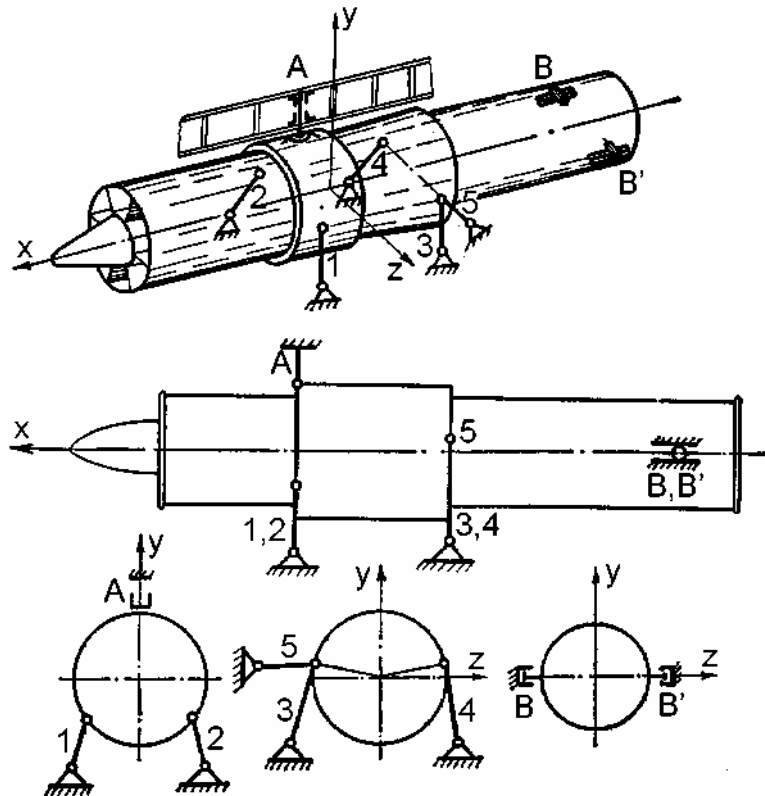
**Fig. 3.13. Design of Nodes A (Fig. 3.11)**

In Fig. 3.14 the engine mounts of **turbojet with the axial-flow compressor** is shown. Here the engine is attached at two supporting locations to load-carrying frames.

The node A carries thrust and is loaded in addition by side loads. The rod 5 takes part in carrying side loads.

The load-carrying element A represents a pin, Fig. 3.15. When mounting the engine, the lower part of this pin enters into the spherical bearing installed on the engine. Such way of attachment in the node A provides some translation along the vertical line and turn at rigging of en-

gine and at its thermal expansion. This pin is attached to a longitudinal beam of fuselage in two nodes. The pin is attached to the top cap of the beam with the steel bolt. The pin is pressed to the lower cap of the beam with a steel wedge. Thus the beam carries out thrust force.



**Fig. 3.14. Engine Mounts of Turbojet with Axial-flow Compressor inside Fuselage**

Considering loading the structure, we see, that the external loads are carried as follows:

$F_x$	I, II	A, 1, 2, 3, 4	$M_x$	I, II	A, 1, 2, 3, 4
$F_y$	I, II	1, 2, 3, 4	$M_y$	I, II	A, 1, 2, 3, 4, 5
$F_z$	I, II	A, 1, 2, 3, 4, 5	$M_z$	I, II	1, 2, 3, 4

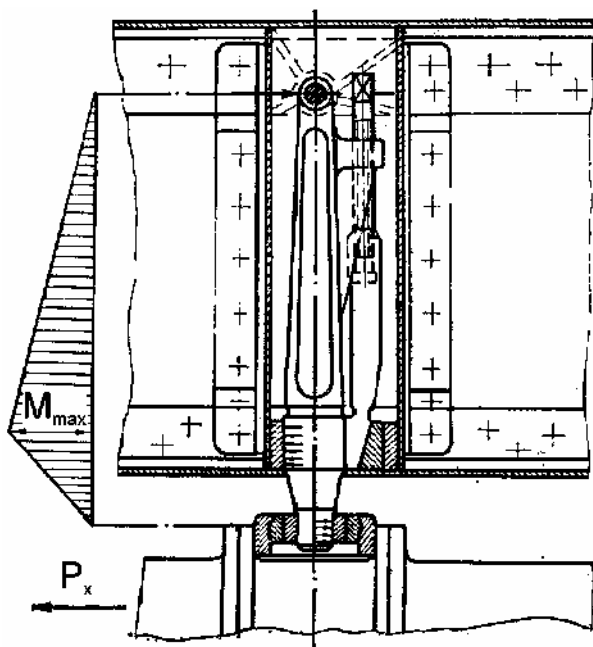
All rods of the structure are made of high-strength alloyed steel. They are attached to the brackets of engine and to the brackets of load-carrying frames with bolts. The rods 3, 4, 5, arranged at the second supporting location, are regulated. It allows changing their length at rigging of engine.

The engine is installed and removed when the tail part of a fuselage is detached with a special hand cart. To simplify mounting, casing of the engine carries on the brackets with rollers. In the fuselage there are directing rails. By these rails the engine is rolled inside of the fuselage, and then attached to engine mounts.

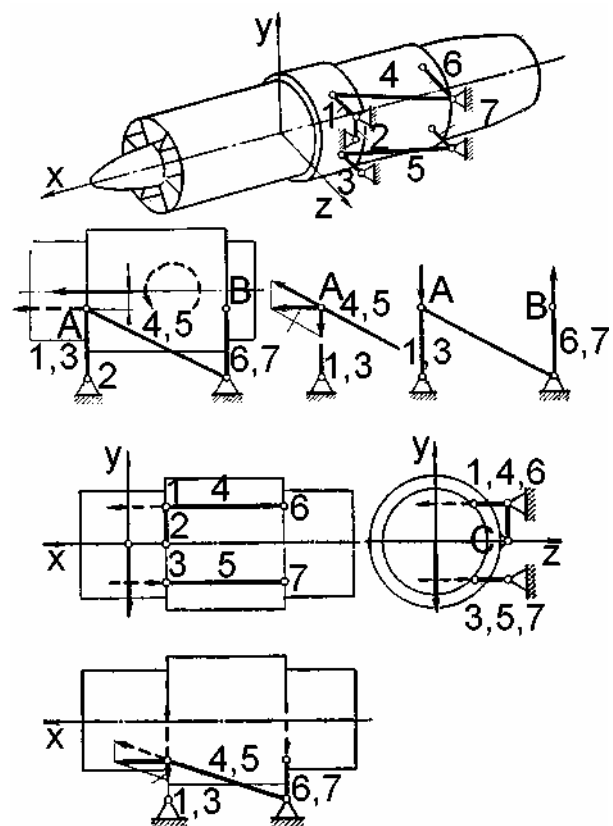
In case of arrangement of two engines side-by-side in the tail part of a fuselage each of them is attached to load-carrying frames and beams of a fuselage similarly. Thus the pin, carrying the engine thrust, can be placed either from above or sideways of engine, depending on convenience of arrangement.

**Engine mounts of (bypass) turbojets to a fuselage.** We shall consider the engine mounts shown in Fig. 3.16.

In this case engines are arranged directly behind the wing torsion box closely to the fuselage. They are attached at two supporting locations to the load-carrying frames of the fuselage. The structure is formed by seven rods. Thus, it has one redundant member (extent of static indefinability).



**Fig. 3.15. Node A**



**Fig. 3.16. Engine Mounts of Bypass Turbojet to Fuselage**

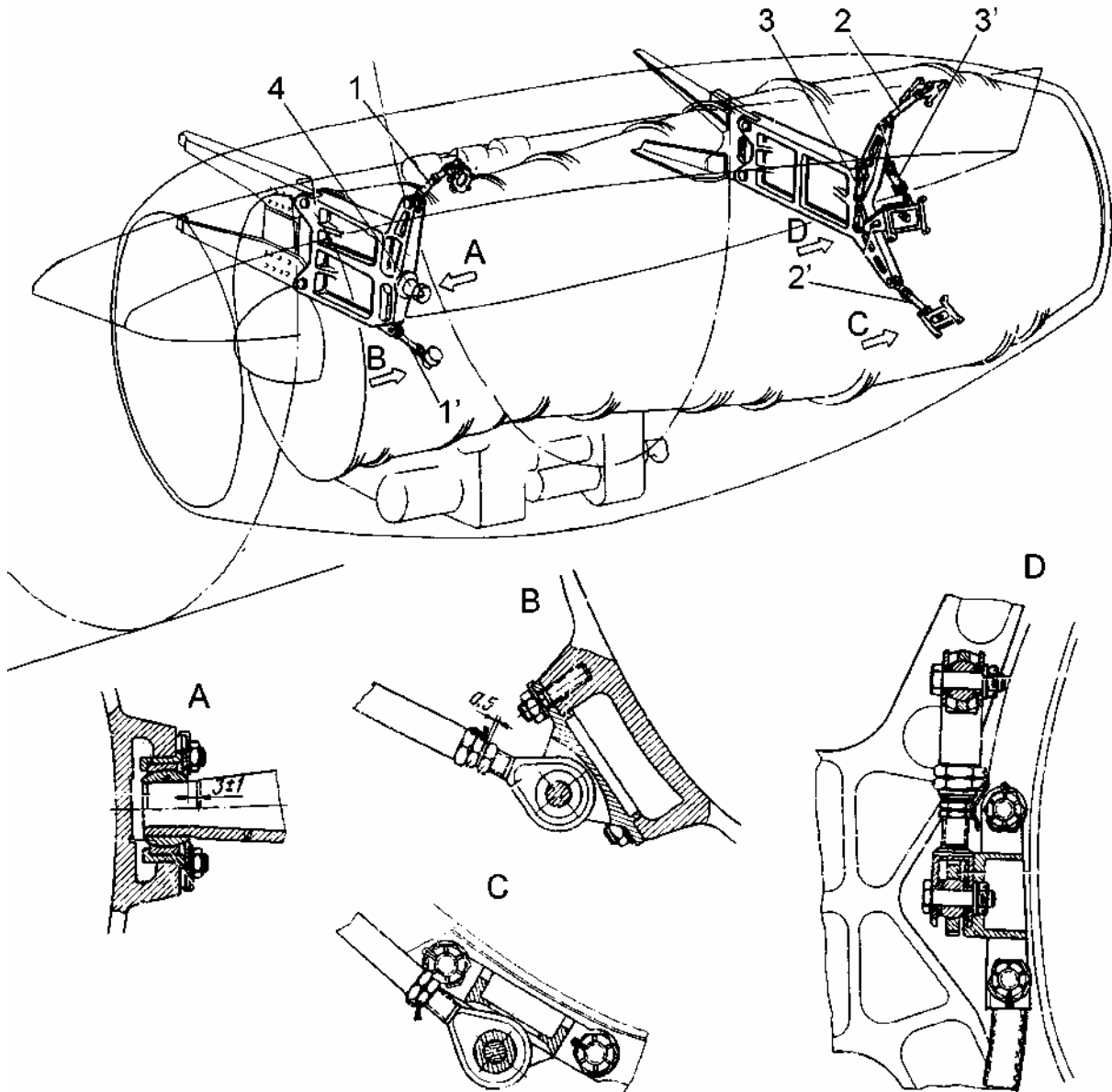
Thrust force and side loadings are carried in nodes A and A'. These nodes are formed by rods 1, 3, 4, 5 arranged almost in horizontal planes. The most part of vertical load is carried by group of rods 1, 2, 3 lying in one vertical plane. Rods 6 and 7 take part in transmission of  $M_y$  moment and side loading.

At rigging statically indeterminable structures, first of all they detach redundant members. Rigging is carried out. Then redundant members are

attached again.

$F_x$	A, A'	1, 3, 4, 5	$M_x$	A, A', B, B', C	1, 2, 3, 4, 5, 6, 7
$F_y$	A, A', C	1, 2, 3	$M_y$	A, A', B, B'	1, 3, 4, 5, 6, 7
$F_z$	A, A', B, B'	1, 3, 6, 7	$M_z$	A, A'	1, 3, 4, 5

**Engine mounts of turbojets with pylons in the tail part of a fuselage.** For instance we shall consider the engine mounts of Yak-40 airplane (Fig. 3.17).



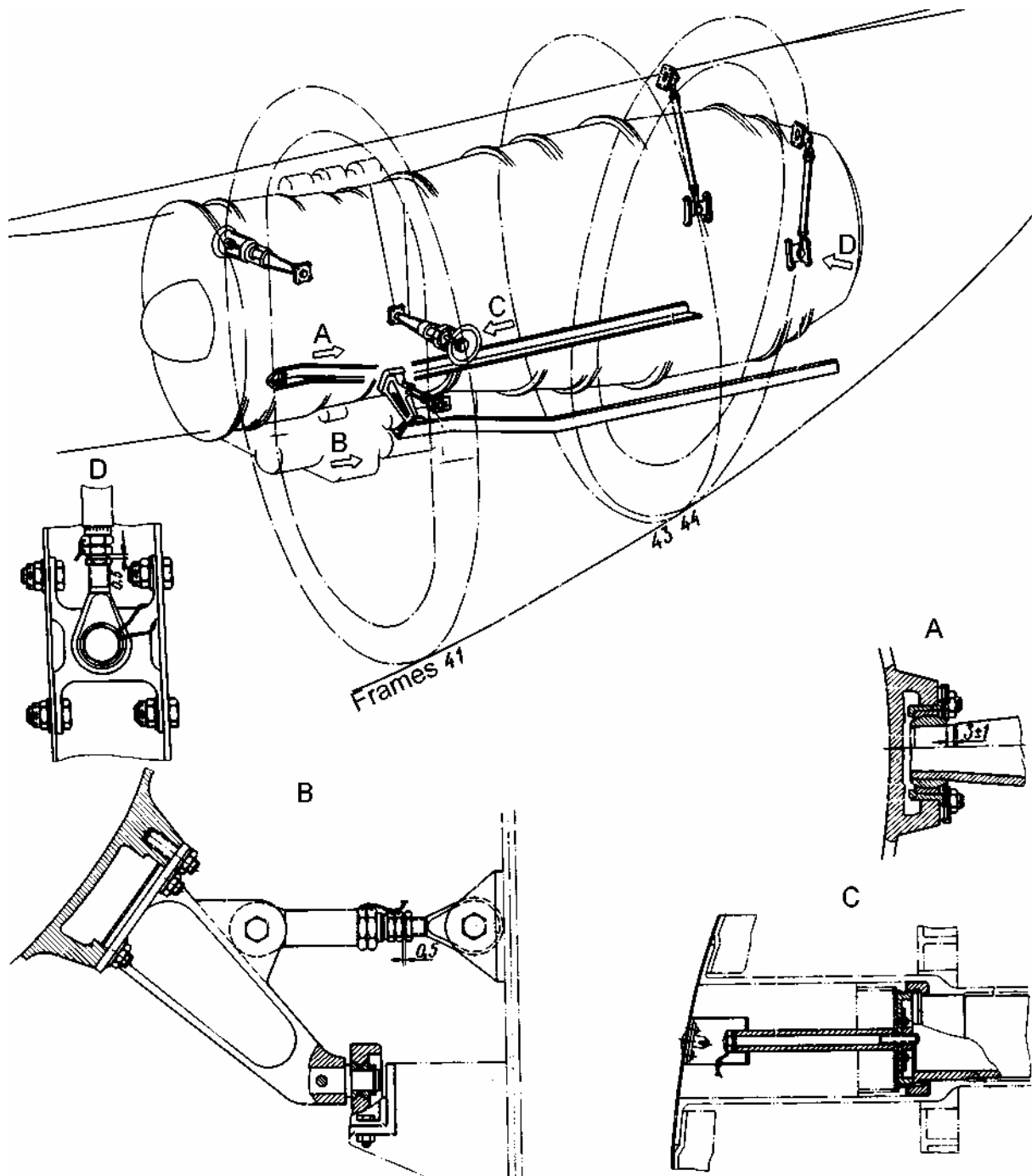
**Fig. 3.17. Engine Mounts of Bypass Turbojet with Pylon to Tail Part of Fuselage**

The scheme of attachment of side engine is shown in Fig. 3.17. It is attached to the transverse load-carrying beams of pylons at two supporting locations. The forward supporting location carries thrust force, vertical

load,  $M_x$  torque and side load. This supporting location consists of the pin 4 and two adjustable rods 1 and 1'. Each rod consists of a pipe and two eye ends with spherical bearings.

The rear supporting location consists of rods 2, 2', 3, 3'. It carries vertical force, side force,  $M_x$  and  $M_y$  moments. Thus thermal expansion of the engine is provided.

The scheme and some elements of engine mounts of **center engine of Yak-40** are shown in Fig. 3.18.



**Fig. 3.18. Engine Mounts of Bypass Turbojet in Tail Part of Fuselage**

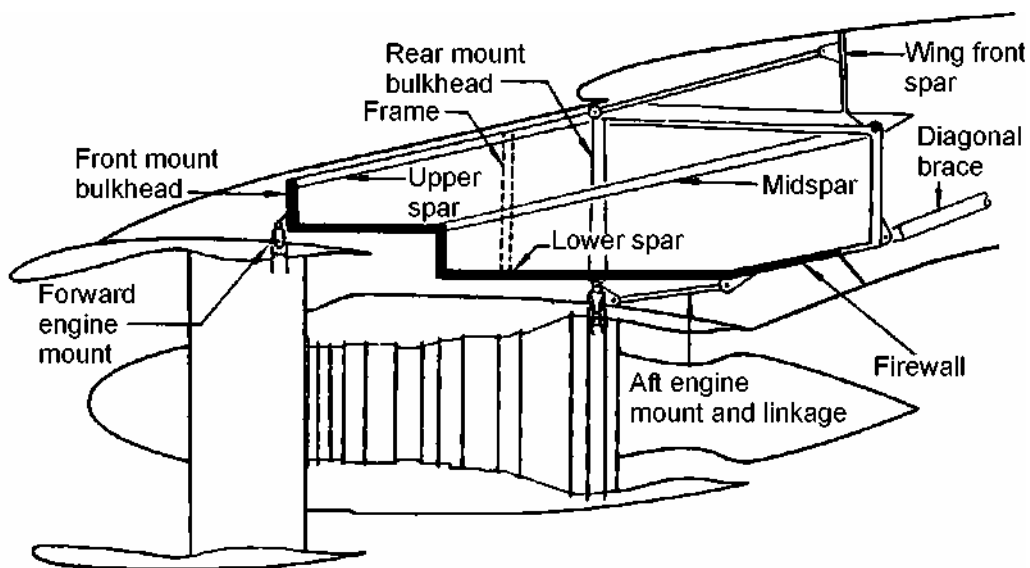
Its attachment to load-carrying frames of the tail part of the fuselage are also performed at two supporting location. The forward supporting location carries thrust force, vertical and side loadings,  $M_x$  and  $M_y$  moments. This supporting location consists of two pins and one horizontal adjustable rod. These pins are installed in sockets of the load-carrying frame and are locked with nuts.

$F_x$	A	1, 1', 2, 2', 4	$M_x$	A, B	1, 1', 2, 2', 3, 3', 4
$F_y$	A, B	1, 1', 2, 2', 3, 3', 4	$M_y$	A, B	1, 1', 2, 2'
$F_z$	A, B	1, 1', 2, 2'	$M_z$	A, B	1, 1', 2, 2', 3, 3', 4

The rear supporting location carries a part of vertical load and  $M_x$  and  $M_z$  moments (about longitudinal and lateral axes). Thus capability of thermal expansion of the engine is provided. When being installed and removed, the engine moves into the tail part of the fuselage by rails.

**Engine mounts of bypass turbojets with pylons under wing.** Such arrangement now the most widely used at passenger and transport airplanes.

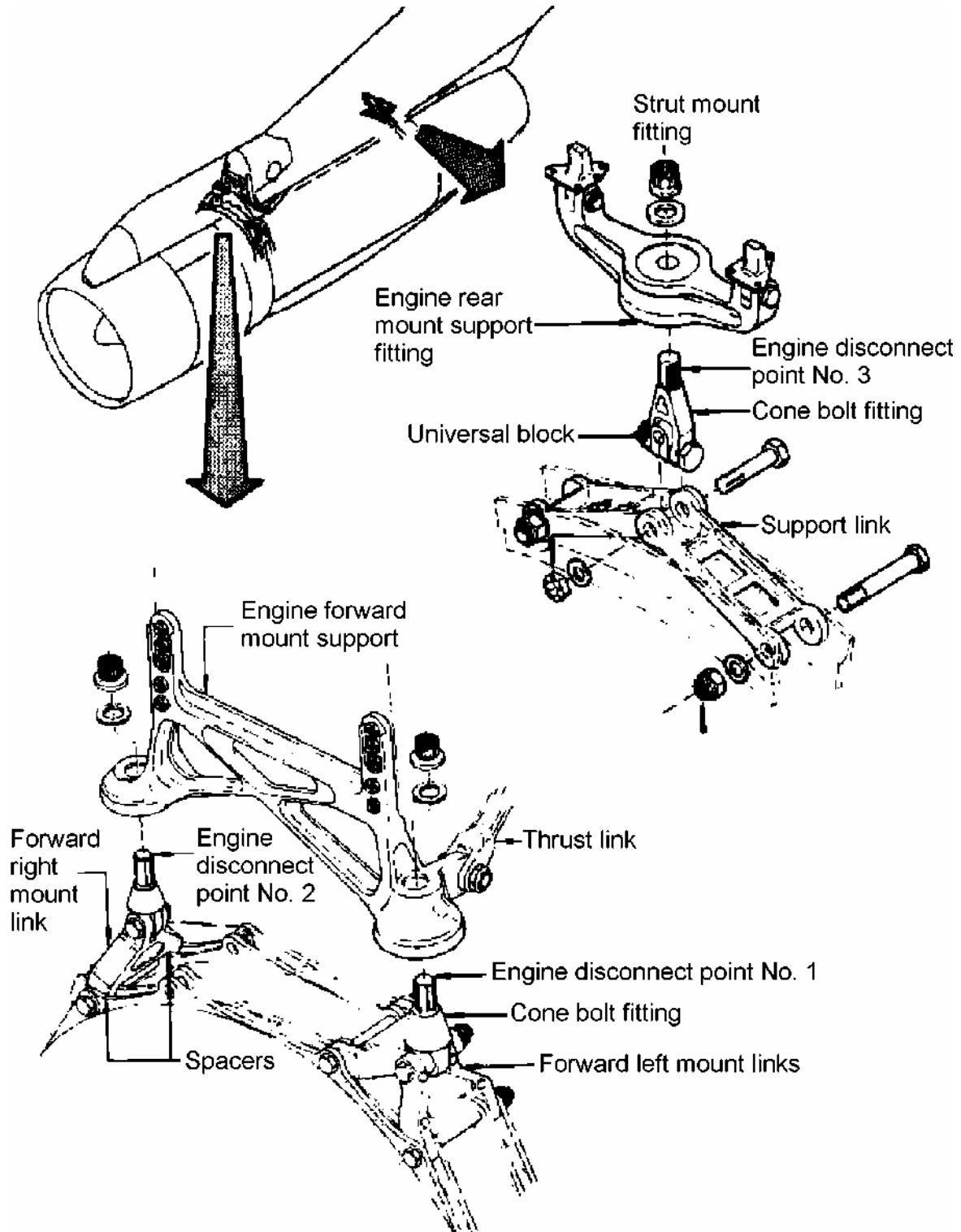
In this case the load-carrying structure of a pylon carries and transmits all loadings, acting from engine and nacelle, to spars and load-carrying ribs of wing. The problem of carrying of these loadings is complicated by the fact that, the engines are usually considerably staggered forward. This made to prevent flutter of wing. However it results in the big sweep of the pylon. This problem can be solved by using longitudinal and transverse beams, arranged inside the pylon. This design is shown in Fig. 3.19.



**Fig. 3.19. Engine Mounts of (Bypass) Turbojet with Pylon under the Wing**



In this case engine mount attaches an engine to a pylon. The design of this engine mount is similar to the one of the engine on pylon at a fuselage. Fig.3.20 shows the other example of engine mount to a pylon under the wing. In this case, the engine is connected to the pylon by three pins, each of them can carry out three forces (axial force and two shear ones).



**Fig. 3.20. Engine Mount of (Bypass) Turbojet to Pylon**

## Theme 4. VIBRATIONS OF POWER PLANTS. SHOCK–ABSORPTION OF ENGINE MOUNTS

### 4.1. Types of Vibrations. Causes of Vibrations. Effects of Vibrations

Process of periodic change of some generalized coordinate in time is called **vibrations (oscillations)**. The number of oscillations which is carried out by a system in a unit of time is called **frequency** of oscillation. Maximum deviation from an equilibrium position is called **amplitude of oscillation (vibrational amplitude)**.

There are free and forced oscillations. **Free** oscillations are these which occur after some initial disturbance of equilibrium state of a system which afterwards remains as it is (Fig. 4.1, a).

**Forced** oscillations are performed under influence of the given shaking forces. The forced vibration frequency is equal to frequency of the shaking force (Fig. 4.1, b).

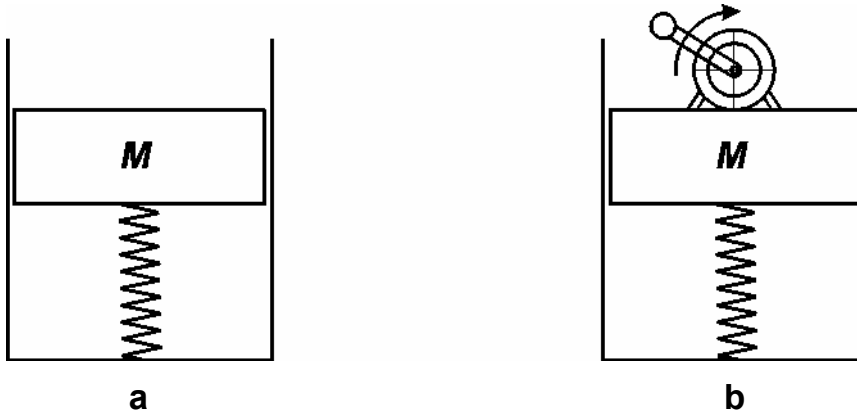


Fig. 4.1. Free and Forced Oscillations

The amplitude of oscillation sometimes reaches such value that it complicates a normal operation of instruments and gears and can cause breaking of structure elements. Vibrations reduce the service life and reliability of parts and equipment and frequently cause fatigue failures of a structure.

One of the main sources of oscillations on airplanes and helicopters is power plant (in particular engines and propellers or rotors).

The engine and propeller or rotor create vibrations of two types: mechanical and acoustical.

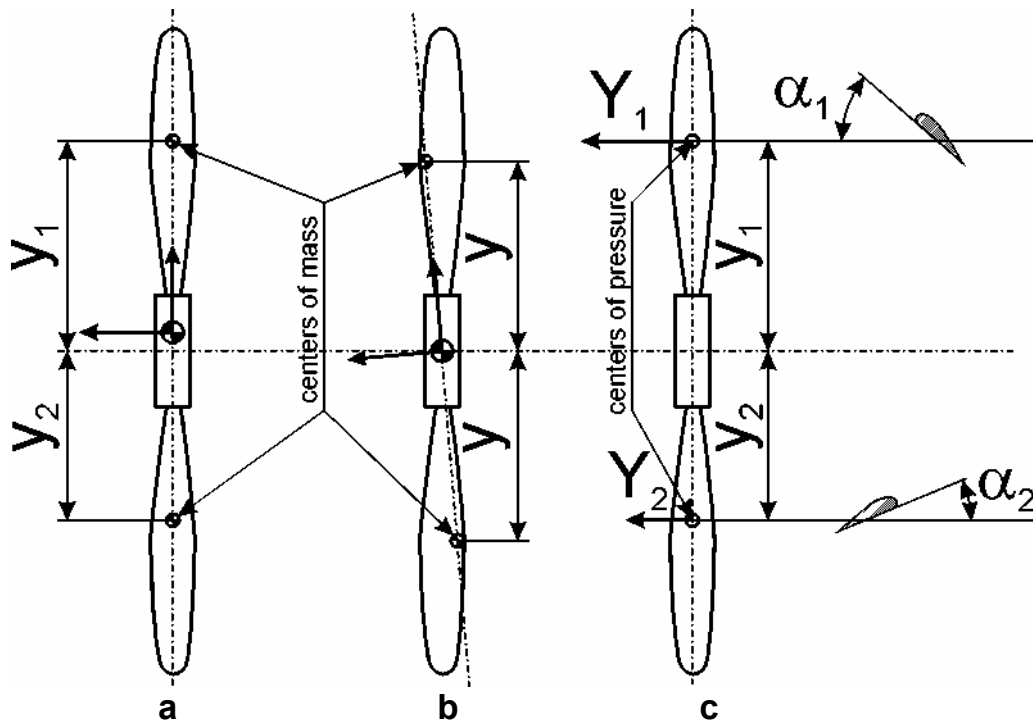
The reasons of vibrations of the engines are as follows:

- Non-uniformity of torque of piston engines;
- Unbalance of inertial force of a translational and a rotary motion of engine parts;
- Static and dynamic unbalance of rotating engine parts.

The reasons of vibrations of a propeller are its static, dynamic and aerodynamic unbalances.

The static unbalance of a propeller (Fig. 4.2, a) appears when the center of mass of a propeller does not coincide with its axis of rotation. Value of this noncoincidence is determined by the tolerance on static balance of a propeller.

The dynamic unbalance of a propeller (Fig. 4.2, b) appears when the axis of rotation of a propeller does not coincide with one of its principal centroidal axes. It happens in a case, when centers of mass of separate blades lie in different planes, normal to axis of rotation. Thus an unbalanced couple of forces, concerning y and z axes, appears.



**Fig. 4.2. Propeller Unbalances**

The aerodynamic unbalance of a propeller (Fig. 4.2, c) appears, if aerodynamic forces of separate blades have various values or directions. It happens, if:

- blades have various blade angles;
- centers of pressure of blades are located on various distances from the axis of rotation;
- there is stall phenomena on blades, which cause oscillations of blades, under big angles of attack;
- there is interference of a propeller with parts of aircraft arranged behind or in front of propeller (a wing or a fuselage).

The shaking forces, appearing when a power plant is under operation, are periodic one. Their frequency is multiply to the rotational speed of

parts of the engine or the propeller. Thus a ratio of frequency of free oscillation  $\rho$  to frequency of shaking force  $\omega_{sh}$  is of great importance.

Oscillation frequency  $\omega_{sh}$  of the engine (propeller) linearly depends on rotational frequency of engine rotor (propeller). The harmonic order of shaking forces and shaking moments of the engine in relation to rotational frequency of crankshaft of piston engine or rotor of turbojets makes:

– for piston engines: 0.5, 1, 2, 0.5(a+1), where a — number of cylinders;

– for turboprops and turbojets: 1, 2,...

The order of propeller harmonics for a k-bladed propeller is: 1, 2... k, mk, where m = 2, 3,...

The most dangerous are vibrations with frequencies n, 2n for piston-engined airplanes; n for turbojet airplanes;  $n_p$ ,  $2n_p$  for turboprop airplanes, where n and  $n_p$  — rotational frequencies of the rotor of engine and the propeller accordingly.

It is necessary to note, that in flight the rotational frequency of the engine and propeller varies almost continuously and forms an area of low frequencies. From the point of view of vibrations, piston engines are the most dangerous ones. Power plants with turbojets are more favorable, because the frequencies of forced oscillations are higher and the rotating masses are better balanced. Turboprops take an intermediate place between turbojets and piston engines.

Vibrations can cause:

1. Decrease of service life and reliability of vibrating parts because of fatigue of materials.

2. Increase of wear and appearance of backlashes.

3. Disturbance of pressure-sealings of pipelines and systems.

4. Problems in operation of instruments and observation of them.

5. Discomfort for passengers and crew. In this event the vibrations can be unacceptable earlier, than they achieve the dangerous values for a structure.

6. Possibility of resonance. Thus the amplitude of movements and stresses in the structure sharply increase.

An engine as a solid body, hingedly attached to aircraft with elastic three-dimensional truss, is a system with six degree of freedom. These are three translations in a direction of three axes and three angular movements around these axes. Hence, we have six frequencies of free oscillations.

Depending on arrangement of the center of mass of engine and elastic center of engine mount they distinguish:

**separated oscillations** — at coincidence of the center of mass of engine and the elastic center of engine mounts;

**doubly connected oscillation** — when the center of mass and the elastic center lie in axis of rotation of propeller or main rotor;

**triply connected oscillation** — generally.

At separated oscillations, frequencies of free oscillations of engine are in a narrow range. Coincidence of the center of mass and the elastic center is provided by a change of shock absorber's stiffness of engine mounts. Usually there are triply connected oscillations, hence movement along one of the axes causes movements and turns around all other axes. Further we achieve a separation of the oscillations by changing the stiffness of shock absorbers and the stiffness of engine mounts. Thus there is a capability to change any frequency of free oscillations to eliminate a resonance, not changing natural frequency in other directions.

## 4.2. Stiffness of Engine Mounts

The stiffness of engine mounts is determined by calculation or experimental ways. For this purpose, the unit generalized force is applied in the center of mass of the engine in the direction of the required axis and the corresponding generalized movement is found. The inverse values to these movements make stiffness of engine mounts. The stiffness of a shock absorber is a derivative (Fig. 4.3)

$$K_A = \frac{dP}{d\delta_A} = \operatorname{tg} \varphi,$$

where  $P$  — external loading;  $\delta$  — motion of the shock absorber caused by this loading.

A stiffness of engine mounts together with shock-absorption along or around one axis is determined as harmonic average of their stiffnesses:

$$\frac{1}{K_{P,P}} = \frac{1}{K_M} + \frac{1}{K_A}; \quad K_{P,P} = \frac{K_M K_A}{K_M + K_A},$$

where  $K_M$  and  $K_A$  — the stiffnesses of engine mounts and shock-absorption along or around the specified axis.

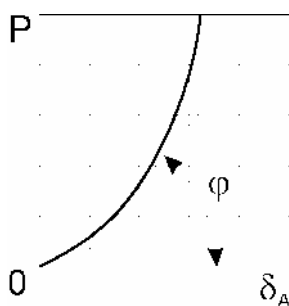


Fig. 4.3. Stiffness of Shock Absorber

Knowing the design of engine mounts, the type and arrangement of shock absorbers, it is possible to determine the frequency of natural oscillations of the system. For separated oscillations these frequencies are determined under the following formulas:

$$v_i = \frac{1}{2\pi} \sqrt{\frac{K_i}{M}}; \quad v_i = \frac{1}{2\pi} \sqrt{\frac{K_i}{J_i}},$$

where  $M$  — mass of the engine;  $J_i$  — mass moment of inertia of the engine around of  $i$  axis.

### 4.3. Oscillations of System with One Degree of Freedom

For an instance we shall consider oscillations of a system with one degree of freedom (Fig. 4.4).

$$\begin{aligned} F &= F_0 \cdot \sin(\omega t); \\ I &= M \cdot \ddot{y}; \\ X &= \lambda \cdot \dot{y}; \\ P &= K \cdot y. \end{aligned}$$

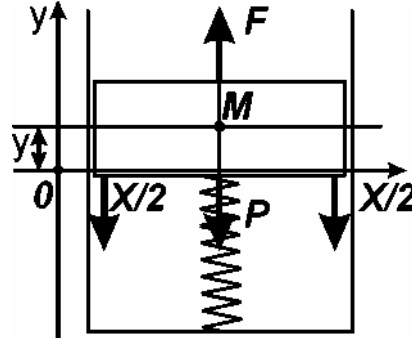


Fig. 4.4. Oscillating System with One Degree of Freedom

In Fig. 4.4 it is designated:  $K$  — stiffness of the spring;  $M$  — mass;  $\lambda$  — drag coefficient;  $P$  — elastic force;  $I$  — inertial force;  $F$  — external force;  $X$  — drag;  $y$  — movement relative to the equilibrium position.

Let the external force change under the harmonic law:  $F = F_0 \sin \omega t$ . Then the equilibrium equation shall be written as:

$$\begin{aligned} M\ddot{y} &= F - X - P; & M\ddot{y} + \lambda\dot{y} + Ky &= F_0 \sin \omega t; \\ \ddot{y} + \frac{\lambda}{M}\dot{y} + \frac{K}{M}y &= \frac{F_0}{M} \sin \omega t; & \ddot{y} + 2\kappa\dot{y} + p^2y &= q \sin \omega t; \end{aligned}$$

where  $p = \sqrt{K/M}$  — natural frequency of oscillations;  $2\kappa = \lambda/M$ ,  $q = F_0/M$ .

The last equation is a linear nonhomogeneous differential equation of the second order with constant factors. Its solution at entry conditions

$y|_{t=0} = y_0$ ,  $y'|_{t=0} = y'_0$  is well-known:

$$\begin{aligned} y &= e^{-\kappa t} \left\{ \left[ y'_0 + \kappa y_0 + \frac{q\omega(2\kappa^2 - p^2 + \omega^2)}{(p^2 - \omega^2) + 4\kappa^2\omega^2} \right] \frac{\sin \sqrt{\kappa^2 - p^2} t}{\sqrt{\kappa^2 - p^2}} + \right. \\ &+ \left. \left[ y_0 + \frac{2\kappa\omega q}{(p^2 - \omega^2)^2 + 4\kappa^2\omega^2} \right] \cos \sqrt{\kappa^2 - p^2} t \right\} + \\ &+ \frac{q}{\sqrt{(p^2 - \omega^2)^2 + 4\kappa^2\omega^2}} \sin \left[ \omega t - \arctan \frac{2\kappa\omega}{p^2 - \omega^2} \right]. \end{aligned}$$

Here the first member represents natural oscillations, and the second one is forced oscillations. The amplitude of forced oscillations can be presented in the following view:

$$A_F = \frac{q}{\sqrt{(p^2 - \omega^2)^2 + 4\kappa^2\omega^2}} = \frac{q/p^2}{\sqrt{\left(1 - \left(\frac{\omega}{p}\right)^2\right)^2 + 4\left(\frac{\kappa}{p}\right)^2\left(\frac{\omega}{p}\right)^2}} =$$

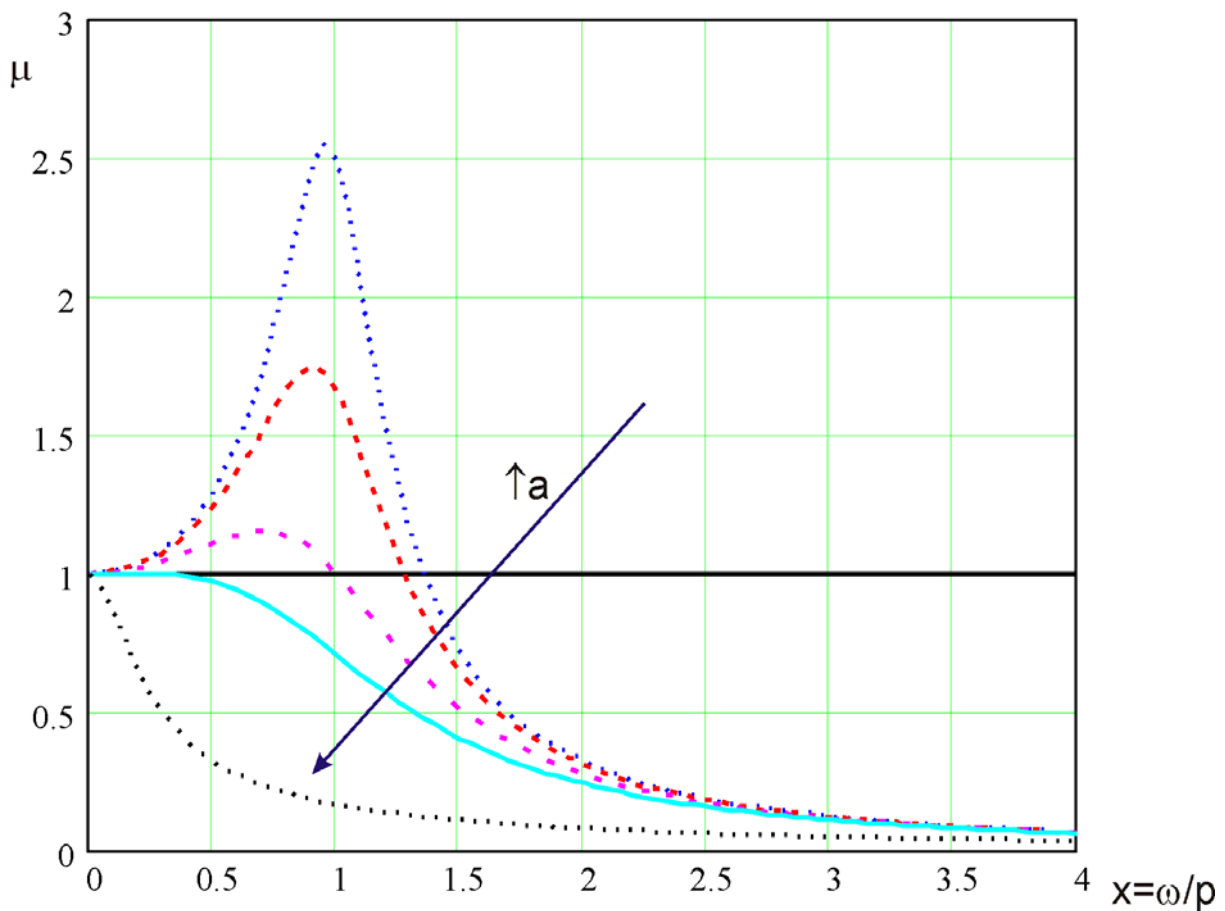
$$= \frac{F_0/M \cdot M/K}{\sqrt{(1-x^2)^2 + 4a^2x^2}} = \frac{F_0/K}{\sqrt{(1-x^2)^2 + 4a^2x^2}} = \frac{y_{ST}}{\sqrt{(1-x^2)^2 + 4a^2x^2}} = y_{DYN},$$

where  $y_{ST} = F_0/K$  — represents the movement caused by the statically applying of the external force amplitude value;  $x = \omega/p$  and  $a = \kappa/p$  — is denoted for short.

Let's work in the concept of **amplification factor** (Fig. 4.5) as a relation of the amplitude of forced oscillations to the movement caused by the statically applied external force amplitude value:

$$\mu = K_{DYN} = \frac{y_{DYN}}{y_{ST}} = \frac{1}{\sqrt{(1-x^2)^2 + 4a^2x^2}}.$$

On the diagram (Fig. 4.5) it is possible to find three zones:



**Fig. 4.5. Amplification Factor**

1.  $\omega^2 \ll p^2 - 2\kappa^2$ ;  $\mu \approx 1$ ;  $y_{DYN} \approx y_{ST}$ ;
2.  $\omega^2 \approx p^2 - 2\kappa^2$ ;  $|\mu| \gg 1$ ;  $y_{DYN} \gg y_{ST}$ ;
3.  $\omega^2 > p^2 - 2\kappa^2$ ;  $|\mu| < 1$ ;  $y_{DYN} < y_{ST}$ .

On the basis of this diagram it is possible to make the following conclusions:

1. The coincidence of frequencies of forced and natural oscillations (the second zone) is inadmissible.

2. The case, that the frequency of forced oscillation is smaller than the frequency of natural one (the first zone), is allowable, but is undesirable. Thus there is a capability to hit in a resonance with higher harmonics of the shaking force. (It occurs in turboprop engine mounts).

3. The case, that the frequency of forced oscillation is bigger than the frequency of natural one (the third zone), is desirable. As the amplitude of forced oscillations is less than the static values of movements.

4. Even at presence of drag (damping), the forced oscillations happen with a frequency of shaking force.

5. The amplitude of the forced oscillations does not depend on entry conditions and time. I. e. the forced oscillations are not damped.

6. At a resonance ( $\omega = p$ ) the amplitude of the forced oscillations remains finite  $A_F = \frac{q}{2\kappa p}$ . And the amplitude reaches a maximum in the

other point ( $\omega = \sqrt{p^2 - 2\kappa^2}$ ), i.e. up to the resonance, and makes:

$$A_F = \frac{qp^2}{2\kappa\sqrt{p^2 - \kappa^2}}.$$

7. In the forced oscillations with drag, a shift of phase of oscillations  $\varphi_F = \arctan \frac{2\kappa\omega}{p^2 - \omega^2}$ , in comparison with the phase of shaking force, al-

ways takes place. The maximal value of the phase shift  $\varphi_F = \frac{\pi}{2}$ , is reached at the resonance.

8. Though amplitudes of the forced oscillations with drag remain finite even at a resonance, under continuous work, in these conditions fatigue failures of elements of the design are probable.

#### 4.4. Shock-absorption of Engine Mounts

To decrease vibrations and forces in engine mounts it is necessary:

– To decrease the values of shaking forces by improvement of balancing of engine and propeller;



- To apply dampers — the oscillation absorber which dissipate a significant part of the energy of oscillation;
- To select the stiffness of engine mounts and shock absorbers for elimination of a resonance.

The essence of shock-absorption is that the engine is attached to the aircraft structure not stiffly, but through elastic joints (shock absorbers) which sharply reduce the oscillation frequency of power plant.

Shock absorbers are installed in the attachment fittings of the engine mounts to the engine or in the attachment fittings of engine mounts to aircraft structure. The second way is undesirable, since shaking loads act directly to engine mounts here; there is a danger of a resonance. Besides the distance from the engine center of mass up to shock absorbers is more, hence, the loads, which act to shock absorbers, are more too.

The struts-dampers are installed in the engine mounts for shock-absorption. Sometimes the shock absorbers are installed in both attachment fittings of engine mounts to engine and in attachment fittings of engine mounts to aircraft structure. Thus the complete separation of the aircraft structure from vibrations of engine is reached (ref. Fig. 3.4).

At oscillations around  $y$  and  $z$  axes (normal to propeller axis) there is a gyroscopic precession oscillation, which is very harmful to propeller and to rotating rotor of engine. They are caused by a gyroscopic effect of the propeller and the rotor of the engine. To decrease them, the stiffness around these axes is limited by required minimum.

The greatest shaking forces act along  $y$  and  $z$  axes and around  $x$  axis. The most effective shock-absorption is installed for their absorption. The stiffnesses in other directions are several times (up to 10) higher.

The stiffness of engine mount is different in different directions. Accordingly, natural frequencies of oscillation and required stiffness of shock absorbers are different in different directions.

#### 4.5. Design of Shock Absorbers

An elastic element of a shock absorber is usually rubber. The shear modulus  $G$  connected with modulus of elasticity  $E$  and Poisson ratio  $\mu$  by formula

$$G = \frac{E}{2(1 + \mu)}.$$

As far as Poisson ratio for rubber makes **0.46**, then shear stiffness of rubber shock absorber is in **2,92** times less than compression one. As it is known, stiffness of compressed element is equal to:

$$K = EF/L,$$

where  $F$  — element cross-sectional area,  $L$  — its length. So different stiffness in different directions is provided by different thickness or cross-section area of rubber along corresponding directions or by loading of rubber with compressive and shear loads. For example, in bushing-type shock absorbers, the thickness of rubber is different in different directions. In addition, that rubber carries compressive stresses or shear stresses depending on direction of loading.

Fig. 4.6. shows that, this design of shock absorber provides:

- **big** angular motion around  $x$  axis, which protect engine mount and aircraft from impulses of engine torque;
- **enough** radial engine motion, which soften action of impulses from unbalanced loads to aircraft;
- **small** motion along  $x$  axis, which is dangerous owing to precession propeller oscillation.

Shock absorbers of turboprops (Fig. 4.7, 4.8) are arranged in a plane, which is parallel to axis of the engine. To provide small stiffness around  $x$  axis, the distance between shock absorbers by width should be small. And to provide high stiffness around  $y$  and  $z$  axes, the distance between forward shock absorbers and struts-dampers should be big.

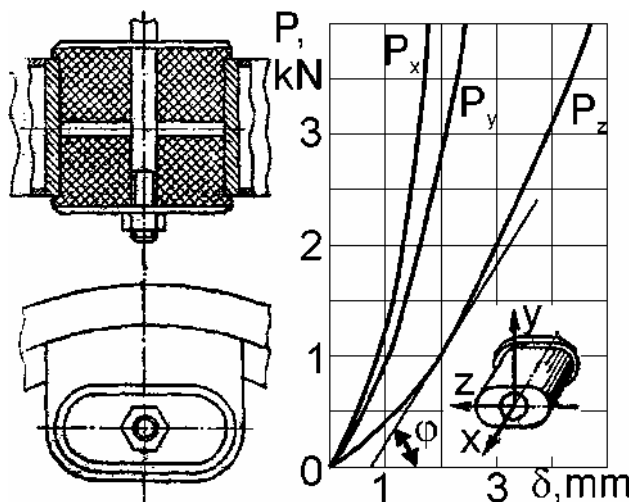


Fig. 4.6. Shock Absorber of Bush-type and Its Characteristics

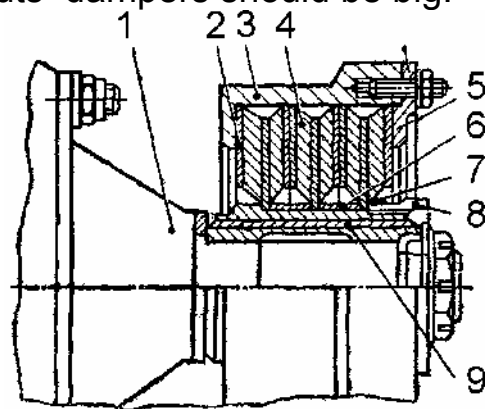
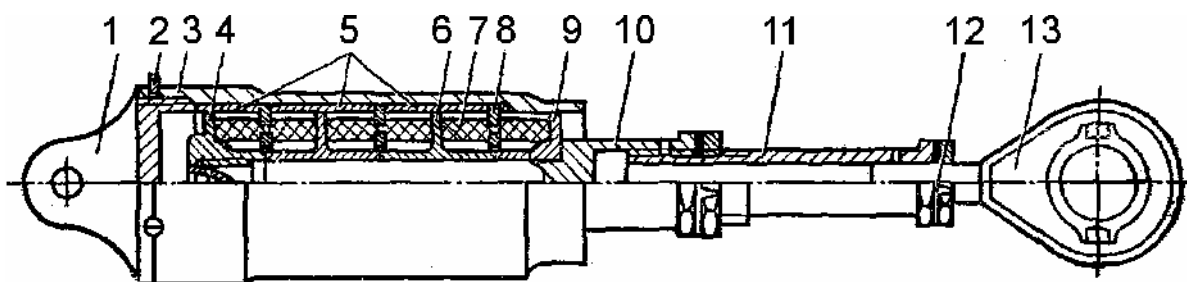


Fig. 4.7. Disk Shock Absorber:  
1 — Engine Trunnion; 2 — Washer;  
3 — Casing; 4 — Disk-Type Shock Absorber; 5 — Flange; 6 — Bushing; 7 — Locking Washer; 8 — Bushing-Type Shock Absorber

When selecting the place for shock absorber, it is necessary to take into account, that fuel and oil act harmfully to rubber of shock absorber. Hence, shock absorbers should be protected with covers or casings. Besides the elastic properties and strength of rubber decrease at a temperature decreasing. To protect the shock absorbers from harmful effect of heat flows of jet engines they sometimes use a forced cooling of shock absorbers.



**Fig. 4.8. Strut-damper:**

**1 — Fork; 2 — Retaining Screw; 3 — Casing; 4 — Nut; 5, 6, 9 — Bushings; 7 — Disk Shock Absorber; 8 — Ring; 10 — Central Rod; 11 — Transitional Bushing; 12 — Locking Washer; 13 — Eye**

## **Theme 5. FUEL SYSTEMS OF POWER PLANT**

### **5.1. Purpose of Fuel Systems and Requirements to Them**

The **fuel system** represents a complex of interacted subsystems intended to feed the engine by fuel in all operating conditions allowable for this aircraft, moreover fuel systems can perform a number of additional functions (such as: cooling of oil, keeping specified center-of-gravity position, etc.).

The **number of requirements** is made to the fuel system. A part of which is general, and the other part depends on the type and purpose of the aircraft (supersonic, air tanker, etc.).

1. A reliable supply of fuel to all engines under all expected operation conditions of this aircraft on ground and in flight.

2. To ensure survivability and flight safety (including fire-prevention).

– Fuel tanks must not be arranged near passenger cabin or crew.

– All metal elements of a fuel system should be interconnected and also connected to the aircraft (and to ground at parking) in order to prevent a discharge of a static electricity.

– At a distance of 500 mm from the end, the wing should not be filled with fuel; to be on the safe side at lightning stroke.

– Fuel pipelines are desirable to be placed inside the fuel tanks. In case of a leakage, the fuel flows in the fuel tanks, instead of flowing in the airframe.

3. The fuel tanks should contain reserve of the fuel required for flight with the specified maximum range (duration) and a stand-by reserve of fuel for 45 minutes of flight.

4. To maintain of the specified center-of-gravity position, fuel should be used in a specified sequence automatically. In case of failure of the automatics, there should be a capability of manual control by fuel tanks

usage.

5. Fuel should be filtered from mechanical impurities and water. The water dissolved in fuel and allocated from it in the fuel tanks or in some elements of the fuel system, must not cause disturbance of fuel system operation.

6. It is necessary to provide protection of units of the fuel system from: corrosion, frosting-up, microorganisms, discharges of a static electricity, overheating.

7. It is necessary to provide sufficient strength, vibration-proof and hermiticity.

8. It is necessary to provide full defueling on ground through easily accessible and convenient in maintenance valves.

9. It is necessary to provide the reliable and continuous control of the fuel system operation.

10. The fuel system should be simple in control, convenient in maintenance. Maintenance ground time of the fuel system should be minimal (it is called maintainability).

## 5.2. Aviation Fuels and Their Properties

The main fuel for up-to-date aviation piston engines and jet engines are liquid mixes of hydrocarbons. They are recovered by gasoline refining. For jet engines aviation kerosene is used. Working process in the piston engines imposes limitations for the period of a mixing, firing and combustion of the fuel. Therefore easily inflammable fuel — aircraft petrol is applied for aviation piston engines.

The engine start-up, thrust, efficiency, reliability of operation, etc. depend on fuel quality.

Let's consider the main properties of fuels, which should be taken into account when designing fuel systems.

**1. Energy properties** of fuel are determined by its **calorific power** and **density**.

The mass calorific power of kerosene and petrol is approximately equal to 43000...44000 kJ/kg.

At a temperature of 20°C density of petrol makes  $\rho = 0.72...0.74$  kg/dm<sup>3</sup>, and of kerosene makes  $\rho = 0.755...0.85$  kg/dm<sup>3</sup>.

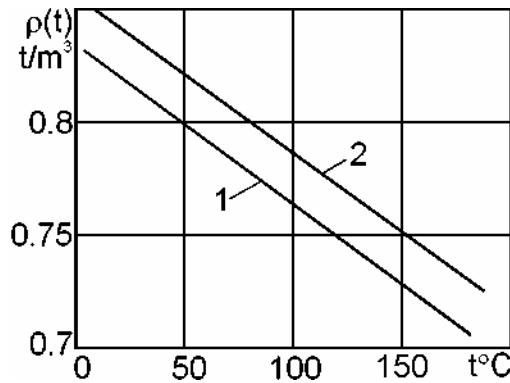
Volumetric calorific power is the product of density of fuel by its mass calorific power. Volumetric calorific power of kerosene is approximately 15% higher, than that of petrol. Hence, when storing the specified mass of fuel onboard the aircraft, smaller tanks are required for the fuels with greater density.

The density of fuel depends on its temperature by the following rela-

tion (Fig. 5.1):

$$\rho_t = \rho_{20} - \alpha(t - 20),$$

where  $\rho_{20}$  — the density of fuel at a temperature 20°C;  $\alpha$  — the temperature correction to the density ( $\alpha = 0.0007...0.0009$  kg/m<sup>3</sup>°C).



**Fig. 5.1. Dependence of Fuel Density on Its Temperature:  
1 — for Fuel T-1; 2 — for Fuel T-5**

Decrease of the density when been heated resulting in increase of a volume of the fuel. This fact shall be taken into account, when designing the fuel tanks. Otherwise there will be a pressing-out of fuel through the vent system. It can cause a fire. It is also inadmissible because we lose fuel.

**2. Saturated vapor pressure (steam tension)** of fuel is the important characteristics of fuel. It determines their vaporability, impacts to the altitude performance of the fuel system.  $P_t$  — the greatest vapor pressure, being above fuel, which is reached at vaporation into a closed vessel at the specified temperature.

Aviation fuels are multicomponent liquids. Their saturated vapor pressure in a fuel tank depends on a ratio of volumes of steam and liquid phases. When pressure decreases, light ends of fuel first pass in a gaseous state into the over-fuel space of the fuel tank. These light ends have a high saturated vapor pressure. Steam tension of the stayed fuel thus is reduced.

At vaporation in the closed volume, the equilibrium state steps with the greater steam tension of liquid end of fuel and the earlier, the smaller ratio of volumes of vapor and liquid phases is. The standard ratio of volumes of vapor and liquid phases is assumed equal 4/1. Thus saturated vapor pressure is designated by  $P_{t4/1}$ .

Fuels with higher saturated vapor pressure (for example petrol) vapors faster. It improves start-up qualities of engines.

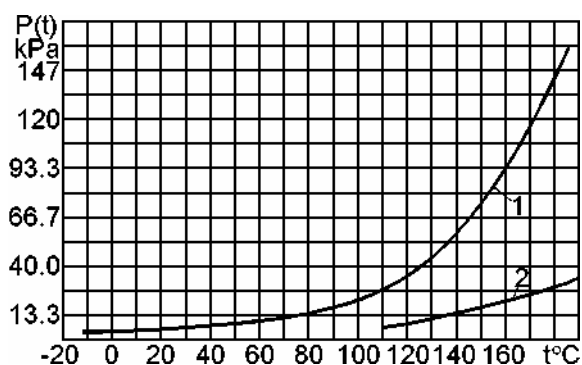
Saturated vapor pressure grows with increase of temperature (Fig. 5.2).

**3. Viscosity of fuel** determines a hydraulic resistance in a fuel system, impacts to activity of the fuel system equipment, to start-up of an engine.

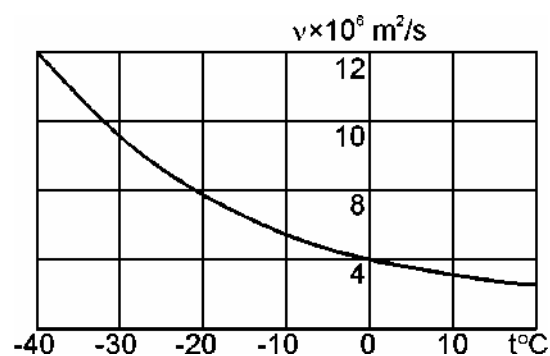
For instance let's cite some values of factor of kinematic viscosity of aviation fuels.

Factor of kinematic viscosity, m <sup>2</sup> /s depending on temperature, °C			
Fuel	-40	20	40
Petrol Б-70	1.75·10 <sup>-6</sup>	0.69·10 <sup>-6</sup>	0.58·10 <sup>-6</sup>
Kerosene T-1	8.59·10 <sup>-6</sup>	1.63·10 <sup>-6</sup>	1.21·10 <sup>-6</sup>

As it is shown in the table, the kinematic viscosity exponentially drops with increase of temperature (Fig. 5.3).



**Fig. 5.2. Dependence of Fuel Saturated Vapor Pressure on its Temperature:**  
1 — for Fuel T-1; 2 — for Fuel T-5



**Fig. 5.3. Dependence of Fuel Kinematic Viscosity on its Temperature**

**4. Presence of water and mechanical impurities in fuel** (a dust, products of corrosion of pipelines and units of the fuel system, etc.) impairs the operation of the fuel system.

Water penetrates into fuel as a result of dissolution of water vapors of air or as a result of condensation in volumes with fuel. Water can be contained in fuel in dissolved and not dissolved condition.

With fuel temperature increase, solubility of water in it increases. Thus an additional quantity of water from air dissolves in fuel. When temperature of fuel decreases this process goes in reverse order. Solubility of water decreases. Definite quantity of water vapors. Some part of the water falls out as water–fuel emulsions and precipitations.

Presence of water increases a corrosion activity of fuel. It results in corrosion of pipelines, appearance of leakages, clogging of fuel by products of corrosion. Besides at cooling of fuel, forming of crystals of ice is possible. These crystals clog fuel filters, equipment, pipelines.

Solubility of water in petrol is higher, than in kerosene. However

negative effect of water in the fuel system on kerosene is stronger, than in the fuel system on petrol. It is caused by the greater viscosity of kerosene. Kerosene keeps the water educed from a solution in a suspension. In petrol water rapidly falls out by gravity. Besides with increase of viscosity, water can hardly been separated in fuel filters.

To fight against an ice formation in the fuel system, the following methods are applied.

- Adding of additives (for example, an «И» liquid — ethylene glycol monoethyl ether), lowering temperature of ice forming.

- Injection of spirit in the most vital filters.

- Heating fuel by hot air from the compressor of the engine or by oil from a lubrication system of the engine before passing the fuel in filters.

**5. Solubility of air in fuel.** 10...20% of total volume of oil fuels consists of dissolved air. Solubility of air in fuel is directly proportional to tank pressure and inversely proportional to density, viscosity and value of surface tension of fuel. With an increase of the flight altitude the pressure drops, and air starts to educe intensively from fuel. It results in powerful swirling of fuel (cool boiling). The stream becomes two–phases, compressible, which leads to pressure fluctuations, vibrations and irregularity in fuel feed to engine and its stoppage.

Oxygen dissolves in fuel better, than nitrogen. Therefore the educing air has more oxygen, than atmospheric air has. It increases explosive risk of air–fuel mixture in the over–fuel space.

When increasing the content of air in fuel, steam tension of fuel increases too.

**6. Thermal stability of fuel.** When increasing the temperature above 120...150°C, a number of aviation fuels grades form insoluble precipitates and resinoid sediments. It results in violation of operation of the fuel system equipment. Filters clogging can result in full failure of the fuel system. The reason of it is oxidation of the chemical compounds containing sulfur, nitrogen and oxygen. These chemical compounds are present in fuel in a small amount.

**7. Charges of a static electricity** store at motion of fuel in pipelines and units of the fuel system. To prevent accumulation of the charge they apply: careful filtering of fuel from impurities, which reduces conductivity of the fuel. It is also apply some antistatic fuel additives. For elimination of discharge of the static electricity, all pipelines and units of the fuel system are electrically bonded. E. g. they are connected in one loop.

There are very many aviation fuel grades. They are divided into two groups: petrol and kerosene. Aviation petrol is marked by the letter "Б" and a figure. This figure designates the octane number of the petrol. Aviation kerosene is marked by the letter "Т" and a figure. This figure desig-

notes a fuel specification. Characteristics of particular fuel can be found in handbooks.

Within the course project of «Airplane Power Plants Systems Designing», the «Power Unit 9.x» software is used. In this software there are characteristics of four aviation fuels. As the software is developed for the educational purposes, it is enough. Students should competently choose fuel for their aircraft. Let's shortly consider the fuel grades, available in the software.

**Fuel T-1.** The specification of aviation kerosene, used for the most part in subsonic aircraft with jet engines, differs in relatively small kinematic viscosity in a design temperature range  $[-60, +60]^{\circ}\text{C}$ , but has relatively a high saturated vapor pressure of fuel  $P_{t4/1}$ .

**Fuel TC-1.** The specification of aviation kerosene, used for the most part in subsonic aircraft with jet engines, differs in small kinematic viscosity in a design temperature range  $[-60, +60]^{\circ}\text{C}$ , and a high saturated vapor pressure of fuel  $P_{t4/1}$ . The abbreviation is decoded as, Fuel sulphurous first, as it is made of the petroleum grades containing big quantity of sulfur. It results in essential carbonization into the nozzles of jet engines. But it is chipper than previous.

**Fuel T-5.** The specification of aviation kerosene, used for the most part in supersonic aircraft with jet engines, distinguish by rather high kinematic viscosity in a design temperature range ( $> 100^{\circ}\text{C}$ ), but rather low saturated vapor pressure of fuel  $P_{t4/1}$ .

**Petrol Б-70.** The specification of aviation petrol is used in piston-engined aircraft.

### 5.3. Structure of Fuel Systems

A fuel system is intended to arrange required fuel capacity onboard an aircraft and to uninterrupted fuel supply into engines under all possible modes and operation conditions for the aircraft.

Fuel systems consist of:

1. Subsystem of fuel storage (fuel tanks).
2. Subsystem of fueling.
3. Engine fuel feed subsystem (fuel use).
4. Subsystem of fuel transfer.
5. Subsystem of venting and pressurization of the fuel system.
6. Fuel dump subsystem.
7. Subsystem of support of the specified center-of-gravity position.
8. Measuring subsystem of the consumption and remaining fuel.
9. Control subsystem of the order of fuel use.



## 5.4. Schemes of Fuel System

The following ways are applied to supply the fuel use from tank: gravity feed, forced out feed and pump feed.

At **gravity feeding** (Fig. 5.4, a) fuel goes into the engine due to potential energy of the difference of levels between the fuel tank (which should be higher) and the engine. This scheme is the easiest and simplest. However at changes of aircraft attitude, this fuel feeding stops. It is applied on aircraft (especially on super light aircraft) with low-powered piston engines, because the required pump inlet pressure of such engines is rather small.

At **forced out feeding** (Fig. 5.4, b) gas is fed in over-fuel space of the fuel tank. It can be air from atmosphere, exhaust gas of piston engine, inert gas from bottles with overpressure (15...30 kPa) which forces out the fuel into the pipeline (thus the fuel tank can be even below the engine though it is undesirable). It results in increasing the altitude performance of the fuel system without using booster pumps. However the mass of the fuel tank, loaded outwards with pressure, also increases. The system has low survivability at a fault of the fuel tank. It is applied usually in a combination with a pump feeding. It is applied in the pure state in external tanks and in rockets or sometimes to transfer into the consumed fuel tank.

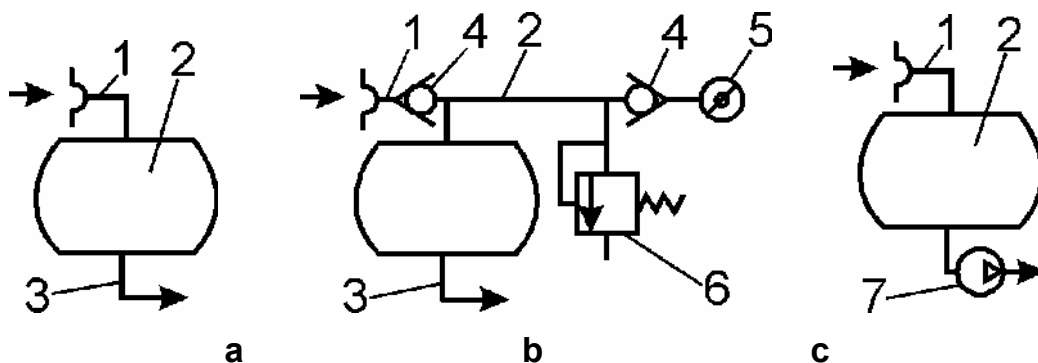


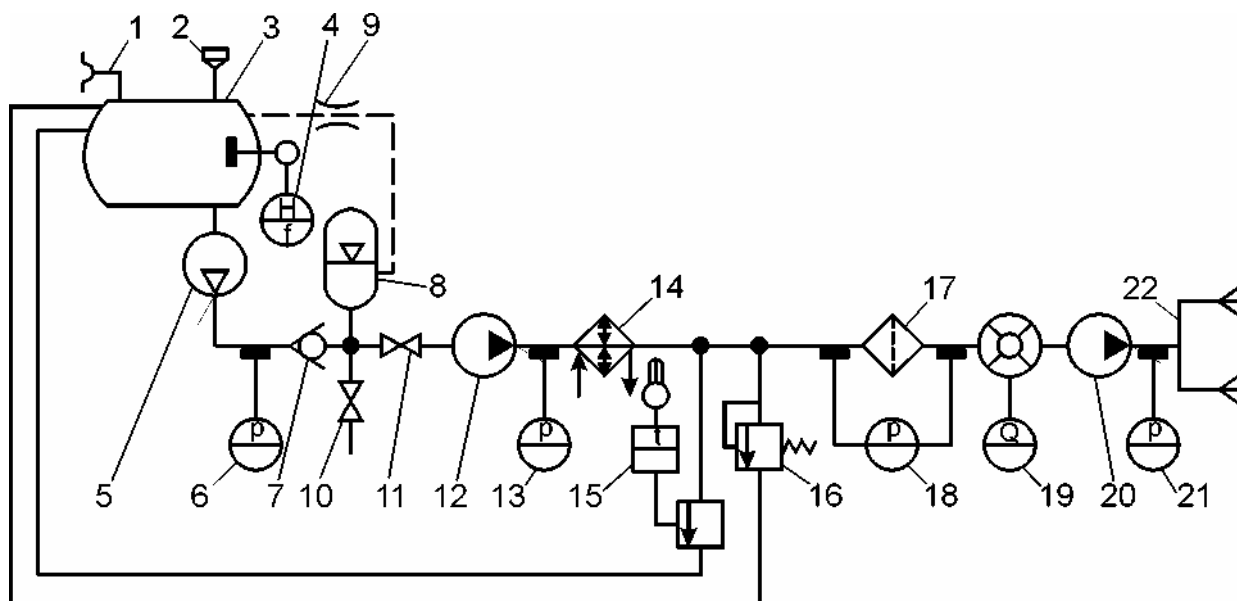
Fig. 5.4. Ways of Fuel Feed from Fuel Tanks:

1 — Air Intake from Atmosphere; 2 — Fuel Tank; 3 — Fuel Supply Pipeline Into Engine; 4 — Check Valve; 5 — Air Intake from Compressor; 6 — Safety Valve; 7 — Buster Pump of Airplane

**The pump feeding** (Fig. 5.4, c) allows to provide: depletion of the fuel tank arranged below the engine; sufficient altitude performance without increasing the mass of the fuel tank. However thus the mass of the fuel system increases due to pumps; fire danger increases at the installation of electric pumps in the fuel tank. Now such way of feeding, in a combination with small overpressure, is the most efficient by mass and widely used.

Let's consider the schematic diagram of the pipeline of fuel feeding

to the engine by the booster pump (Fig. 5.5). The fuel tank 3 is fueled through the filler neck or the pressure fueling connection 2. The fuel tank is vented through the vent subsystem 1. On up-to-date aircraft the multi-stage fuel boost is applied. Usually one booster pump is installed in the fuel tank, it is so-called **booster pump of airplane** (BPA) 5. Then one booster pump on the engine, it is so-called **fuel back-up pump** (FBP) 12. And at last there is the main pump of engine, it is so-called **main fuel pump** (MFP) 20. Thus BPA creates required pressure at the fuel back-up pump inlet. And FBP provides required pressure at the main fuel pump inlet. Advantages of such two-stage scheme of fuel boost are the following:



**Fig. 5.5. Scheme of Fuel Feeding Pipeline to Engine by Booster Pump**

- Smaller total mass of BPA and FBP, and smaller power to drive them in comparison with one pump, if this pump provided required pressure at the main fuel pump inlet;

- Smaller pressure in pipelines, hence, smaller leakages of fuel.

The check valve 7 provides the required direction of fuel motion. It is required at the parallel installation of two BPA. When one of them fails, fuel will not overflow back in the fuel tank. The check valve is also required before connection of the fuel accumulator 8 or at opening the valve of a cross-feed 10. The fuel accumulator serves for short-time supply of the engine with fuel at outflow of fuel from the pump, and also for damping of pressure fluctuation behind the pump.

The fuel shut-off valve 11 cuts off the fuel feeding in the engine at a fire, and also at replacement the engine on ground. At cooling oil by fuel the fuel-oil cooler 14 is installed in the fuel feeding pipeline. It also heats fuel. Thus the filter 17 is protected from a frosting-up, and fuel atomization

is improved. If the fuel consumption for a feed of the engine is less, than required for cooling oil, the part of fuel passed the cooler returns back in the fuel tank by the interconnecting pipeline supplied with the thermostat 15. The relief pipeline with the relief valve 16 serves for a by-pass of fuel in the fuel tank at overpressure (after the engine stops).

The test and measuring equipment is represented by: fuel quantity gauge 4, the pressure switch 6, the pressure gauge 13, the filter failure switch 18, the flowmeter 19 and the pressure gauge 21 before the fuel nozzle manifold 22.

At a great amount of fuel, big fuel tanks are required for its storage. Difficulties of arrangement of the big fuel tanks and the big inertial loads on their walls force to use some fuel tanks of smaller sizes. Some fuel tanks can be connected in group. There are two types of the connection of fuel tanks: in parallel, in serial.

At a **parallel connection of fuel tanks** (Fig. 5.6) the pipeline from each fuel tank comes to the fuel supply pipeline into engine. The parallel connection of the fuel tanks enables to control of fuel use flexibly, but thus the installation of BPAs and recuperators is required in each fuel tank. So mass of the fuel system increases. Such scheme increases survivability of the fuel system. Because system can be switched to the other tank if one tank damaged. But this switching required changing of pump modes; it means that the pumps should be multimode.

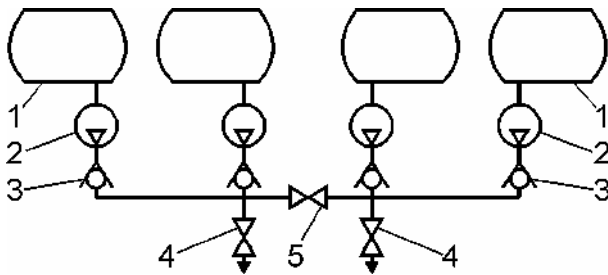
At a **serial connection of fuel tanks** (Fig. 5.7) the pipeline from each fuel tank (except for consumed fuel tank) comes into the consumed fuel tank, and from the consumed fuel tank there is other pipeline to the engine. Such a scheme allows to lower mass of the fuel system, as pumps, transferred fuel from one fuel tank to another have mass smaller than BPA. Besides, the recuperator is arranged only in the consumed fuel tank. This scheme has high survivability, as one consumed fuel tank is easier to protect from damages. Here the temperature equalization of fuel going to the engine is reached. The system reacts to start/stop of the pumps weakly. The system guarantees the fuel reserve. Such scheme is usually called as the scheme with a consumed fuel tank, now it has the widest application.

If there is more than one engine on the airplane, the following schemes of fuel feeding are applied:

**Independent feeding** at which fuel from definite group of fuel tanks is fed to the particular engine (Fig. 5.6 or 5.7).

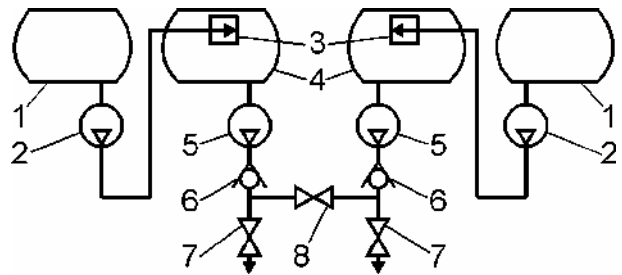
**Centralized feeding** at which fuel from one consumed fuel tank is fed to all engines (Fig. 5.8).

**Centralized-independent feeding** at which fuel from each consumed fuel tank is fed to particular group of engines (for example to the engines arranged on one outer wing) (Fig. 5.9).



**Fig. 5.6. Parallel Connection of Fuel Tanks:**

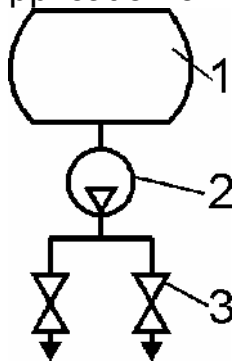
1 — Fuel Tank; 2 — BPA;  
3 — Check Valve; 4 — Shut-Off Valve; 5 — Cross-Feed Valve



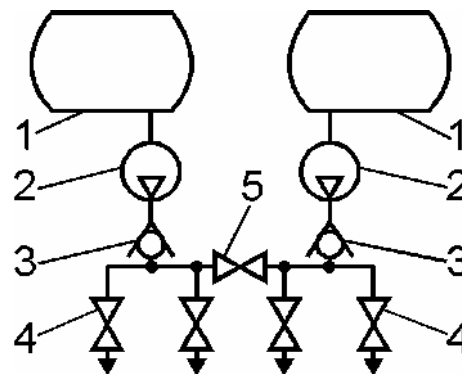
**Fig. 5.7. Serial Connection of Fuel Tanks:**

1 — Fuel Tank; 2 — Transfer Pump;  
3 — Transfer Valve; 4 — Consumed Fuel Tank;  
5 — BPA; 6 — Check Valve; 7 — Shut-Off Valve;  
8 — Cross-Feed Valve

To provide reliability of the fuel feeding to engines, it is applied: the cross-feed, duplicating or redundancy of the booster pump of aircraft (BPA), application of recuperators, etc.



**Fig. 5.8. Centralized Feeding:**  
1 — Fuel Tank; 2 — BPA;  
3 — Shut-Off Valve



**Fig. 5.9. Centralized-Independent Feeding:**  
1 — Fuel Tank; 2 — BPA; 3 — Check Valve;  
4 — Shut-Off Valve; 5 — Cross-Feed Valve

**Cross-feed** (cross-over) of engines with fuel. It is applied on aircraft with more than one engine, except using the centralized feeding. The cross feed lies in the connection of pipelines going from each consumed fuel tank to each engine with the additional pipeline. In a normal situation this **fuel cross-feed line** is blocked by valves (cross-feed valves). At failure of a pump in one of consumed fuel tanks, the cross-feed valve will be open, and fuel from the consumed fuel tank with working pump will start going to both engines. Thus fuel from the group of fuel tanks with the failed booster pump cannot be used, that affects the range and center-of-gravity position. In other situation, when one engine fails, fuel from two consumed fuel tanks can feed the working engine.

**Duplicating of BPA** represents the installation in parallel two working BPA, each of them is capable to provide feeding of the engine by fuel.

**Redundancy of BPA** represents the installation, in parallel with working BPA, another BPA normally stopped, each of them is capable to provide feeding of the engine with fuel. The second BPA can have other type of the drive.

**The recuperator** represents the section of a consumed fuel tank, supplied with devices eliminating outflow of fuel from the pump. In the simplest case these devices are check valves, which pass fuel only into the section. Sometimes the fuel accumulator is applied to supply of fuel feeding in the engine at negative acceleration.

## 5.5. Calculations of Fuel Systems

At designing a fuel system the following problems are solved.

1. Calculation the required capacity of fuel tanks.
2. Arrangement of this fuel on the aircraft.
3. Designing the scheme of fuel feeding to engines and calculation of altitude performance of the fuel system.
4. Designing the scheme and calculation of venting subsystem.
5. Designing the scheme and calculation of dumping subsystem.
6. Designing the scheme of fueling subsystem.
7. Calculation of sequence of fuel usage.
8. Designing the subsystem of supporting the specified center-of-gravity position (for supersonic airplanes).
9. Designing the subsystem of fuel transferring (for tankers).

The first six problems are required within the course project. Let's consider them in more detail.

### 5.5.1. Calculation the Required Capacity of Fuel Tanks

The total capacity of fuel tanks, required to storage fuel onboard the aircraft, can be presented as follows:

$$W_t = W_L + W_{res} + W_{un} + W_{vap} + W_{str} + W_{free}, \quad (5.1)$$

where  $W_L = \frac{m_f}{\rho_{45^\circ}}$  — fuel capacity to fly to the specified range, m<sup>3</sup>;  $m_f$  — fuel mass to fly to the specified range, kg;  $\rho_{45^\circ}$  — density of fuel at a temperature of 45°C, kg/m<sup>3</sup>;

$W_{res} = \frac{m_{res}}{\rho_{45^\circ}}$  — capacity of the fuel reserve, m<sup>3</sup>;

$m_{res}$  — mass of the fuel reserve, kg;  $W_{un} \approx 0.01W_t$  — capacity of unusable fuel, m<sup>3</sup>;  $W_{vap} \approx 0.002(W_L + W_{res})t$  — capacity of fuel vaporized in atmosphere, m<sup>3</sup>;  $t$  — flight time, hour;  $W_{str} \approx (0.02...0.03)W_t$  — capacity of fuel tanks held by structural components and units, m<sup>3</sup>;  $W_{free} \approx 0.02W_t$  —

free volume of the fuel tanks required for expansion of fuel at heating, m<sup>3</sup>.  
 Finally we obtain the required capacity of the fuel tanks onboard the aircraft, m<sup>3</sup>:

$$W_t = \frac{1 + 0.002 t}{(0.94 \dots 0.95)} \left( \frac{m_f + m_{res}}{\rho_{45^\circ}} \right). \quad (5.2)$$

### 5.5.2. Fuel Arrangement Onboard Aircraft

The problem of fuel arrangement onboard the aircraft breaks up on a number of sub problems:

1. Allocate volumes in the airframe.
2. Calculate the total available volume.
3. If this available volume is less than the required capacity, pass to the item 1 (to allocate additional volumes).
4. For four– or six–engined aircraft it is necessary to carry out splitting of fuel tanks into groups according to number of engines.

In an airframe, fuel is usually stored in the interspar part of wing, on combat aircraft fuel can be stored in the fuselage too. Sometimes they apply fixed external tanks (L-39, Be-30). Sometimes they use the volume into the interspar parts of fins (MiG-25, Il-62M).

Calculation of the volume of the fuel tank is carried out by the simple geometrical formulas. This problem will be discussed in practical classes.

Splitting of the fuel tanks to groups can be carried out in the same assumptions using the same formulas. This problem also will be discussed in the practical classes.

### 5.5.3. Altitude Performance of Fuel System

**Altitude performance of fuel system** is the maximum flight altitude of aircraft, at which the fuel system provides uninterrupted fuel feeding to the engine.

**Cavitation** is a process of air–vapor bubbles formation in a depression and their following collapse in a pressure space.

Cavitation is a harmful phenomenon. Thus the two–phase medium (air + fuel) is formed. It results in decrease of mass fuel flow through the pump, in pressure fluctuation, in malfunction of fuel feeding and even in engine stopping. Cavitation usually destroys surfaces, it appears near to which. But as fuel is multicomponent medium, formation and collapse of the bubbles is spread in a time. Thus destruction does not take place. A place of the most probable appearance of the cavitation is a pump inlet. Thus the problem of designing calculation of altitude performance is determination of fuel pumps operation conditions without cavitation. The

problem of checking calculation is determination of altitude performance of the existing fuel system.

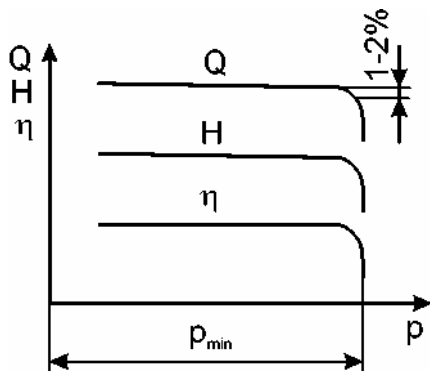
Theoretically, the fuel system failure (cavitation) takes place, when pressure equals to saturated vapor pressure of fuel. Practically, pressure into the pump will be less than the pump inlet pressure. It results in appearing the cavitation earlier, than pressure will be lowered up to saturated vapor pressure at fuel pump inlet. That is at the smaller altitude. Therefore for reliable operation of pumps, it is necessary to create some overpressure at the pump inlet. It is called the cavitation margin  $\Delta P_{cav}$  (Fig. 5.10). Then the minimal required pump inlet pressure providing pump operation without cavitation, is determined by the relation

$$P_{in} \geq P_{in.min} = P_{t4/1} + \Delta P_{cav}.$$

This condition should be satisfied at all flight modes, at all possible load factors and temperatures. This condition is the main condition in the altitude performance calculation.

The cavitation margin and the pressure difference, created by the pump, for pumps of various purposes, are shown in the table below.

To carry out the calculation of the altitude performance of a fuel system, it is necessary to know:



**Fig. 5.10. Cavitation Characteristics of Pump**

a) lay-out diagram of the fuel system in three projections, with the directions and lengths of pipelines and the locations of units;

b) required fuel volume flow depending on the flight mode (M, H, throttling back);

c) fuel properties (density, kinematic viscosity, saturated vapor pressure) depending on temperature;

d) cavitation and pressure properties of pumps.

Type of the pump	$\Delta P_p$ , MPa	$\Delta P_{cav}$ , MPa
BPA	0.1...0.15	0.01...0.025
FBP	0.5...0.6	0.03...0.08
MFP	6...8	0.15...0.3

According to NLGS-3 the calculations are carried out for the most adverse and heavy operating conditions (Fig. 5.11):

1. Flight at the altitude of maximum speed (for airplanes with the ceiling above 11 km, it is 11 km). The engine power is full throttle power or power augmentation. The load factors are the most adverse in their effect

to pressure at FBP inlet (for civil aviation  $n_x = -0.3...0.3$ ;  $n_y = -0.5...3$ ;  $n_z = 0$ ). Designed fuel temperature is maximum (for subsonic airplanes it is more or equal to  $40^\circ\text{C}$ ; for supersonic ones it is more or equal to  $100^\circ\text{C}$ ).

2. Flight on the ceiling. The altitude is: the absolute ceiling — for non-maneuverable; the zoom altitude — for maneuverable. The flight speed corresponds to the ceiling (Fig. 5.11). The engine power is full throttle power or power augmentation. Load factors are  $n_x = 0$ ,  $n_y = 1$ ,  $n_z = 0$ , i. e. inertial losses are absent.

3. Flight on the mode corresponding to maximum fuel consumption. The engine power is full throttle power or power augmentation. Fuel is the most viscous of used on the airplane. Designing fuel temperature is minimal ( $-50...-60^\circ\text{C}$ ). One BPA feeds two engines with the cross-feed valve opened.

Conditions of operating of the fuel system with failed BPA (inertial losses are assumed equal to zero).

4. Takeoff power. The altitude is 2000 m.

5. Cruising power. The altitude is 6000...8000 m.

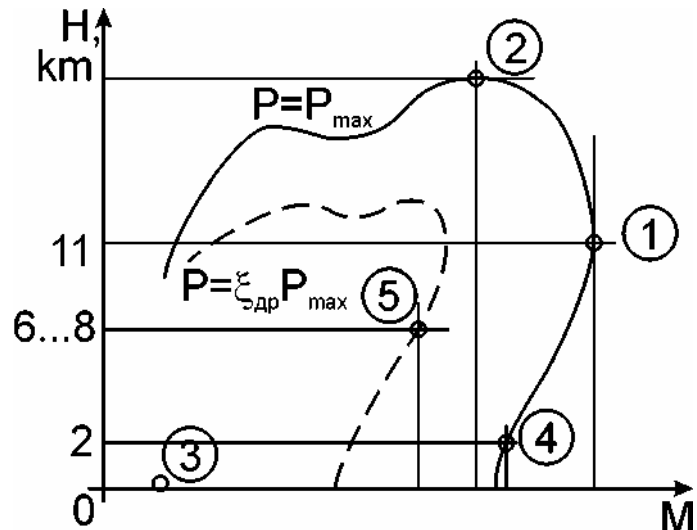


Fig. 5.11. Flight Condition Diagram of Airplane

### 5.5.4. Altitude Performance Calculation

Let's consider two Bernoulli's equations: the first for cross-sections T-T (tank) — I-I (inlet) and the second for cross-sections A-A (aircraft) — E-E (engine) (Fig. 5.12):

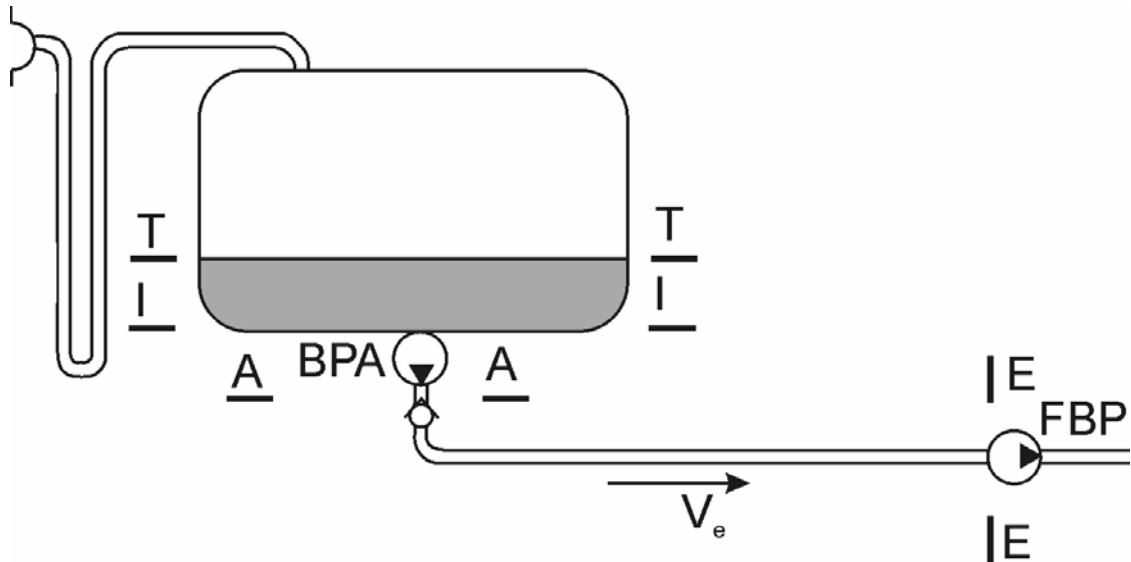
$$P_H + \Delta P_T + \rho_F g y_T + \frac{\rho_F V_T^2}{2} = P_{IN}^{BPA} + \rho_F g y_I + \frac{\rho_F V_I^2}{2} + \Delta P_{hid}^{T-I} + \Delta P_{IN}^{T-I}; \quad (5.3)$$

$$P_{OUT}^{BPA} + \rho_F g y_A + \frac{\rho_F V_A^2}{2} = P_{IN}^{FBP} + \rho_F g y_E + \frac{\rho_F V_E^2}{2} + \Delta P_{hid}^{A-E} + \Delta P_{IN}^{A-E}, \quad (5.4)$$

where  $P_{INmin}^{BPA} = P_{t4/1} + \Delta P_{cav}^{BPA}$ ,  $P_{INmin}^{FBP} = P_{t4/1} + \Delta P_{cav}^{FBP}$ . (5.5)

Let's consider the values, which are included in these equations.





**Fig. 5.12. Fuel Pump Feeding to FBP**

$P_H$  — Pressure at the specified altitude (taken from ISA).

$\Delta P_T$  — Overpressure created in the fuel tank by the ram pressure (opened venting) or by the pressurization system (closed venting). Pressurization can be carried out by bottles with neutral gas or by the engine compressor. Value of the ram pressure, used for creation of overpressure in a fuel tank, depends on the angle  $\varphi$  of vent air intake tube cut:

$$\Delta P_T = q \cdot K_\varphi = q \cdot \cos 1.5(90 - \varphi),$$

where  $q$  — air ram pressure;  $K_\varphi$  — utilization factor of the ram pressure (this formula is valid for  $\varphi = [30 \dots 90]^\circ$  on subsonic flight speed) (Fig. 5.13).

$\rho_F, P_{t4/1}$  — density and saturated vapor pressure of fuel.

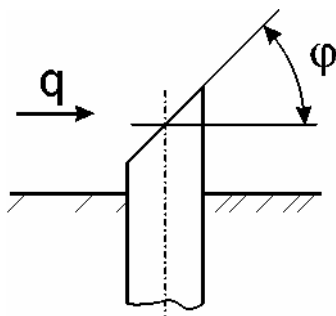
$y_i, V_i$  — level and fuel speed in the corresponding point of the pipeline.

$g$  — acceleration of gravity.

$\Delta P_{hyd}^{i-k}$  — hydraulic losses into the pipeline section i-k. They are summed up of friction losses  $\Delta P_{fr}$  and form losses  $\Delta P_{form}$ . Pressure losses caused by friction are determined by formula

$$\Delta P_{fr}^{i-k} = \lambda \frac{l_{i-k}}{d} \frac{\rho_F V_i^2}{2},$$

where  $\lambda$  — friction drag coefficient;  $l_{i-k}$  — length of the pipeline segment i-k;  $d$  — diameter of the pipeline;  $\rho_F$  — density of fuel;  $V_i$  — fuel speed in the point  $i$  of the pipeline (if volume flow  $W$  of fuel is known, the speed



**Fig. 5.13. Vent Air Intake**

is determined by formula  $V_i = 4W/\pi d^2$ ).

The friction drag coefficient  $\lambda$  depends on the fuel flow mode determined by the Reynolds' number

$$\text{Re} = \frac{V_F d}{\nu} = \frac{4W}{\pi d \nu},$$

where  $\nu$  — factor of kinematic viscosity.

For the laminar fuel flow mode, when  $\text{Re} \leq 2300$ , the friction drag coefficient  $\lambda = 64/\text{Re}$ .

For the turbulent flow mode, when  $\text{Re} \in (3000; 10^5)$ , the friction drag coefficient  $\lambda = 0.3164/\sqrt[4]{\text{Re}}$ ; when  $\text{Re} \in (10^5; 5 \cdot 10^6)$ , the friction drag coefficient  $\lambda = 0.09/\sqrt[3]{\text{Re}}$ .

For flexible hoses the friction drag coefficient should be increased by 30%  $\lambda_{F.H} = 1.3\lambda$ .

Form losses appear when changing the cross-section or direction of the stream, that promotes vortex generation. Form pressure losses are determined by formula

$$\Delta P_{form}^{i-k} = \sum \xi_{form} \frac{\rho_F V_i^2}{2},$$

where  $\xi_{form}$  — factors of form losses into the segment i-k;  $V_i$  — the greater fuel speed value in the location of form losses. Values of the factors of form losses are taken from handbooks. In the course project, they are taken from methodical textbooks.

Inertial pressure losses are caused by inertial forces in fuel pipelines. These forces appear at motion of aircraft with acceleration. They are determined by formula

$$\Delta P_{IN}^{i-k} = \rho_F g [n_x l_x + (n_y - 1) l_y + n_z l_z],$$

where  $n_i$  — load factor along an  $i$  axis;  $l_i$  — projection of the segment of the pipeline i-k on an  $i$  axis, taking into account a sign.

From the first Bernoulli's equation (5.3), it is possible to obtain the altitude performance of the fuel system with pump feeding (neglecting the fuel speed in the tank  $V_T$ ):

$$P_H = P_{INmin}^{BPA} + \Delta P_{hyd}^{T-I} + \Delta P_{IN}^{T-I} - \rho_F g (y_T - y_I) - \Delta P_T + \frac{\rho_F V_I^2}{2}. \quad (5.6)$$

To simplify and in view of their small value, we neglect the hydraulic  $\Delta P_{hyd}^{T-I}$  and the inertial  $\Delta P_{IN}^{T-I}$  losses, the difference of levels of fuel  $y_T - y_I$  and the velocity at BPA inlet  $V_I$ , when calculating the altitude performance of BPA:

$$P_H \approx P_{INmin}^{BPA} - \Delta P_T; \quad P_{IN}^{BPA} = P_H + \Delta P_T \geq P_{INmin}^{BPA}. \quad (5.7)$$

For designing calculation we use the inequality:

$$\Delta P_T \geq P_{INmin}^{BPA} - P_H. \quad (5.8)$$

From Bernoulli's equation for cross-sections A-A and E-E (5.4), it is possible to obtain FBP inlet pressure:

$$P_{IN}^{FBP} = P_{OUT}^{BPA} - \Delta P_{hyd}^{A-E} - \Delta P_{IN}^{A-E} + \rho_F g(y_A - y_E) + \frac{\rho_F}{2}(V_A^2 - V_E^2) \geq P_{INmin}^{FBP}. \quad (5.9)$$

Writing down the difference of BPA outlet and inlet impact pressures, we obtain the pressure difference created by the pump:

$$\Delta P_{BPA} = (P_{OUT}^{BPA} - P_{IN}^{BPA}) + (\rho_F g y_A - \rho_F g y_I) + \left( \frac{\rho_F V_A^2}{2} - \frac{\rho_F V_I^2}{2} \right), \quad (5.10)$$

from here, we obtain the formula for BPA outlet static pressure  $P_{OUT}^{BPA}$ :

$$P_{OUT}^{BPA} = (P_{IN}^{BPA} + \Delta P_{BPA}) + \rho_F g(y_I - y_A) + \frac{\rho_F}{2}(V_I^2 - V_A^2). \quad (5.11)$$

Let's substitute (5.11) in (5.9), then we obtain inlet FBP pressure:

$$P_{IN}^{FBP} = P_{IN}^{BPA} + \Delta P_{BPA} - \Delta P_{hyd}^{A-E} - \Delta P_{IN}^{A-E} + \rho_F g(y_I - y_E) + \frac{\rho_F}{2}(V_I^2 - V_E^2) \geq P_{INmin}^{FBP}. \quad (5.12)$$

From the equation (5.6) we obtain BPA inlet static pressure  $P_{IN}^{BPA}$ :

$$P_{IN}^{BPA} = P_H + \Delta P_T - \Delta P_{hyd}^{T-I} - \Delta P_{IN}^{T-I} + \rho_F g(y_T - y_I) - \frac{\rho_F V_I^2}{2}. \quad (5.13)$$

Substituting  $P_{IN}^{BPA}$  from (5.13) in (5.12), we obtain the formula for checking calculation:

$$P_{IN}^{FBP} = P_H + \Delta P_T + \Delta P_{BPA} - \Delta P_{hyd}^{T-E} - \Delta P_{IN}^{T-E} + \rho_F g(y_T - y_E) - \frac{\rho_F V_E^2}{2} \geq P_{INmin}^{FBP}, \quad (5.14)$$

or, if to express  $P_H$ , for obtaining the altitude performance:

$$P_H = P_{INmin}^{FBP} + \Delta P_{hyd}^{T-E} + \Delta P_{IN}^{T-E} - \rho_F g(y_T - y_E) - \Delta P_T - \Delta P_{BPA} + \frac{\rho_F V_E^2}{2}. \quad (5.15)$$

For designing calculation (BPA selection) it is possible to obtain:

$$\Delta P_{BPA} \geq P_{INmin}^{FBP} - P_H - \Delta P_T + \Delta P_{hyd}^{T-E} + \Delta P_{IN}^{T-E} - \rho_F g(y_T - y_E) + \frac{\rho_F V_E^2}{2}. \quad (5.16)$$

Thus, for the pump fuel feeding to FBP we use the equation (5.7), for checking calculation by the condition of altitude performance of BPA; the equation (5.14) or (5.15) is used, for checking calculation by the condition of altitude performance of FBP. In designing calculation we use inequalities: (5.8) for selection of overpressure in the fuel tank and (5.16) for selection BPA.

To increase the altitude performance of the fuel system, we can do the following:

1. Applying the fuel with the minimal saturated vapor pressure.
2. Applying pumps with good cavitation characteristics. That is with the small cavitation margin.
3. Protecting fuel from heating:
  - heat-insulating fuel tanks, applying in flight cooling system;
  - feeding fuel first of all from the most heated fuel tanks;
  - fuelling the aircraft with cooled fuel.
4. Reducing length of pipelines (by rational arrangement).
5. Reducing the hydraulic form losses.
6. Using the closed vent system (pressurization from the compressor or from the neutral gas system).
7. Degassing the fuel before filling.

## 5.6. Venting and Pressurization of Fuel System

The purpose of venting and pressurization system is providing the pressure inside the fuel tanks in the definite limits, ensuring fuel feeding to engines, fuel filling and dumping. To provide the fuel system operation without cavitation, decreasing of fuel vaporation, keeping the shape and strength of the fuel tanks (a thin-walled shell structure), a definite overpressure shall be supported in the fuel tanks. Otherwise during feeding or fuel dumping, pressure in the fuel tank will be decreased. It complicates feeding and makes contortion of the fuel tank possible. At closed fueling the venting system shall provide a free vent of air from the fuel tanks. Otherwise the fuel tank can be torn by overpressure.

Depending on type and purpose of the aircraft, the vent system is designed under various schemes. The system connecting the tank air-space with an atmosphere is called **an open vent system**. The system connecting the tank airspace with any source of air (the compressor of the engine, bottles with gas) is called **a closed vent system**. The system connecting the tank airspace either with an atmosphere, or with any source of air, depending on flight conditions, is called **a combined vent system**.

Venting of fuel tanks group can be separate (personal) or joint (collective). In the latter case, the consecutive or parallel connection of fuel tanks is possible (Fig. 5.14).

Essential disadvantage of the separate venting is different pressures in fuel tanks. This pressure difference causes uncontrolled overflow of fuel between the fuel tanks. It disturbs the specified sequence of fuel usage and effects on aircraft center-of-gravity position.

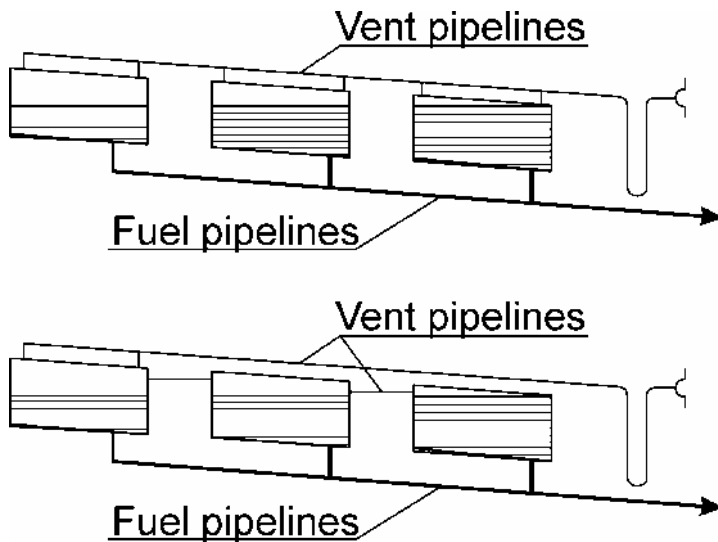


Fig. 5.14. Venting of Fuel Tanks Group

The connection of group of fuel tanks with the over-all venting pipeline makes the system more complicated and heavier. But thus equal pressure in fuel tanks is provided.

Input of pipelines into the fuel tanks should be designed so, that at maximum fuel quantity, fuel tank were provided with direct (not through the fuel) venting of the tank airspace with an atmosphere or any

source of air.

If at changes of aircraft attitude, various points of the fuel tank appear upper (that is probably in fuel tanks of irregular shape), for greater reliability of venting, input in the fuel tank is doubled.

At the **open vent system**, the air passes through small external pipes — air intakes of various types (Fig. 5.15).

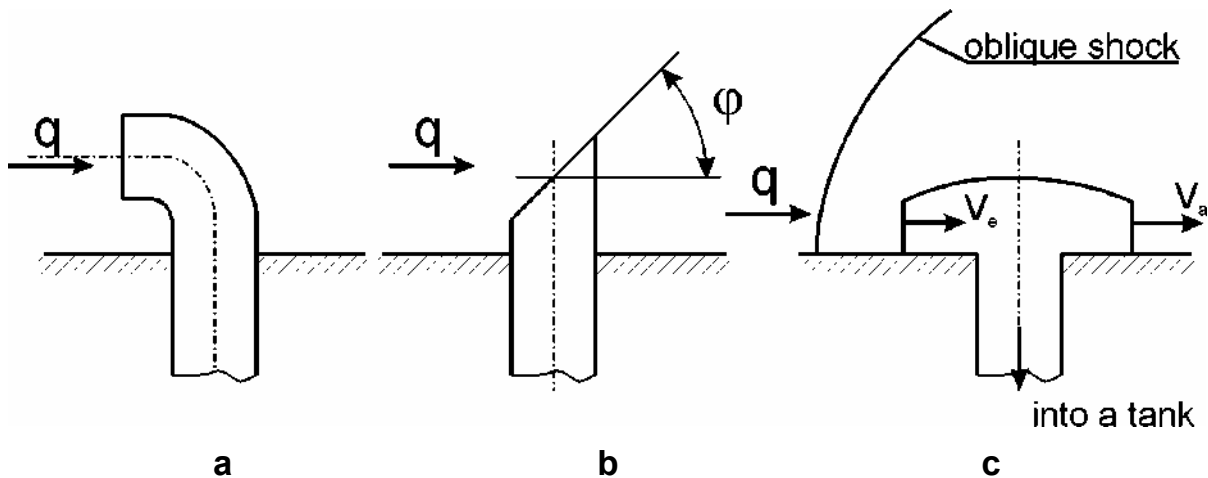


Fig. 5.15. Types of Vent Air Intakes

The air intake (Fig. 5.15, a) fully uses the ram pressure. Thus  $K_\phi = 1$ . However for up-to-date aircraft maximum ram pressure, which can be reached in flight, considerably exceeds required fuel tank pressurization. To decrease used ram pressure, air intakes with a cant (Fig. 5.15, b) are applied, or special safety valves are installed. Selection of air intake is carried out after comparison of various versions.

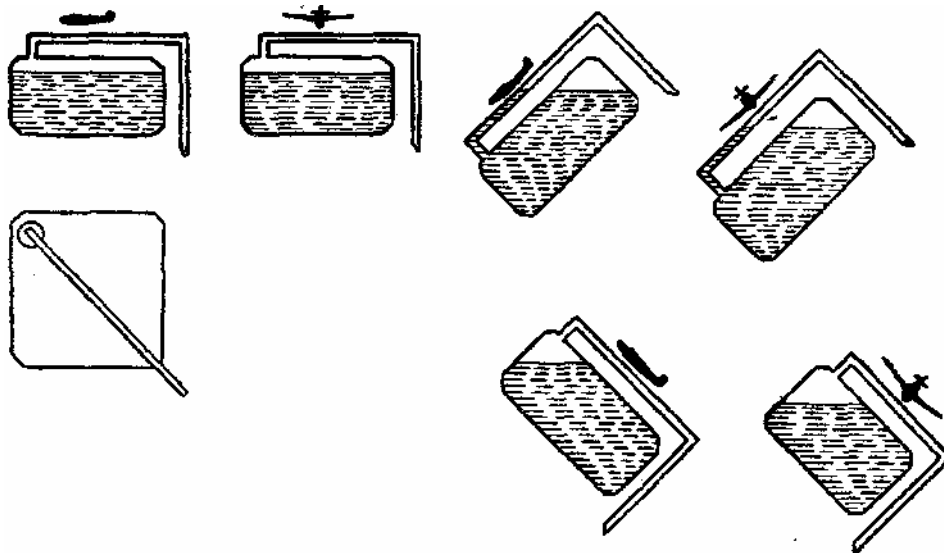
At supersonic flight speed, an oblique shock appears in front of the vent air intake. Behind the oblique shock, the increased pressure is

formed. It results that the overpressure in the tank can appear too big. In this case it is necessary to use safety valves.

Other solution of this problem is application of the air intake with blowing out. Such air intake can work at both supersonic, and subsonic flight speeds (Fig. 5.15, c).

The vent air intake is usually arranged on a top surface of the aircraft, to prevent dust and mud getting into the fuel system on ground jointly with air.

To prevent flowing out of fuel through a vent system at changes of aircraft attitude, the venting pipeline is designed as a loop in a vertical plane (Fig. 5.16). However in the lower part of such loop–shape segment of pipeline, fuel is accumulated and overlaps the venting. For collecting and removing the fuel from the vent system in such cases, drain tanks are installed.



**Fig. 5.16. Arrangement of Venting Pipeline**

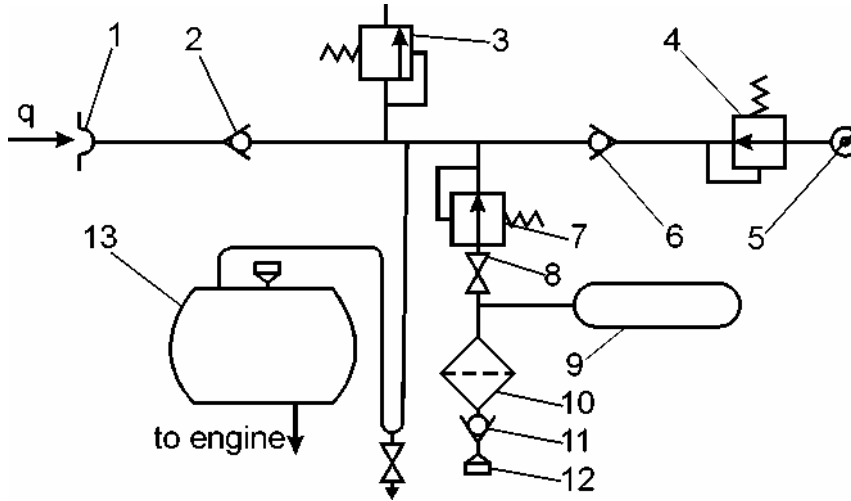
It is necessary to protect the vent air intakes from icing.

At the **closed vent system**, the fuel tank pressurization is carried out by air from compressor of the engine, air or neutral gas from onboard bottles. The closed vent system allows supporting the required pressure in over–fuel space. However disadvantages of the closed vent system is big mass of bottles and lower survivability of fuel tanks in case of damage.

In case of gas feeding from bottles, nitrogen is usually used as a neutral gas. Carbon dioxide is seldom used. It is well dissolved in fuel; therefore its required consumption is somewhat higher. For aircraft with piston engines, they sometimes use filling of fuel tanks with exhaust gas. Thus it is not necessary to keep gas in bottles. However exhaust gas has high temperature and contains vapor of water and mechanical impurities. For cooling and filtering the exhaust gases, special filters, coolers and

condensers are installed in the system. For jet aircraft, it is impossible to use exhaust gas due to a high content of oxygen within them.

When ram pressure, at the open vent system, is insufficient to provide the required fuel tank pressurization in all flight regimes, **combined vent systems** are applied (Fig. 5.17).



**Fig. 5.17. Scheme of Combined Vent System**

In this case, air in air intake from atmosphere 1 passes through the check valve 2 and the safety valve 3, then goes into the fuel tank 13. As soon as pressure in the vent system has lowered below the specified value, the reducing valve 4 opens and air from engine compressor 5 is directed towards the fuel tank. The safety valve 3 serves for protection of the vent system at failure of the reducing valve 4.

In the check valve 2, the small hole for pass in warm air from the compressor of engine is done. It prevents icing and provides blowing out of condensate, water and snow, which get into the open vent air intake.

Air going from the compressor of the engine, has high temperature, therefore it is cooled in the air-to-air radiator in front of inlet of the vent system.

Instead of air intake from engine compressor 5, they sometimes use bottles with neutral gas 9. In this case, filler neck 12, check valve 11 and filter 10 are used for their feeling on ground. For switching on the neutral gas system, it is necessary to open shut-off valve 8, after that gas from bottle 9 starts going to fuel tank 13 through the reducing valve 7.

### 5.6.2. Calculation of Vent System

Efficiency of the vent system is provided by calculations, as a result of which diameter of vent pipeline ( $d_v$ ) and the angle ( $\varphi$ ) of the vent air intake are determined.

The critical load conditions, causing the heaviest operating condi-

tions of vent system, are:

- diving of the airplane;
- fuel dumping;
- pressure fueling (including inflight refueling).

Let's consider the calculation of venting at diving in detail.

At diving of an airplane the atmospheric pressure strongly grows, therefore enough air supply should go through the vent system into the fuel tanks, in order to prevent their contortion due to the pressure difference outside and inside of the fuel tanks. This critical load conditions is the main for the maneuverable airplanes, admitting steep glide, up to diving.

Let's consider steady diving of airplane from the altitude  $H_1$  with the horizontal speed  $V_x$  and the vertical speed  $V_y$ . Within one second the airplane is lowered up to the altitude  $H_2 = H_1 - V_y \cdot 1$ . Due to the difference of density at these altitudes  $[\rho(H_2) - \rho(H_1)]$  for supplying required overpressure in the fuel tank  $\Delta P_T$  at the same time  $t = 1c$  it is necessary to inject the following volume of air (consumption) into the fuel tank:

$$W_a = \frac{\Delta M_a}{t \cdot \rho_{av}} = \frac{k_T A_T [\rho(H_2) - \rho(H_1)]}{t \cdot \rho_{av}}. \quad (5.17)$$

In the formula (5.17) the following factors and values are designated:  $k_T$  — hollow factor of fuel tank, that is taken 0.75 — for consumed fuel tanks and to 1 — for all others;  $A_T$  — volume of the calculated fuel tank;  $\rho_{av}$  — average density of air between altitudes  $H_1$  and  $H_2$ , which can be taken equal

$$\rho_{av} = \frac{\rho(H_2) + \rho(H_1)}{2}. \quad (5.18)$$

From the other hand, this air consumption is equal to the product of cross-sectional area of the vent pipeline by speed of air into it:

$$W_a = \frac{\pi \cdot d_v^2}{4} V_v. \quad (5.19)$$

Whence we obtain expression for the diameter of vent pipeline:

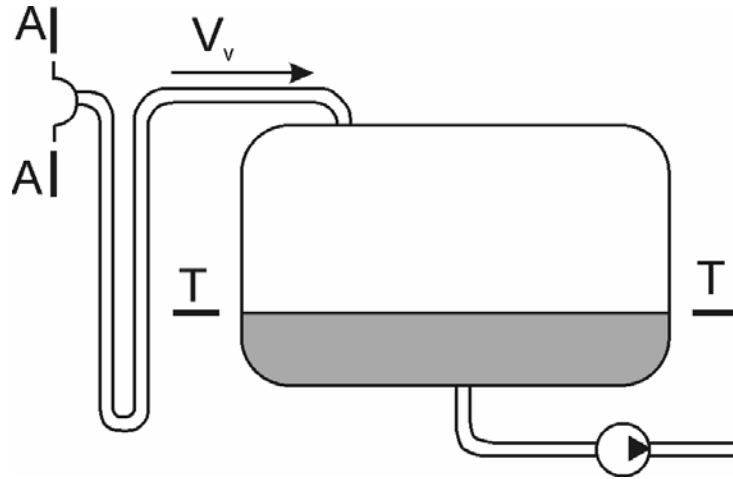
$$d_v = \sqrt{\frac{4 \cdot W_a}{\pi \cdot V_v}}. \quad (5.20)$$

To find  $V_v$ , we write down Bernoulli's equation for cross-sections A-A (atmosphere) and T-T (tank) of vent system (Fig. 5.18):

$$P_H + \frac{k_\varphi \rho_{av} (V_x^2 + V_y^2)}{2} + \rho_{av} g y_A = P_H + \Delta P_T + \frac{\rho_{av} V_T^2}{2} + \rho_{av} g y_T + \xi_{equ} \frac{\rho_{av} V_v^2}{2} + \Delta P_{IN}.$$

Neglecting the difference of levels of cross-sections ( $y_A - y_T$ ); the speed of fuel in the fuel tank  $V_T$  and inertial losses  $\Delta P_{IN}$ , we obtain:





**Fig. 5.18. Design Model of Venting at Diving**

$$\frac{k_{\varphi} \rho_{av} (V_x^2 + V_y^2)}{2} = \xi_{equ} \frac{\rho_{av} V_v^2}{2} + \Delta P_T. \quad (5.21)$$

Whence, speed of air into the vent pipeline:

$$V_v = \sqrt{\frac{2 (q_{\varphi} - \Delta P_T)}{\rho_{av} \xi_{equ}}}, \quad (5.22)$$

where  $\xi_{equ}(d_v) = \left( \lambda_v \frac{l_v}{d_v} + \sum \xi_{form} \right)$  — equivalent factor of the hydraulic pressure losses,  $q_{\varphi} = \frac{k_{\varphi} \rho_{av} (V_x^2 + V_y^2)}{2}$  — ram pressure in the vent air intake.

Substituting (5.22) in (5.20), we obtain the equation for diameter of the vent pipeline:

$$d_v = \sqrt{\frac{4W_a}{\pi} \sqrt{\frac{\rho_{av} \xi_{equ}(d_v)}{2 (q_{\varphi} - \Delta P_T)}}}, \quad (5.23)$$

where  $l_v$  — the total length of the vent pipeline from the air intake up to the fuel tank;  $\sum \xi_{form}$  — the total factor of form losses in this pipeline;  $\lambda_v$  — the friction drag coefficient, determined by the same formulas, but for air.

Solving the equation (5.23) by any numerical or a graphical method, we find the required diameter of the vent pipeline, from the condition of diving  $d_v^{(1)}$  (Fig. 5.19).

Calculations of the vent system at dumping and at the pressure fueling are carried out in the same way. Then we should use the biggest of them.

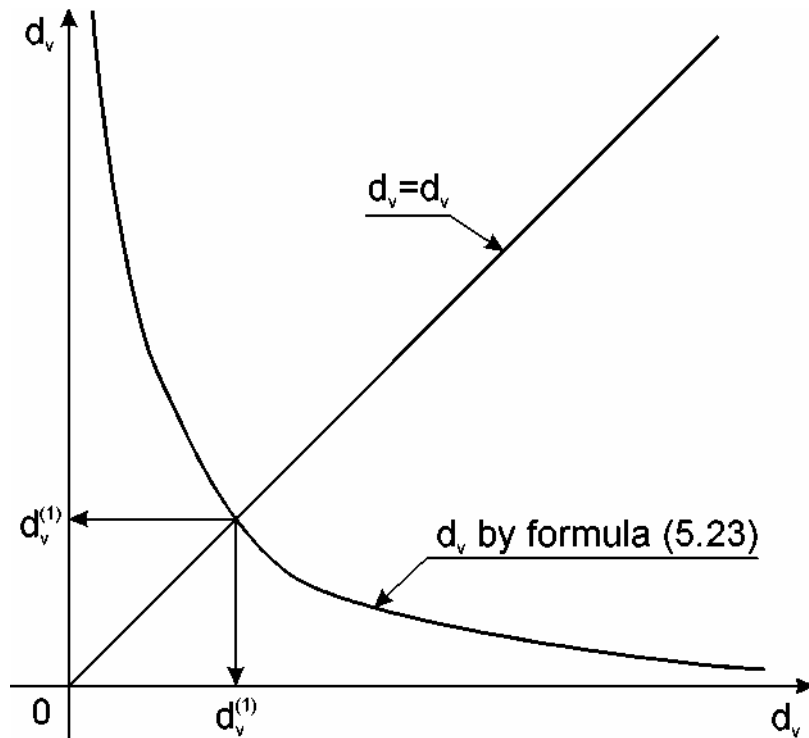


Fig. 5.19. Solving Equation (5.23) by Graphical Method

### 5.7. Designing the scheme and calculation of fuel dump system

Many transport and passenger airplanes are equipped with in flight fuel dump system. Dumping of fuel is carried out in the following cases:

- Before emergency landing, if by conditions of structural strength there is a limitation on the maximum landing mass.
- If presence of fuel onboard is undesired by fire safety (landing with the failed landing gear or engine, etc.).
- If landing is carried out on runway of insufficient length.

The fuel dump system shall provide:

1. Dumping the specified volume of fuel during specified period. The time of dumping is normalized and according to NLGS-3 depends on volume of the fuel tanks:

- t less than 7 minutes  $V_T < 10000 \text{ dm}^3$ ;
- t less than 12 minutes  $V_T < 20000 \text{ dm}^3$ ;
- t less than 15 minutes  $V_T < 30000 \text{ dm}^3$ .

2. Supporting the centre-of-gravity position within acceptable limits.

3. Fire safety. I. e. fuel at dumping must not get into the engine or on other parts of the aircraft.

Depending on arrangement of the fuel system, sizes and capacity of fuel tanks, to provide these requirements, dumping of fuel is designed **by gravity, forced out or by pumps**.

Let's consider calculation of fuel dump system.

During time  $d\tau$  the volume of fuel in the fuel tank is changed by value:

$$F_d V_d d\tau = -F_T dH, \quad (5.24)$$

where  $F_d = \pi d_d^2 / 4$  — cross-sectional area of the dumping pipeline;  $V_d$  — speed of fuel into the dumping pipeline;  $F_T$  — cross-sectional area of the fuel tank by the horizontal plane at  $H$  level;  $H$  — fuel level into the fuel tank at the time  $\tau$ , counting from the exit point of the dumping pipeline in atmosphere (Fig. 5.20).

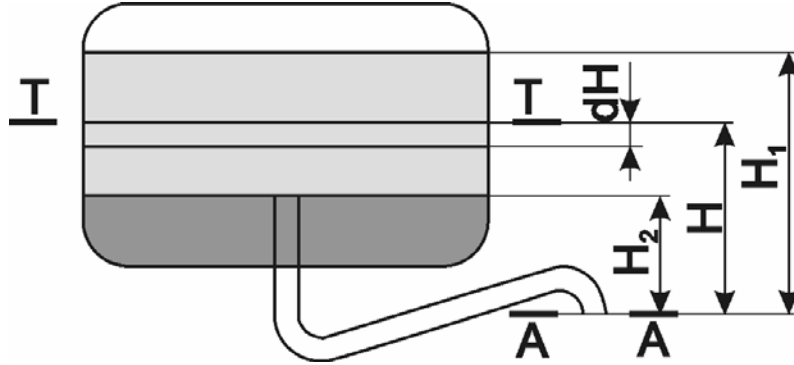


Fig. 5.20. Scheme of Dumping

Experience shows, that for fuel tanks of any shape, it is possible to carry out calculation by average area of the fuel tank:

$$F_T \approx F_{av} = \frac{V_T}{H_1 - H_2},$$

where  $H_1$  and  $H_2$  — initial and final levels of fuel in the fuel tank, counting from the exit point of the dumping pipeline in an atmosphere, Fig. 5.21.

To determine the speed of fuel in the dumping pipeline, we write Bernoulli's equation for cross-sections A-A and T-T:

$$P_H + \Delta P_T + \rho_F g H + \frac{\rho_F V_T^2}{2} = P_H + \rho_F g 0 + \frac{\rho_F V_d^2}{2} + \xi_{equ} \frac{\rho_F V_d^2}{2} + \Delta P_{IN},$$

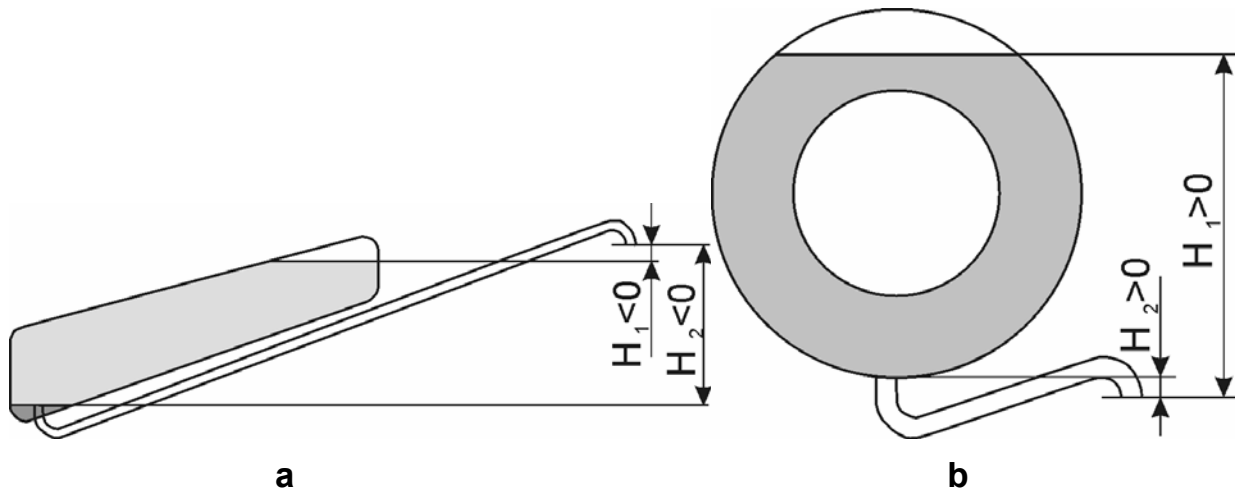
neglecting the speed of fuel in the fuel tank  $V_T$  and inertial pressure losses  $\Delta P_{IN}$ , we obtain:

$$\rho_F g H + \Delta P_T = (1 + \xi_{equ}) \frac{\rho_F V_d^2}{2};$$

whence: 
$$V_d = \frac{\sqrt{2g}}{\sqrt{1 + \xi_{equ}}} \sqrt{H + \frac{\Delta P_T}{\rho_F g}}. \quad (5.25)$$

Let's substitute the expression for dumping speed (5.25) in the equation of volumes (5.24)

$$F_d \frac{\sqrt{2g}}{\sqrt{1 + \xi_{equ}}} \sqrt{H + \frac{\Delta P_T}{\rho_F g}} d\tau = -F_T dH,$$



**Fig. 5.21. Law of Signs of Fuel Levels**

$$d\tau = -\frac{F_T}{F_d} \frac{\sqrt{1 + \xi_{equ}}}{\sqrt{2g}} \frac{dH}{\sqrt{H + \Delta P_T / \rho_F g}}$$

Integrating the last expression, we find:

$$\tau = \frac{F_T}{F_d} \frac{\sqrt{1 + \xi_{equ}}}{\sqrt{2g}} \int_{H_2}^{H_1} \frac{dH}{\sqrt{H + \Delta P_T / \rho_F g}},$$

whence

$$\tau = \frac{2F_T}{F_d} \frac{\sqrt{1 + \xi_{equ}}}{\sqrt{2g}} \left[ \sqrt{H_1 + \frac{\Delta P_T}{\rho_F g}} - \sqrt{H_2 + \frac{\Delta P_T}{\rho_F g}} \right]. \quad (5.26)$$

By formula (5.26) it is possible to determine the dumping time at known diameter of the pipeline (checking calculation) or to select the diameter of the pipeline by the specified dumping time (designing calculation) (Fig. 5.22).

At high fuel dump rate, an air funnel may appear at the dumping connection. It causes increase of the hydraulic pressure losses and increases the dumping time. In such cases the confuser outlet is installed above the dumping connection (Fig. 5.23). This outlet provides smooth variation of the flow cross-section area and prevents hitting of air into the dumping pipeline.

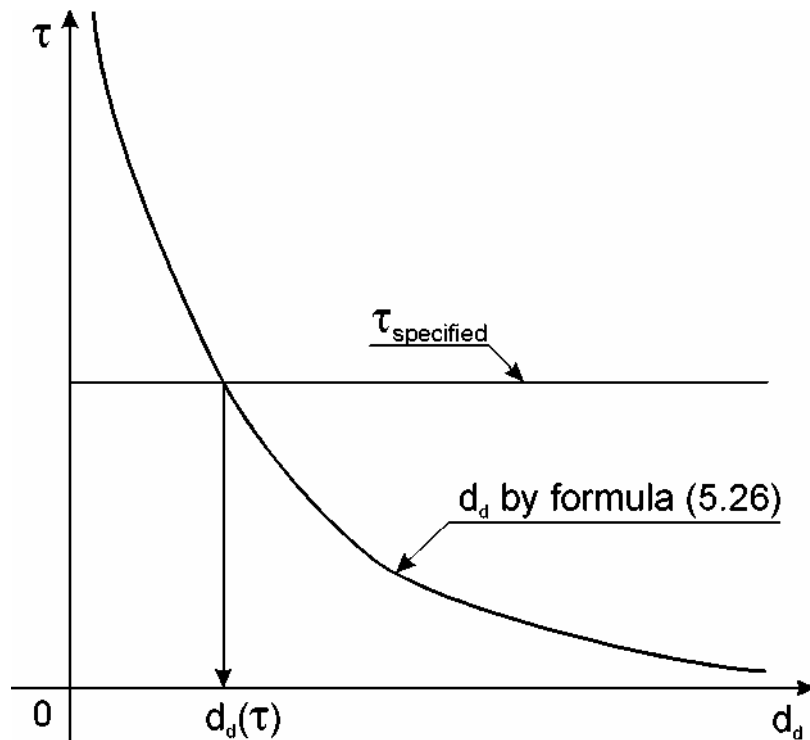
In the consumed fuel tank, the outlet connection of dumping pipeline is expedient to design outstanding inside the fuel tank (ref. Fig. 5.18). It provides the guaranteed fuel reserve for landing.

## 5.8. Fueling System

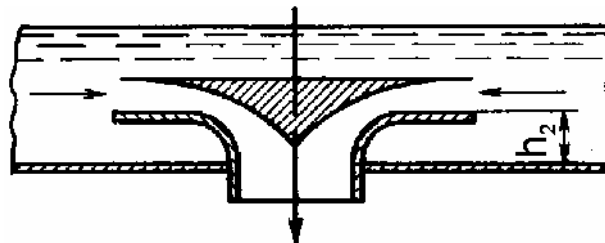
There are two **ways** of fueling: overwing fueling and pressure one.

At **overwing fueling**, the filtered fuel is fed by the flexible hose through the fuel discharge nozzle directly to the filler neck (Fig. 5.24) from the fueling device. It is arranged on the upper side of the fuel tank. In case

of grouping of fuel tanks, fuel can overflow into the other fuel tanks by connecting pipes. The filler necks are designed according to used standards. The fueling time of all fuel tanks should not exceed the specified value (about 10 minutes).



**Fig. 5.22. Determining Dumping Pipeline Diameter Depending on Specified Dumping Time**



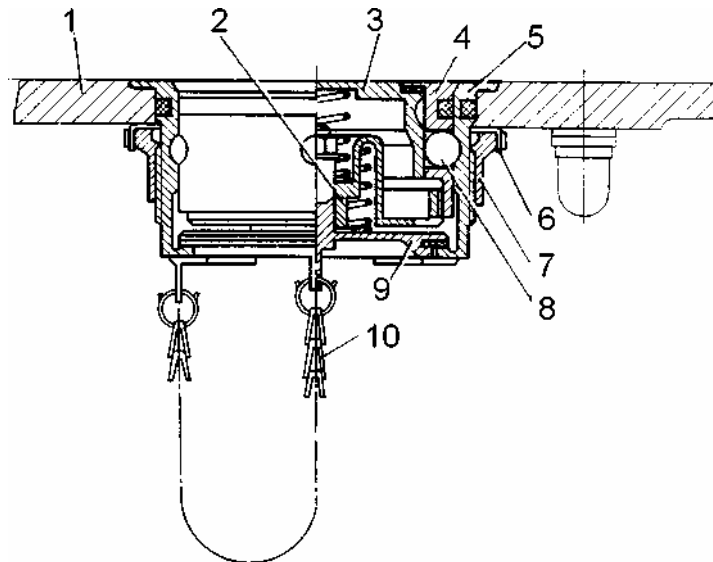
**Fig. 5.23. Confuser Outlet of Dumping Connection**

Overwing fueling has a number of disadvantages.

[–] Long fueling time. (The fueling rate in connecting pipelines is low. It is required to open and close each filler neck, to move the discharge hose, to switch on and off the pump of priming device for filling.)

[–] Necessity of location of ground maintenance personnel near the filler necks arranged usually on the wing. Ladders, step-ladders, long hoses are required for this purpose. These movements lead to damages of paint coating of the wing. In winter these movements on icy wing surface are dangerous for the people.

[–] Fire danger is increased at fuel vaporation.



**Fig. 5.24. Filler Neck for Overwing Fueling**

**1 — Skin; 2 — Bush; 3 — Button; 4 — Plug Casing; 5 — Filler Neck Casing; 6 — Locking Washer; 7 — Nut; 8 — Ball; 9 — Valve; 10 — Chain**

[–] A moisture and a dust can probably get into the fuel tank through the upper filler necks.

These disadvantages can be removed at **pressure fueling**. In this case the filtered fuel is fed by the flexible hose to the fueling connection (Fig. 5.25) of the aircraft from the fueling device.

The pressure fueling provides a number of advantages.

[+] Low fueling time.

[+] The fueling is carried out through one or two fueling connection. (Therefore such fueling may also be referred to as «single–point pressure fueling»).

[+] To decrease the length of discharge hoses and for maintainability, these fueling connections are arranged in the lower parts of aircraft. Thus people movement on wings is not needed.

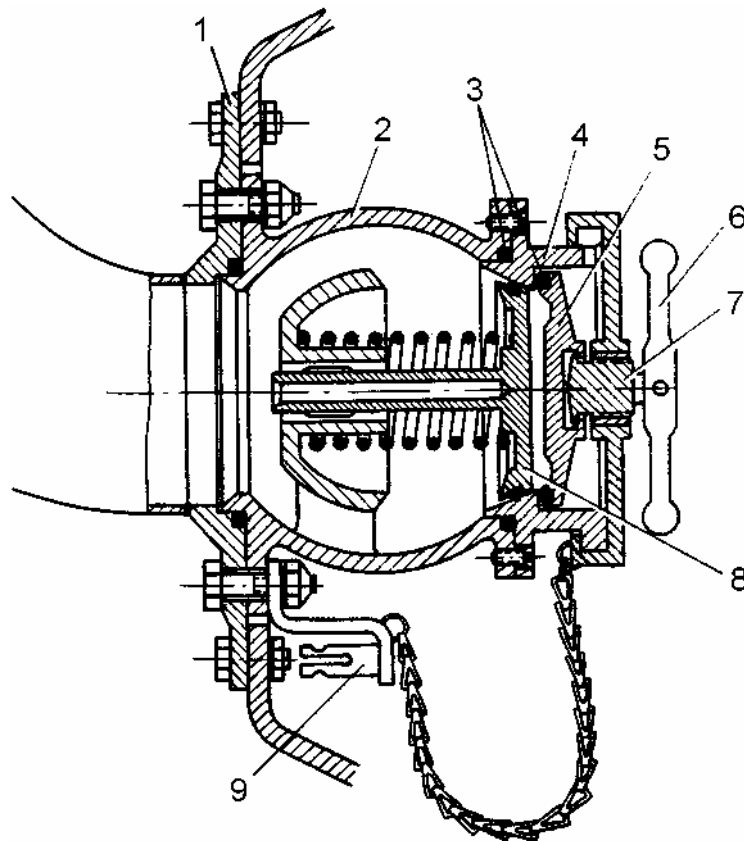
The fueling connections and nozzles of discharge hoses are designed by the international standards.

The sequence of the pressure fueling can be different (serial or joint). Serial tank fueling takes more time, than joint. But the serial fueling provides a capability to fill not all fuel tanks with fuel, but only specified.

Thus there are also a number of disadvantages:

[–] A high speed of fuel flow within the pipelines (more than  $25 \text{ dm}^3/\text{s}$ ), a big length of fueling pipelines and presence of special control devices cause in the big hydraulic pressure losses. Hence, the high power of the pumps installed at fueling devices is required.

[–] The fueling pipelines and the control devices are arranged on-board the aircraft and increase its mass.



**Fig. 5.25. Filler Neck for Single-Point Pressure Fueling:**  
**1 — Adapter Connector; 2 — Casing; 3 — Sealing; 4 — Flange;**  
**5 — Covering; 6 — Handle; 7 — Rod; 8 — Valve; 9 — Grounding Jack**

[–] The maximum fuel quantity is impossible to be filled by pressure fueling owing to response of maximum level valves. Therefore devices for overwing fueling are kept onboard the aircraft.

## **5.9. Structural Components of Fuel System**

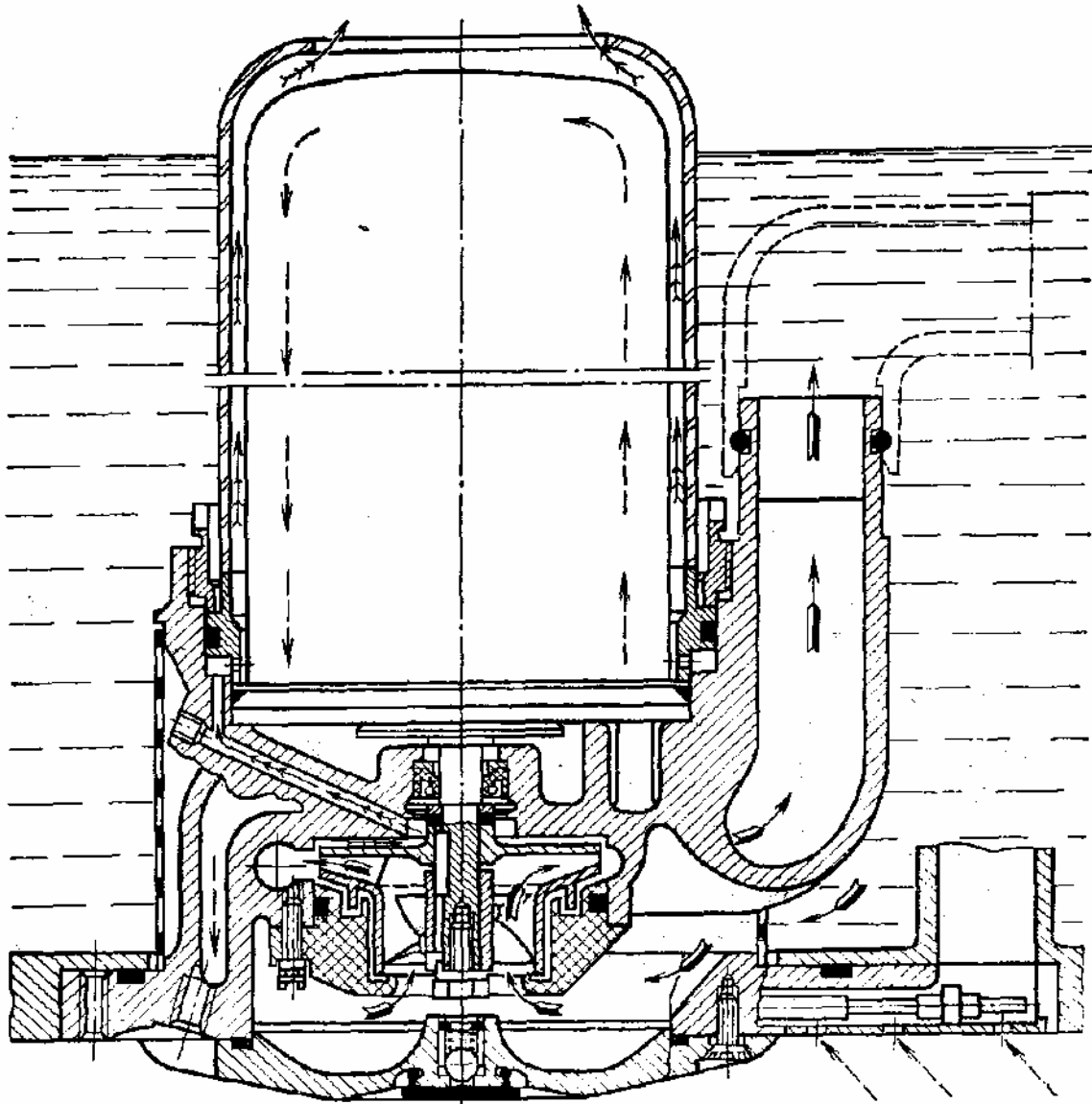
### **5.9.1. Fuel Pumps**

Fuel pumps can have various types of the drive.

Electro-powered centrifugal pumps represent monoblock version of the pump and the electric motor (Fig. 5.26).

The **direct-current drive** provides ease of regulating of the consumption and high reliability.

The **alternating-current drive** provides high efficiency at power of pumps more than 1.5 kW. Absence of collectors decreases fire danger into strongly rarefied atmosphere at high altitudes at the increased moisture. A disadvantage of the alternating-current drive is fixed (or stepwise changeable) rotational speed.

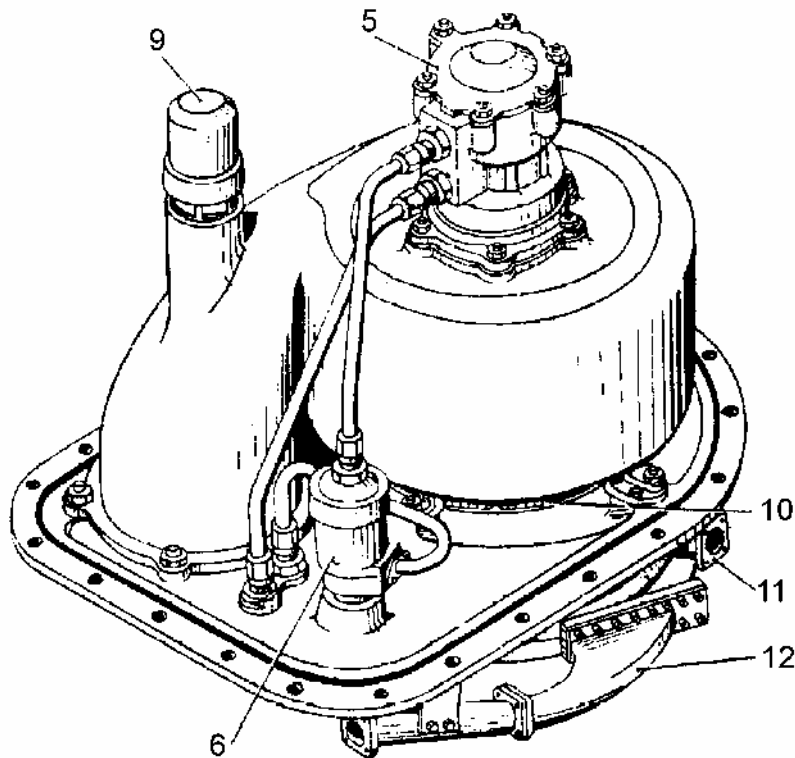
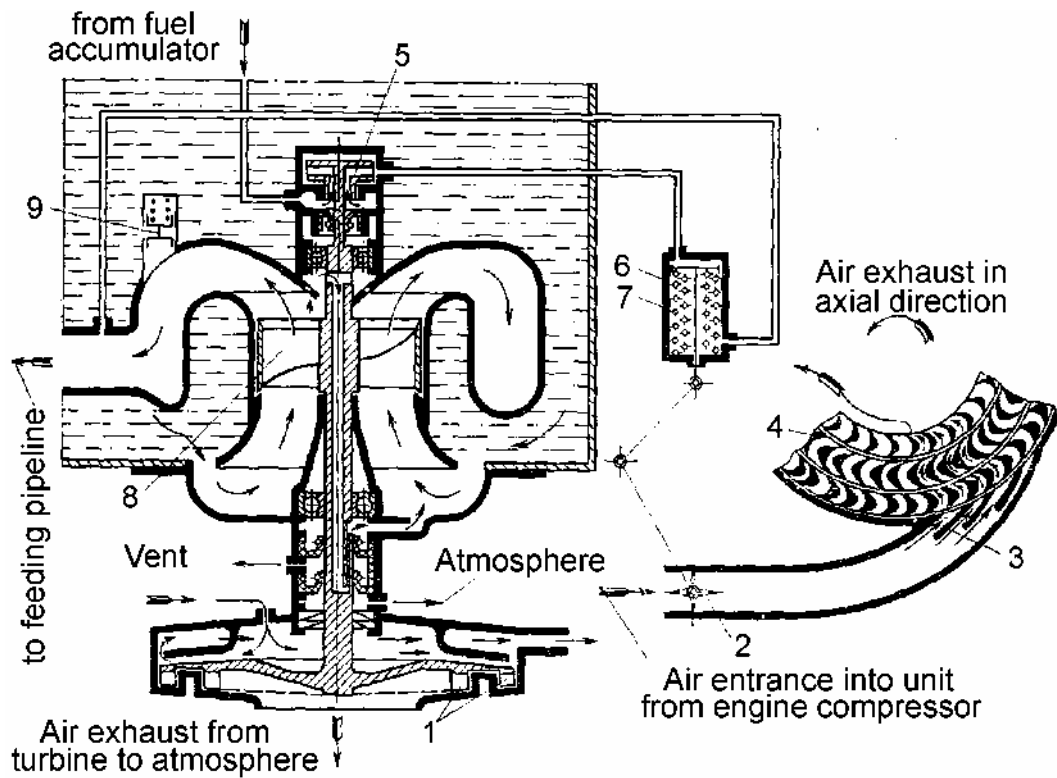


**Fig. 5.26. Electro-Powered Fuel Pump inside Fuel Tank**

Common disadvantages of the electric drive are high fire danger, necessity of cooling of pumps, necessity of pressure seal of the drive, small resource of seals and bearings, necessity of high-power generators availability.

The **pneumatic turbodrive** pump is shown in Fig. 5.27. In this case high pressure air supplied by turbojet compressor rotates the turbine, which in it's turn rotates the wheel of the pump. Such type of the drive has small mass and overall dimensions, high reliability. Disadvantages of pneumatic turbodrive are increase of turbojet specific fuel consumption while the air bleeding from compressor. Necessity of the starting electro-powered fuel pump availability, until the turbojet is idle throttle. There is a capability of the turbine racing by centrifugal forces during fuel outflow from the pump.

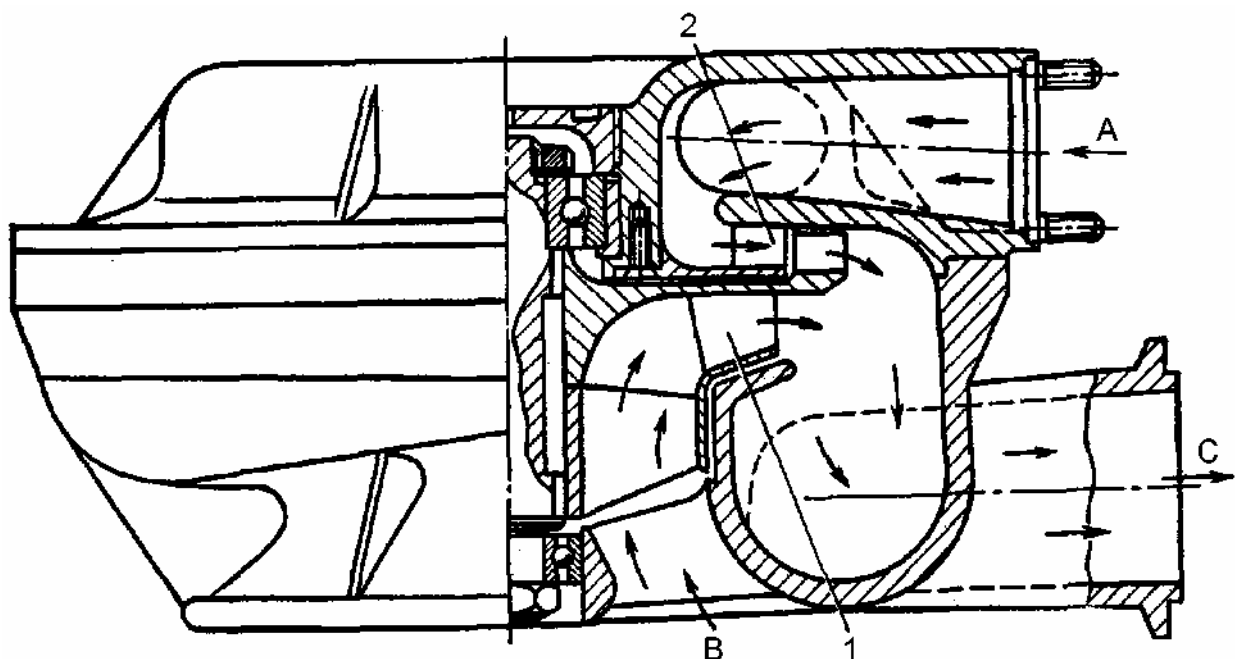




**Fig. 5.27. Pneumatic Turbdrive Pump:**

- 1 — Pneumatic Turbine; 2 — Shutter; 3 — Inlet Guide Vanes; 4 — Second Stage Pneumatic Turbine Guide Vanes; 5 — Pressure Gauge; 6 — Control Valve; 7 — Spring; 8 — Pump Working Wheel; 9 — Safety Valve; 10 — Filter; 11 — Air Outlet to Atmosphere; 12 — Air Inlet

The hydraulic turbodrives pump (Fig. 5.28) is the fuel turbine connected to the pump working wheel (impeller). Feeding of hydraulic turbodrives is carried out by the high-pressure fuel bled after FBP. The hydraulic turbodrives provide the minimal mass and overall dimensions of pump unit. It is not required any seals, as the fuel, rotating the turbine, is mixed up with the fuel, which is pumped up. The hydraulic turbodrives work successfully at the increased temperatures. Therefore it is widely used on supersonic aircraft. Starting and stopping the hydraulic turbodrives pumps does not impact to their resource. Disadvantages of the hydraulic turbodrives are big mass of high-pressure fuel pipelines, requirements of additional power of FBP for pumping this high-pressure fuel and requirements of starting electro-powered fuel pump availability.



**Fig. 5.28. Hydraulic Turbodrives Pump:**  
**1 — Pump Working Wheel; 2 — Turbine Guide Vanes; A — High-Pressure Fuel Inlet; B — Fuel Inlet from Tank; C — Fuel Outlet into Pipeline**

### 5.9.2. Fuel Tanks

Fuel tanks should meet the following requirements:

1. Strength, tightness;
2. Small mass;
3. Corrosion stability.

Usually the fuel tanks include:

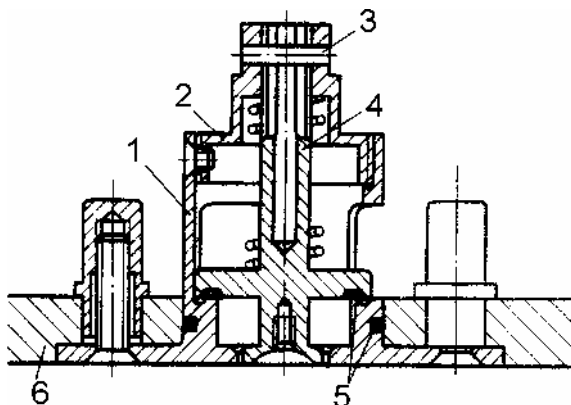
1. Shell;
2. Bulkheads for stiffness (diaphragm);
3. Attachment fittings;

- 4. Protective self-seal;
- 5. Fuel tank equipment:
  - 5.1. Filler neck;
  - 5.2. Outlet connection;
  - 5.3. Recuperators (in consumed fuel tank);
  - 5.4. Vent or pressurization line connection ;
  - 5.5. Drain valve (Fig. 5.29);
  - 5.6. Dumping connection;
  - 5.7. Fuel quantity gauges;
  - 5.8. Pumps;
  - 5.9. Electrical bonding.

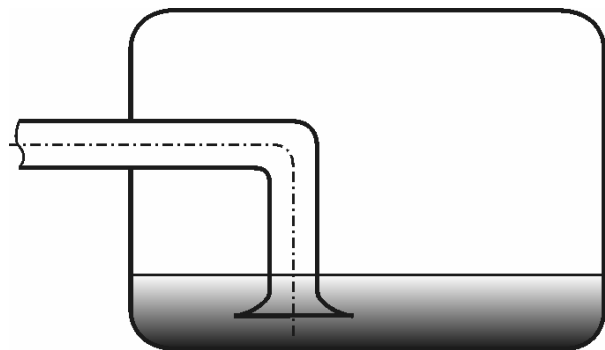
The **diaphragms** are intended to:

- maintain strength and stiffness of the fuel tanks;
- soften the hydraulic shocks during changes of aircraft attitude;
- quiet the surface of fuel, that reduces vaporation.

The **outlet connection** is usually arranged at some distance from the fuel tank bottom. It prevents hitting of condensate into the engine. If the fuel tank is flat, the outlet connection is placed sideways, Fig. 5.30.



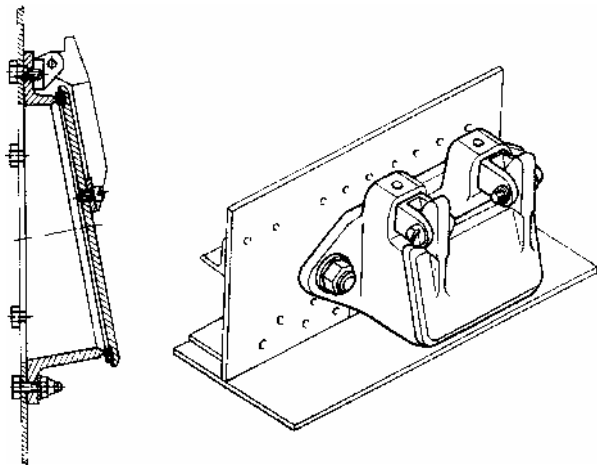
**Fig. 5.29. Drain Valve:**  
 1 — Casing; 2 — Cover; 3 — Rod;  
 4 — Valve; 5 — Sealing; 6 — Bottom  
 Wing Skin



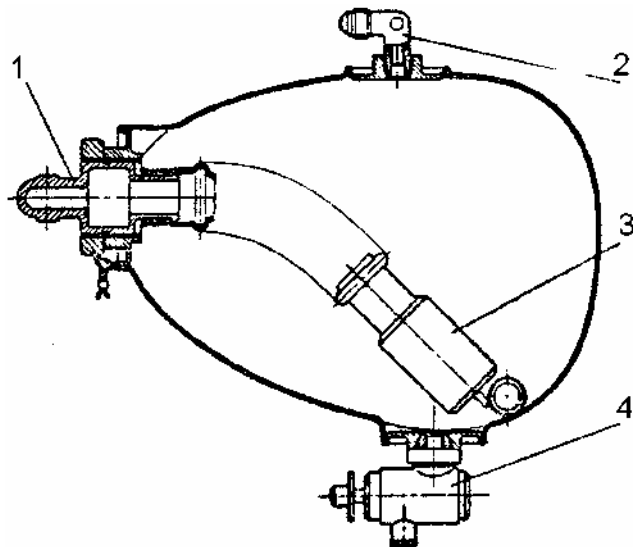
**Fig. 5.30. Versions of Arrangement  
 of Outlet Connection**

To provide a reliable feeding of the engine with fuel during changes of aircraft attitude, sometimes two outlet connections are installed or special recuperators are applied. In the simplest case, such recuperator is a consumed section of consumed fuel tank equipped with a set of check valves, Fig. 5.31. Such type of recuperator is used at non-maneuverable aircraft. At acrobatic maneuverable airplanes, whole consumed fuel tank can be made as a recuperator, Fig. 5.32.

At combat maneuverable aircraft, fuel pumps with recuperators (Fig. 5.33) and fuel accumulators (Fig. 5.34) are usually used.

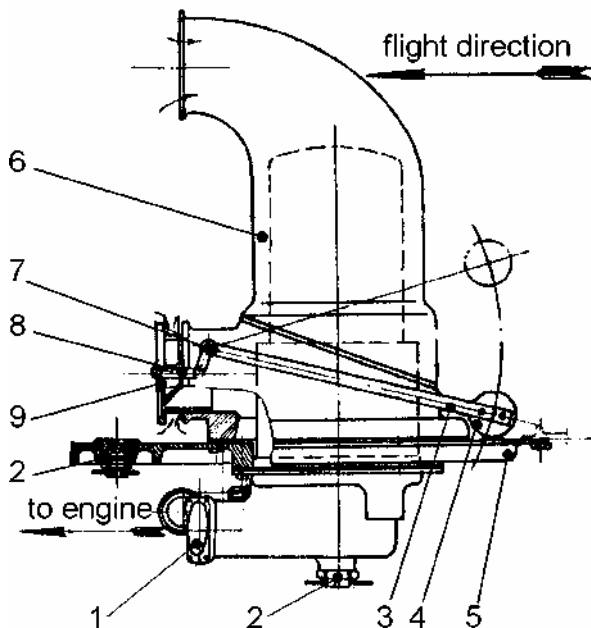


**Fig. 5.31. Check Valves in Consumed Section Ribs**

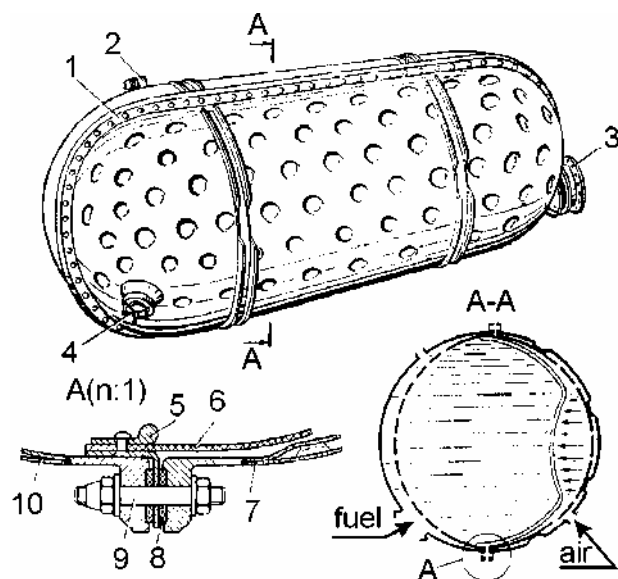


**Fig. 5.32. Consumed Fuel Tank of Acrobatic Airplane: 1 — Fuel Usage Pipeline; 2 — Venting; 3 — Fuel Inlet; 4 — Drain Valve**

Fuel tanks can be: internal (stiffen, flexible, integral) and external (under-wing, ventral, laid on and conformal). Internal fuel tanks can be placed into a fuselage or engine nacelles, a wing or a fin.



**Fig. 5.33. Fuel Pump with Recuperator:**  
 1 — BPA; 2 — Drain Valve;  
 3 — Valve Level; 4 — Weight;  
 5 — Base Plate; 6 — Valve Casing;  
 7 — Weight Axis of Rotating;  
 8 — Carrier; 9 — Valve



**Fig. 5.34. Fuel Accumulator:**  
 1 — Flange; 2 — Pipeline Returning Fuel to Pump; 3 — Fuel Pipeline;  
 4 — Pressurization Pipeline;  
 5 — Shape; 6 — Diaphragm; 7 — Air Shell; 8 — Sealing; 9 — Bolt;  
 10 — Fuel Shell

Fuselage fuel tanks:

[+] have the shape convenient for fuel use;

[+] have smaller mass, in comparison with wing fuel tanks (because of strength);

[-] take a payload volume in the fuselage.

Wing fuel tanks:

[+] unload the wing in flight;

[+] free payload volume in the fuselage from fuel;

[-] complicate fuel use (because the fuel tank is flat);

[-] have the bigger surface area subjected to damage.

**Rigid fuel tanks:**

[+] have the stiffness independent of the airframe;

[-] require big cut-outs to install and remove them from airplanes;

[-] do not completely use the allowable volume of the airframe;

[-] can be damaged by vibrations;

[-] have big mass.

Now they are applied on light airplanes with piston engines.

The stiffen fuel tanks are made of sheets of aluminium alloys such as AMr and AM $\zeta$ . These alloys allow deep draw-forming, embossings, they are corrosion-resistant and well welded. Recently composite materials are also used for manufacturing stiffen fuel tanks.

By its design, the cover of rigid fuel tanks usually consists of a shell and two welded bottoms (Fig. 5.35). Welding provides tight connection. The fuel tank is welded from outside on special flanging separated by embossings. It eliminates hitting of a flux and a fire scale inside of the fuel tank and reduces residual stresses after welding.

Bulkheads are usually produced of duralumin. As duralumin is bad welded, bulkheads are attached to the shell with rivets. The head of rivets sometimes are welded for tightness.

Installation of stiffen fuel tank is usually carried out with special support assemblies, to which fuel tank is held down with attaching straps (Fig. 5.36).

After manufacturing, all fuel tanks are tested for tightness and vibration strength.

On military airplanes, the fuel tanks are usually covered with a self-seal which is capable to tighten shell-holes in the fuel tank.

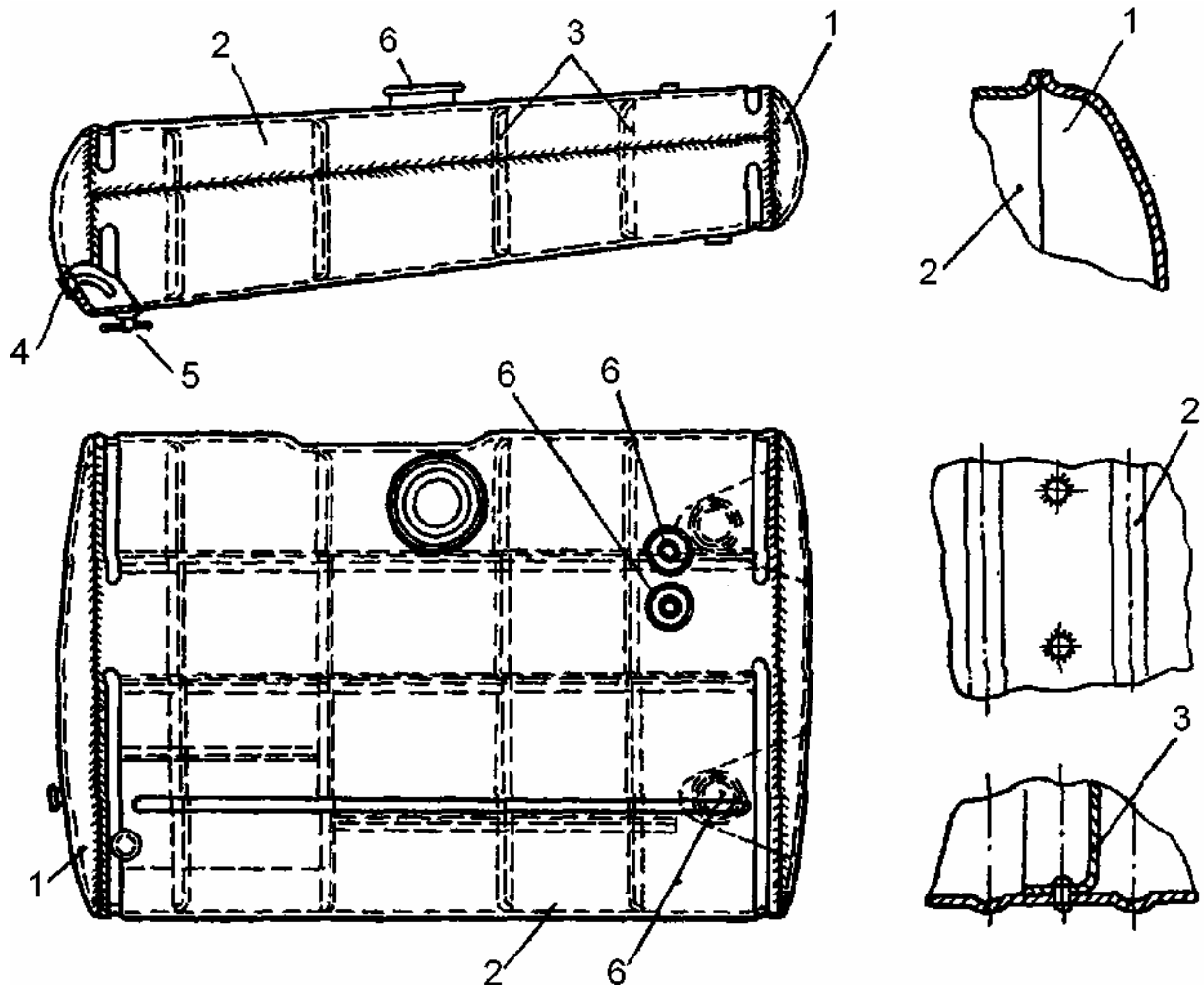
**Flexible fuel tanks** (Fig. 5.37):

[+] are installed and removed through cut-outs of small sizes;

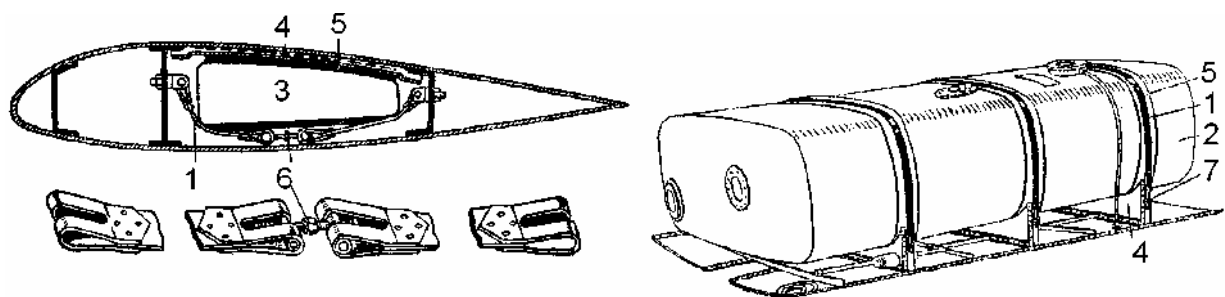
[+] are stable to vibrations;

[+] have good survivability;

[+] have mass smaller, than rigid ones;



**Fig. 5.35. Design of Rigid Fuel Tanks:**  
**1 — Bottom; 2 — Shell; 3 — Bulkhead; 4 — Fuel Outlet; 5 — Drain Valve;**  
**6 — Fuel Tank Equipment Pipeline Connections**

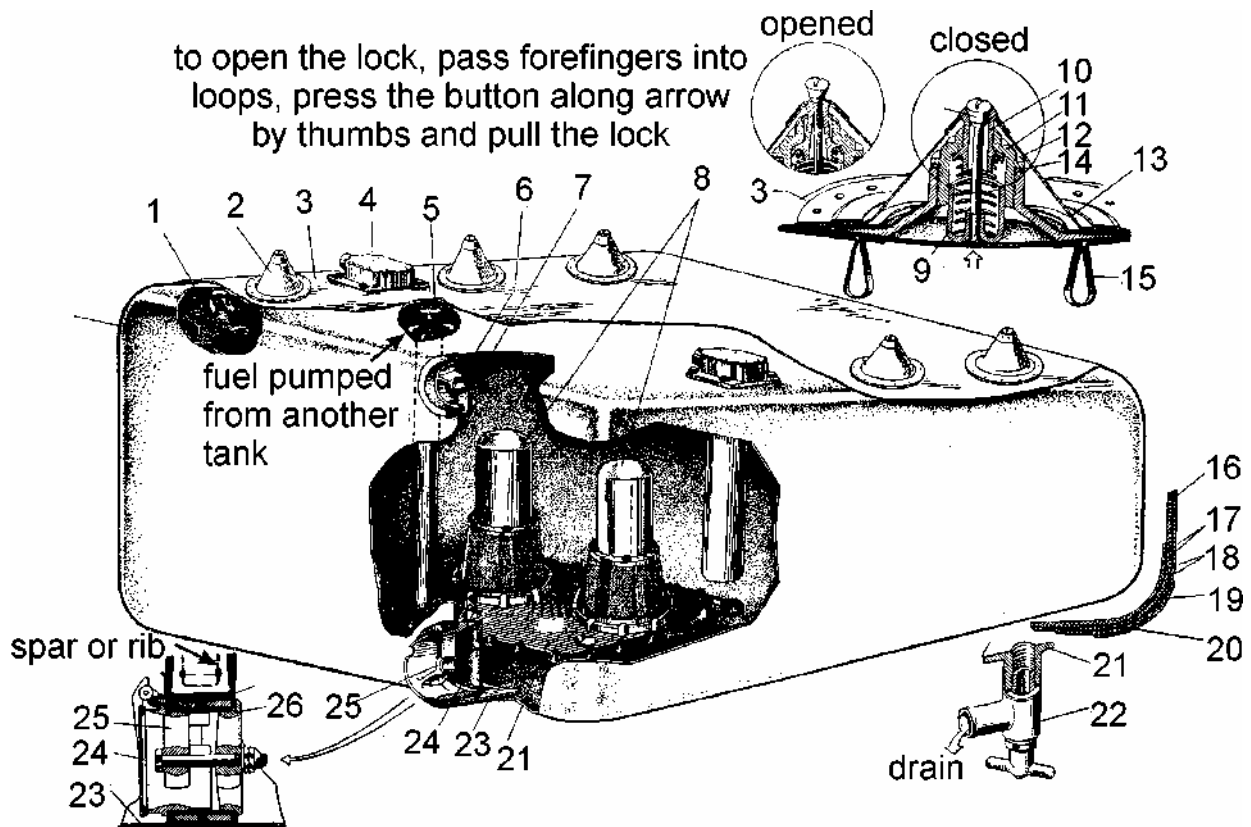


**Fig. 5.36. Installation of Rigid Fuel Tank with Attaching Straps:**  
**1 — Attaching Strap; 2 — Tank; 3 — Bulkheads; 4 — Support Assemblies;**  
**5 — Felt Plate; 6 — Tender; 7 — Bolt, Attaching Strap with Rib Node**

[–] lose elasticity at low and high temperatures, hence appearance of cracks is possible;

[–] have mass greater than integral fuel tanks;

[–] require presence of container inside the airframe;



**Fig. 5.37. Flexible Fuel Tank:**

- 1 — Bushing of Vent Connection; 2 — Lock Attaching Tank to Bracket; 3 — Container; 4 — Sensor of Fuel Quantity Gauge; 5 — Fuel Transfer Pipeline from Other Tank; 6 — Vent Hole; 7 — Tank; 8 — Booster Pumps; 9 — Knob; 10 — Retaining Screw; 11 — Catch; 12 — Casing of Lock; 13 — Metal Cone; 14 — Locking Washer; 15 — Loop; 16 — External Layer of Tank; 17 — Internal Layer of Tank; 18 — Glue Layer; 19 — Internal Strap; 20 — External Strap; 21 — Plate Attaching Pumps; 22 — Drain Valve; 23 — Hold-Down; 24 — Check Valve; 25, 26 — Bushes

[–] are subject to dissolution in fuel, hence periodic flushing is required.

The cover of flexible fuel tanks usually consists of several layers. Inside there is a layer of fuel resistant rubber. The following layer is usually the self-seal, tightening shell-holes. Then there is a layer of vulcanized rubber. The external layer is the protective cord (strong rubber cloth).

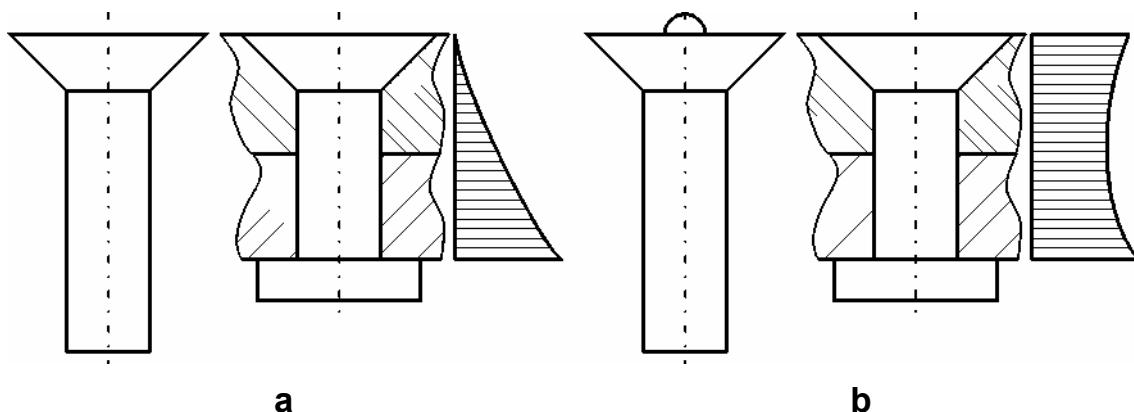
The fuel tank equipment is vulcanized to walls of the fuel tank, and besides, is fastened to the container.

**Integral fuel tanks** represent the closed pressurized section of an airframe. Integral fuel tanks:

- [+] use the allowable volume completely;
- [+] have minimal mass;
- [+] do not require big cut-outs for maintenance;
- [–] are subjected to aerodynamic, vibrational, thermal effects;

- [-] require special measures to provide full fuel use;
- [-] require tightening. Tightening process is complicated, detrimental to health, not always reliable.

Now the fasteners with the radial interference, ensuring tightness without hermetic, obtained a wide application. For example, when riveting by usual rivets, radial interference appears in snap rivet head zone only (Fig. 5.38, a). While riveting by rivets with compensator, radial interference generates practically along whole pack thickness (Fig. 5.38, b). The very radial interference gives tightness of riveted joint. Such fasteners reduce labouriousness, reduce assemblies production time, allow to reduce using of harmful hermetic in manufacture, decrease mass.



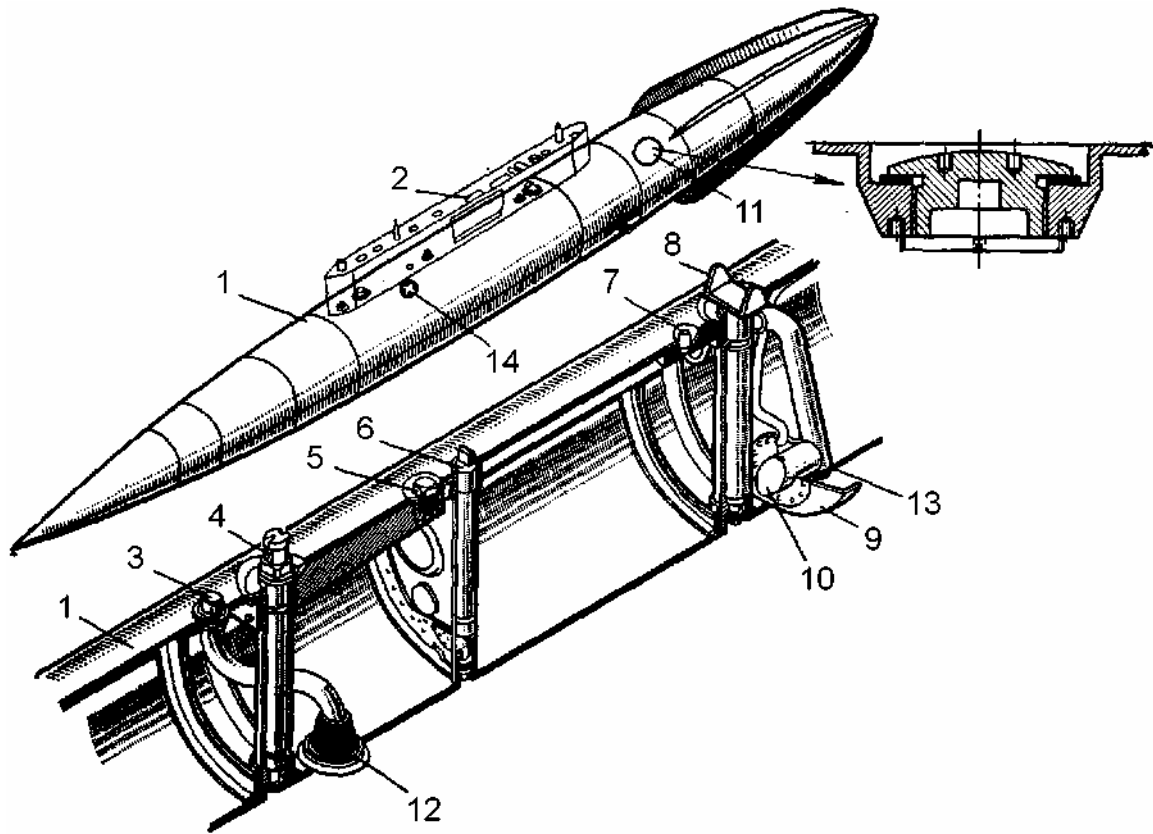
**Fig. 5.38. Using Fasteners with Radial Interference**

**External (under-wing, ventral, laid on and conformal) fuel tanks** represent rigid fuel tanks, extruded directly to a ram airflow. Just like the integral fuel tank, it is subject to aerodynamic, vibrational, thermal effects. Besides, the shape of such fuel tanks should provide the minimal external drag at the specified volume. The shape of fuel tanks should not cause stalls, disturbance of stability and controllability of the aircraft.

On manoeuvrable airplanes external fuel tanks are usually designed droppable (Fig. 5.39). It allows to get rid fast from superfluous mass and to improve a manoeuvrability of the airplane if necessary. As a result there are no pumps inside them. As the fuel pump is very expensive. Therefore fuel feeding from external fuel tanks is usually provided by forcing out. Thus the tanks pressurization with overpressure increases strength of fuel tanks.

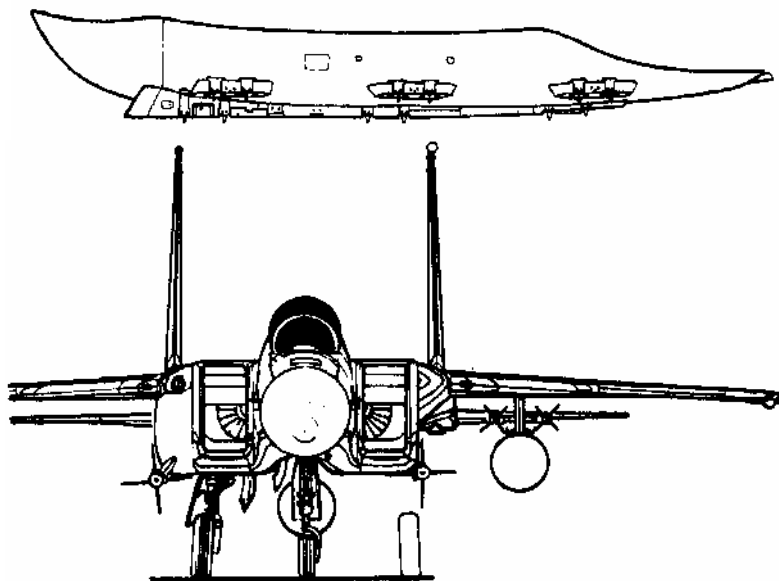
Laid on fuel tanks differ from under-wing or ventral only in arrangement. And they are usually not droppable. Conformal fuel tanks represent external fuel tanks, attached to the airplane without clearances. Thus total aerodynamic drag of the system (the airplane plus the fuel tank) can even be decreased. The external weapon-pylon bases can be attached to such conformal fuel tank, Fig. 5.40.





**Fig. 5.39. Ventral Fuel Tank:**

**1 — Ventral Tank; 2 — Pylon for Attaching Tank; 3 — Fuel Outlet Connection; 4 — Forward Stop; 5 — Bush of Pyropusher; 6 — Eyebolt; 7 — Pressurization Connection; 8 — Rear Stop; 9 — Pressurized Bulkhead; 10 — Check Valve; 11 — Filler Neck; 12 — Drain Valve of Forward Section with Filter; 13 — Drain Valve of Rear Section; 14 — Filler for Fuel Pumpdown on Ground**



**Fig. 5.40. Conformal Fuel Tank of F-15 Airplane**

## **Theme 6. LUBRICATION SYSTEMS OF POWER PLANTS**

### **6.1. Purpose of Lubrication Systems and Requirements**

A **lubrication system** represents a set of units intended for lubrication of the engine, heat rejection from engine units, and also the removal of solid particles, which are formed between surfaces subjected to friction, in all aircraft operating conditions.

Even short-term breaks in oil feeding result in the engine overheating, destruction of its bearings, wedging of pistons, or a rotor, breakage of connecting rods and, as a result, the engine failure.

In power plants, oil is also used as hydraulic liquid in various devices (such as a propeller pitch control mechanism, fuel control units, rpm governor, etc.).

The lubrication system consists of two sections: external and internal. The external section is a component of the aircraft power plant. The internal section is a component of the engine. In our course we shall consider only the external section of lubrication system.

#### **Requirements to lubrication systems**

1. Reliable oil supply to the engine at the temperature and pressure due to specifications, in any aircraft operation conditions.
2. Minimum power consumption during heated oil cooling process.
3. Reliable oil filtering from mechanical impurities and gases.
4. Fast oil heating, to reduce the heating-up time of the engine.
5. Preventing oil ejection through venting and overflow of the engine at all engine settings.
6. Preventing oil overflows from the tank into the inoperative engine.
7. Minimal mass of the system.
8. The sufficient strength, vibration strength, leak resistance, and minimal hydraulic pressure losses of the system elements.
9. Fire safety.
10. Maintainability (convenient access to the lubrication system for maintenance and repair).
11. On multi-engine aircraft, each engine should have an independent lubrication system.
12. Pipelines and units of lubrication systems should be of brown color.

### **6.2. Aviation Oils**

For the reliable and continuous operation of a lubrication system, oil

properties are very important. Operating conditions for oils in engines are complicated and various, therefore oil specifications are characterized by strict requirements to their physical and chemical properties.

**Viscosity** is one of the major properties of oil. If oil viscosity is insufficient, the liquid layer of the oil cannot stay in clearances and is extruded at heavy loadings. As a result, dry friction emerges, which increases the wear of parts.

If oil viscosity is too high, then:

- friction forces increase, which causes power losses;
- oil does not get to units with small clearances;
- increased pressure in lubrication system is required for pumping oil;
- engine start-up is complicated.

The value of oil viscosity is measured by the factors of viscosity. Just as with fuel, there are two factors of oil viscosity: kinematic and dynamic ones. The factor of kinematic viscosity of aviation oils is in the range:  $\nu = 2.5...25 \text{ mm}^2/\text{s}$ .

The viscosity of oil drops with the increase of temperature. In the past, most aviation oils were made from petroleum, by acid (MK) or selective (MC) clearing. The viscosity of such oils drops sharply with increase in temperature. High temperature in lubrication system results in the active vaporation and oxidation of these oils. The resulting deterioration of their physical and chemical properties, worsens the lubrication system operation. Now, in heat-stressed engines, synthetic oils are widely used. They are formed on the basis of esters. The viscosity-temperature relation for such oils is weaker. Besides, these oils have a broad range of operation temperatures.

The **density** of aviation oils makes  $\rho = 850...950 \text{ kg/m}^3$ .

The **thermal-oxidative durability of oil** is characterized by maximum temperature at which the oil can be used. When this temperature is exceeded oil is coked, i.e. oil sediment appears, and a film is formed on the greased parts. The type of oil to be used is determined by the type of the engine.

## 6.3. General Characteristics of Lubrication Systems

### 6.3.1. Lubrication Systems for Piston Engines

Lubrication systems for piston engines are the most complicated and loaded ones. Piston engines have a big friction area. There is high pressure ( $\geq 25 \text{ MPa}$ ) on these surfaces, which increases the heat transfer into oil ( $Q = 3400...6800 \text{ J}/(\text{kw}\cdot\text{min})$ ). Hence, big oil flow rate through the engine ( $2.7...5.4 \text{ dm}^3/(\text{hour}\cdot\text{kw})$ ) is required.

Second feature of lubrication system of piston engines is that, oil directly contacts with a combustion zone, that causes high oil consumption ( $C_p = 8...16 \text{ g}/(\text{hour}\cdot\text{kW})$ ).

To provide lubrication system operation in these conditions, oils with viscosity 20...25 sSt at 100°C are required, for example MC-20, MK-22 (the number in the marking designates viscosity in sSt at 100°C). In winter conditions less viscous oils are applied.

The recommended oil temperature makes 65...75°C at the input connection of an engine, and 105...125°C at the output connection. Delivery oil pressure is in the range of 0.5...0.8 MPa.

### **6.3.2. Lubrication Systems of (Bypass) Turbojets**

The principal type of friction in these engines is rolling friction, therefore, the power consumption required to overcome the friction forces and the amount of the transferring heat are insignificant (85...170 J/(daN·min)). Oil flow through the engine makes 3...5 dm<sup>3</sup>/min per bearing. The oil consumption basically comes from the losses through the breather system and makes  $\approx 1 \text{ g}/(\text{daN}\cdot\text{hour})$ .

Under these conditions oils with low viscosity (MK-6, MK-8, and also synthetic oils) are applied.

The allowable input oil temperature makes 25...80°C, and the output 100...130°C. The oil delivery pressure for turbojets is in the range 0.1...0.4 MPa.

### **6.3.3. Lubrication Systems of Turboprops**

Due to the complexity and the loadings of the turboprops lubrication systems, they are intermediate between the lubrication systems of piston engines and turbojets. The difference between these systems consists in employing the propeller gear to transmit the high power. The pinions of this propeller gear require, thus, flood lubrication.

The turboprop heat-to-oil transfer (680...1400 J/(kW·min)) is higher, than turbojets', but is lower than piston engines'. So, the oil flow (0.65...0.98 dm<sup>3</sup>/(kW·hour)) is intermediate too.

It is not desirable to apply here oils with high viscosity, since the rate of the propeller blade angle change drops, and engine start-up is complicated. A mixture of oils with high and low viscosity (e. g. 75 % of MC-20 and 25 % of MK-8) is usually used to lubricate turboprops.

## **6.4. Lubrication System Schemes. Methods of Separation Air from Oil**

There are two types of lubrication systems: closed (circulating) and

open, the latter is applied to the products of single utilization. The closed type systems are used in aviation.

### Direct Scheme of Lubrication System

The simplest scheme of lubrication system is the **direct scheme** (Fig. 6.1). In this case oil circulates through the following path: oil tank – engine – cooler – oil tank.

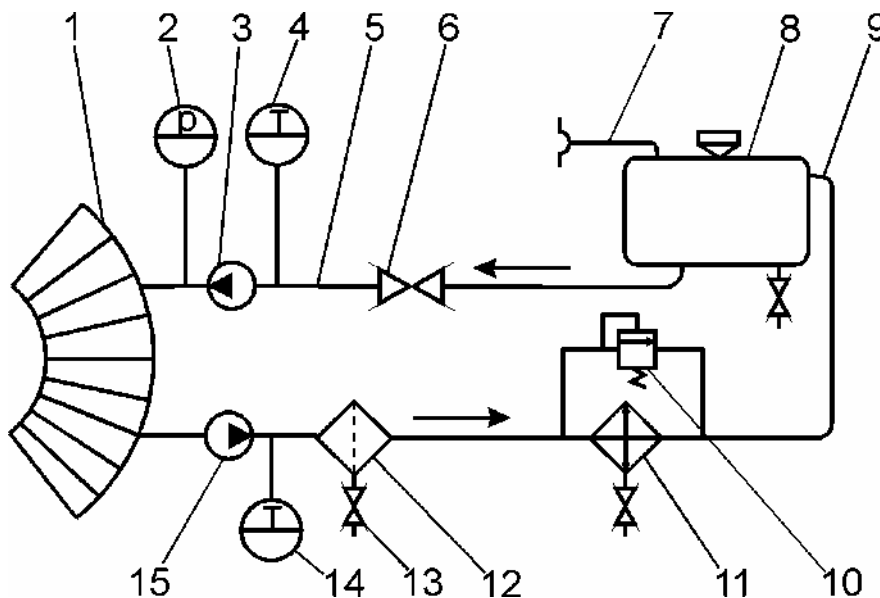


Fig. 6.1. Direct Scheme of Lubrication System

**1 — Engine; 2 — Manometer; 3 — Pressure Pump; 4 — Thermometer of Oil at Engine Inlet; 5 — Suction Pipeline; 6 — Shut-Off Valve; 7 — Venting; 8 — Oil Tank; 9 — Scavenge Pipeline; 10 — By-Pass Valve; 11 — Cooler; 12 — Filter; 13 — Drain Valve; 14 — Thermometer of Oil at Engine Outlet; 15 — Scavenge Pump**

The section of lubrication system from oil pressure pump to scavenge pump (including them) is an internal pipeline of the engine 1. The external lubrication system includes: oil tank 8, cooler 11, suction (tank – engine) 5 and scavenge (engine – cooler – tank) 9, pipeline sections, venting pipeline 7, drain pipes and valves 13, instruments for measurement of pressure 2 and temperature 4, 14 of oil and oil meter.

The suction pipeline should be straight, short and have a large diameter. To improve operating conditions of the pressure pump 3, it is possible to install the oil tank above the pump.

The shut-off valve 6 is sometimes installed in the suction section, to prevent the oil overflow from the tank into the inoperative engine through the switched off pump. In this case it is necessary to interconnect this valve with the engine starting system.

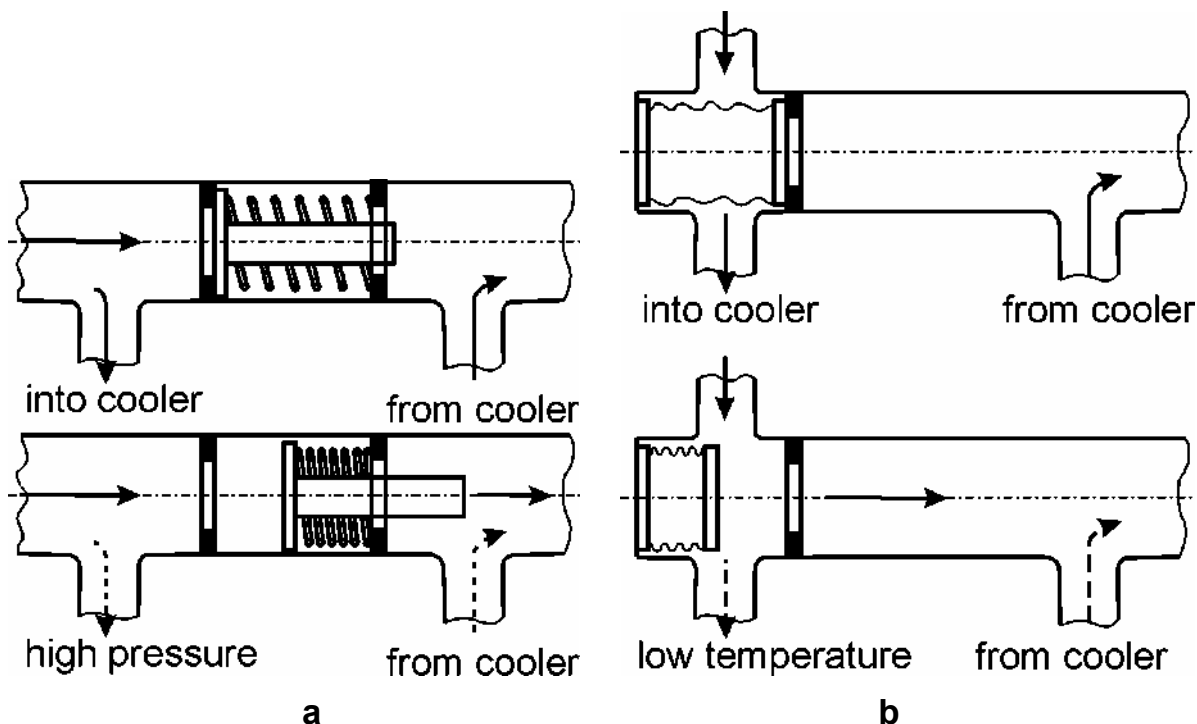
For protection of lubrication system against chips, calx, dust and flux coming from the oil tank, it would be expedient to install the filter in the

«tank – engine» pipeline section. But thus there will be a high pressure losses in the suction pipeline. Thus altitude performance of lubrication system will decrease. Therefore filters 12 are installed in the scavenge section. If metal chips emerge on the filter surface, that is the first indicator of violation of normal engine operation. Additional filter grids are installed in the filler necks and in the tank outlet connections.

To prevent the oil and its emulsion from accumulation in the engine casing, the scavenge pumps 15 should have 2...3 times more volume flow, than pressure pumps have. As the pumps except for the emulsion, pump out big quantity of air, it results in additional saturation of oil by air.

The cooler 11 is equipped with the by-pass valve 10, intended to prevent the cooler core rupture at low oil temperatures and provide the fast oil heating-up. As at the open valve, oil circulates, omitting the core. When oil is heated and the pressure drops, the valve closes and brings the cooler into operation.

The spring-loaded valves (Fig. 6.2, a) are the simplest in design and the most reliable. Their essential disadvantages consist in the capability of passing of cold oil through the cooler at the low engine power. Thus oil flow is small and resistance of cooler is small too. It increases heating-up time of oil, and in flight (for example, at gliding) causes its super-cooling. It can cause the oil solidification and breaking of oil flow.



**Fig. 6.2. Schemes of By-pass Valves**

For this reason, the thermostatic valves (Fig. 6.2, b) are preferable. Such valves react to the oil temperature variation due to the silphon filled

with low-boiling fluid. With heating of oil, the silphon extends, and the valve closes. Thus oil will flow only through cooler core.

It is necessary to note the difference between the purpose of the fuel tank and oil tank venting systems. In fuel tanks, the air feed is required to provide specified overpressure in the fuel tank. In oil tanks, the air discharge is required, because of the air comes in plenty from the engine together with air-oil emulsion.

It is recommended to direct venting connections of oil tank downwards in order to prevent the oil accumulation, solidification in them and, as a result, breaking of the oil tank. For the same reason, venting should be directed to a warm place, such as a cooler tunnel situated behind the oil cooler.

Venting of lubrication system can be open and closed (through a breather system).

When designing the lubrication system, it is necessary to provide full, fast and reliable drain of oil from the oil tank, cooler, filter sediment trap and pipelines. Convenient access to drain valves should be provided. It is necessary to prevent hot oil from hitting onto hands of ground maintenance personnel, electrical wiring, exhaust pipes, and wheel tyres.

Operation of lubrication system is inspected by oil pressure and temperature at engine inlet and by oil meter.

### **Air-from-Oil Separation**

When passing through the engine, the oil is saturated with air, gases and, in piston engines, with fuel vapors. During lubrication of various units, bubbles of air are in addition shattered. As a result, the air-oil emulsion is formed, this emulsion is pumped out by scavenge pumps. Thus there is additional oil saturation by air. Behind the cooler, pressure of oil drops, and additional air educes from it. Thus oil, in which there are bubbles of various sizes, gets in the oil tank. Size of these bubbles changes from molecular up to 10...20 mm in diameter.

Large bubbles rapidly float and separate from oil. The less sizes of bubbles, the less its speed of float, and it is more difficult to separate them from oil.

The floating speed is directly proportional to the squared float diameter and inversely proportional to the oil viscosity. Thus, the more time the oil in the tank is available, the less air comes to suction pipeline. The higher the oil temperature and the lower the oil viscosity becomes, the less time it takes for the bubble to get the surface

$$V_y \sim D^2 / \mu(t).$$

The following measures are assumed to reduce the contents of air in oil:

1. Reducing hydraulic pressure losses in scavenge pipeline of oil.
2. Minimally required difference between oil flows of scavenge and pressure pumps should be chosen.
3. Installation of special de-aerators in the oil tank. They use a principle of coagulation — enlargement of gas particles in two-phase medium.
4. Centrifugal de-aerators have the wide application now. They provide the complete air-from-oil separation, and better altitude performance of lubrication system; however they require a lot of power.

Let's note the disadvantages of the direct scheme of lubrication system:

[–] High back pressure, owing to presence of the cooler in a scavenge pipeline;

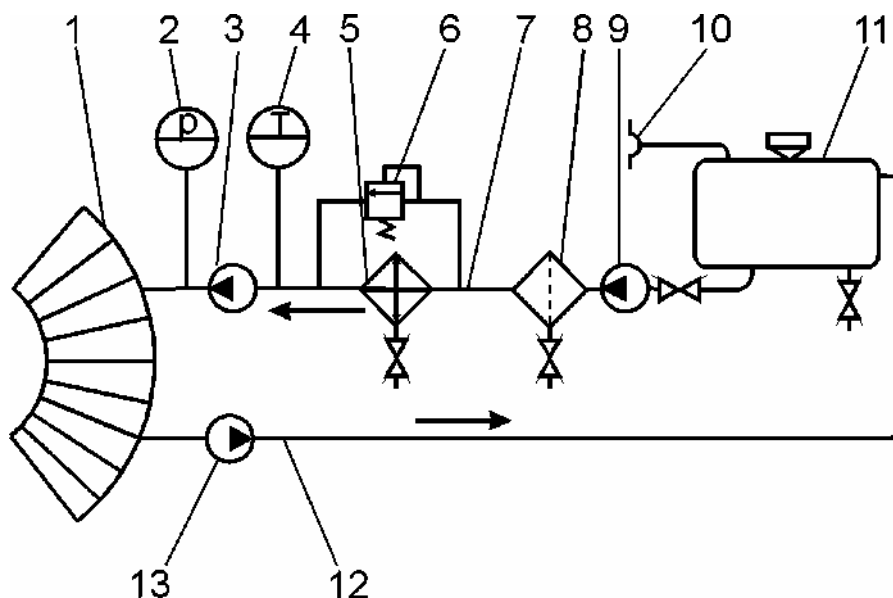
[–] Low temperature in the oil tank; causing the high oil viscosity and the low speed of the air bubbles floating.

[–] Altitude performance of the direct scheme lubrication system makes 7...8 km.

Owing to these disadvantages, other schemes of lubrication system are widely used.

### Reverse Scheme of Lubrication System

The oil circulates according to the scheme (Fig. 6.3): oil tank 11 – cooler 5 – engine 1 – oil tank.



**Fig. 6.3. Reverse Scheme of Lubrication System**

- 1 — Engine; 2 — Manometer; 3 — Pressure Pump; 4 — Thermometer of Oil at Engine Inlet; 5 — Cooler; 6 — By-Pass Valve; 7 — Suction Pipeline; 8 — Filter; 9 — Transfer Pump; 10 — Venting; 11 — Oil Tank; 12 — Scavenge Pipeline; 13 — Scavenge Pump**



Hot oil, without cooling in the cooler, gets directly to the oil tank. The oil viscosity considerably decreases. The speed of air separation proportionally increases. Besides, through absents of cooler, the air bubbles are not shattered and keep the relatively big diameter; that also promotes fast air separation.

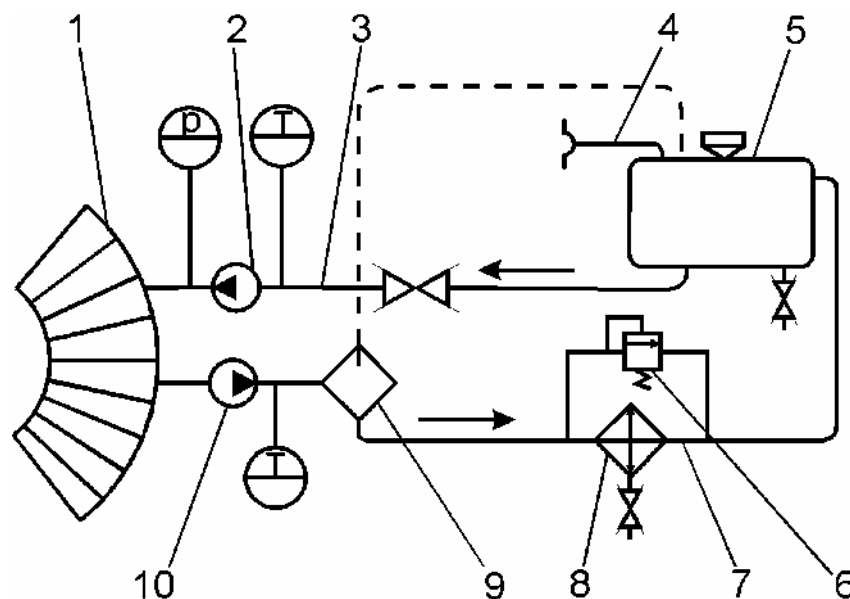
When the oil cleared from air flows through the cooler, its heat rejection increases. Since the oil from the cooler passes directly to the pressure pump, the system reacts faster to the regulating of cooling.

But the pressure pump cannot provide the oil pumping through the cooler. Therefore, the additional pump 9 in the segment «oil tank – cooler – engine» should be installed in the reverse scheme of lubrication system. It is a disadvantage of the reverse scheme.

The altitude performance of the reverse scheme lubrication system makes 10...12 km.

### Single-Circuit Scheme with Centrifugal De-aerator

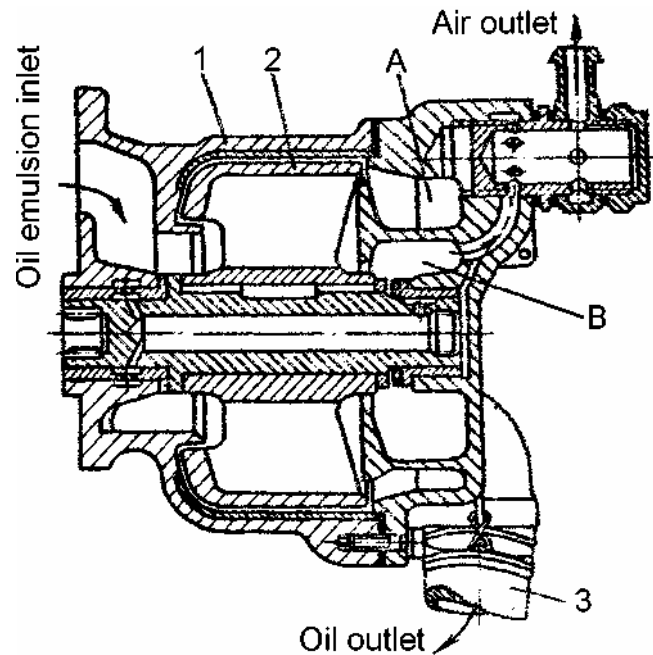
The altitude performance of the direct scheme can be increased by the installation of the centrifugal de-aerator 9 in the oil scavenge pipeline (Fig. 6.4).



**Fig. 6.4. Scheme of Lubrication System with Centrifugal De-aerator**  
 1 — Engine; 2 — Pressure Pump; 3 — Suction Pipeline; 4 — Venting;  
 5 — Oil Tank; 6 — By-Pass Valve; 7 — Scavenge Pipeline; 8 — Cooler;  
 9 — Centrifugal De-Aerator; 10 — Scavenge Pump

The centrifugal de-aerator (Fig. 6.5) is installed before the cooler, thus making the process of air separation far easier (as the oil is hot). Besides heat rejection in the cooler also increases, as oil is cleared from air.

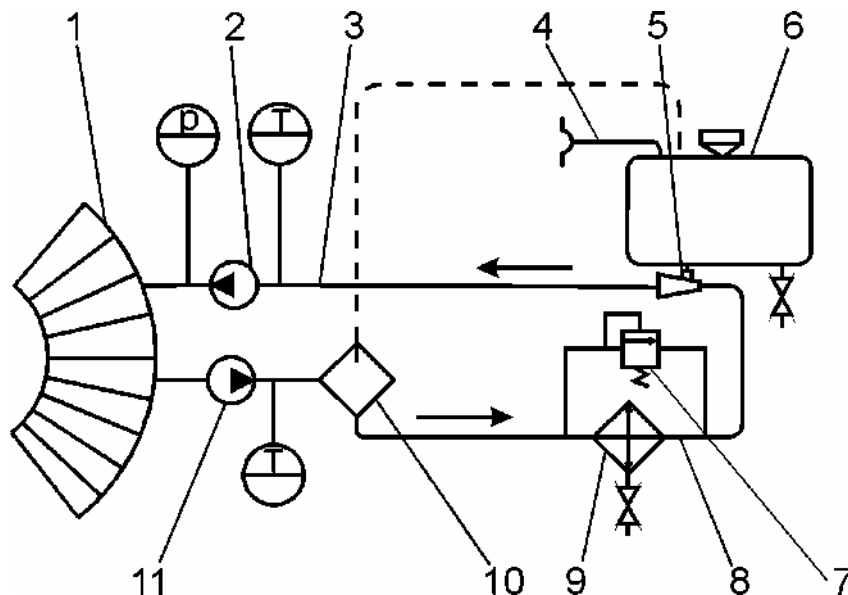
This scheme is the most used on up-to-date turbojet airplanes.



**Fig. 6.5. Centrifugal De-aerator**  
**1 — Casing; 2 — Impeller; 3 — Oil Exit Pipeline Connection;**  
**A and B — Annular Cavities**

### Shorted Schemes

As the centrifugal de-aerator is capable to clear the oil from air, there is no need for oil circulation through the oil tank. This fact has resulted in the development of the shorted lubrication system (Fig. 6.6, 6.7).



**Fig. 6.6. Shorted Scheme Lubrication System with Ejector**  
**1 — Engine; 2 — Pressure Pump; 3 — Suction Pipeline; 4 — Venting;**  
**5 — Ejector; 6 — Oil Tank; 7 — By-Pass Valve; 8 — Scavenge Pipeline;**  
**9 — Cooler; 10 — Centrifugal De-Aerator; 11 — Scavenge Pump**

According to this scheme the oil circulates within the contour «engine – scavenge pump 11 – de-aerator 10 – cooler 9 – pressure pump 2 – engine». The ejector 5 serves for compensation of the oil consumption only.

The scheme provides.

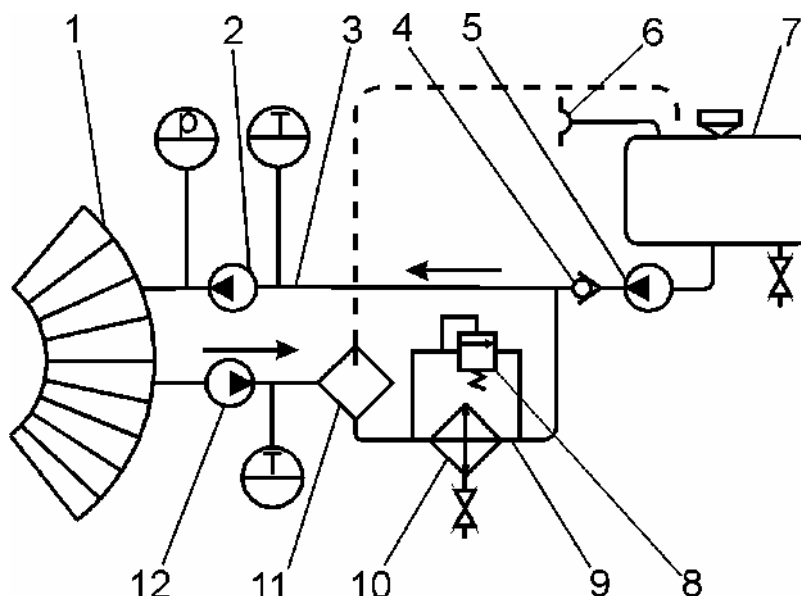
[+] Fast oil preheating (as only the oil circulating within the contour is preheated).

[+] Good altitude performance (due to the capability of creation of backup in the main pump inlet, Fig. 6.7).

[+] Small capacity (hence mass) of the oil tank. The oil tank contains only the store of oil consumed in flight.

[-] As the tank is not used for cooling oil, the cooling surface of the cooler (hence mass) should be increased.

The shorted schemes of lubrication system are used on short- and medium range turboprop airplanes (An-10, An-24).



**Fig. 6.7. Shorted Scheme Lubrication System with Pump**

**1 — Engine; 2 — Pressure Pump; 3 — Suction Pipeline; 4 — Check Valve;  
5 — Transfer Pump; 6 — Venting; 7 — Oil Tank; 8 — By-Pass Valve;  
9 — Scavenge Pipeline; 10 — Cooler; 11 — Centrifugal De-Aerator;  
12 — Scavenge Pump**

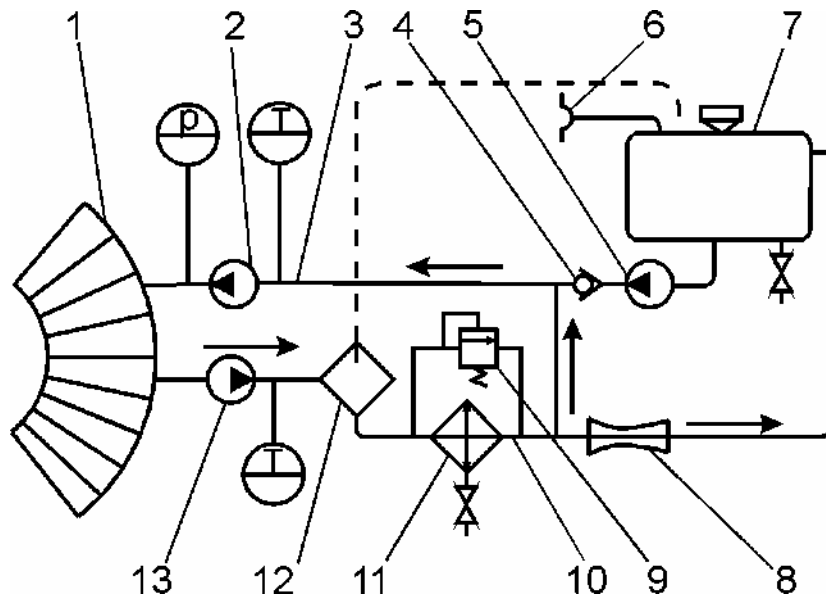
### **Double-Circuit Scheme of Lubrication System**

Double-circuit scheme of lubrication system (Fig. 6.8) has the boost pump 5 and two contours of oil flow: main and additional.

The main contour includes: pressure pump 2 – engine – scavenge pump 13 – centrifugal de-aerator 12 – cooler 11 – pressure pump. Approximately 90% of oil circulates in this contour. The residual oil passes in

the oil tank for preheating through the additional circuit (cooler – oil tank 7 – boost pump). The diameter of bleed 8 in the additional circuit is selected by condition of passing of the oil optimal quantity (approximately 10%) through it.

The double-circuit scheme has the same advantages as the shorted scheme. It is applied in power plants of long-range airplanes, such as Il-62.



**Fig. 6.8. Double-Circuit Scheme Lubrication System**

1 — Engine; 2 — Pressure Pump; 3 — Suction Pipeline; 4 — Check Valve; 5 — Transfer Pump; 6 — Venting; 7 — Oil Tank; 8 — Bleed; 9 — By-Pass Valve; 10 — Scavenge Pipeline; 11 — Cooler; 12 — Centrifugal De-Aerator; 13 — Scavenge Pump

## 6.5. Calculation of Lubrication System

### 6.5.1. Calculation of Required Capacity of Oil Tank

The oil tank capacity is calculated by condition of filling up the tank with maximum amount of oil and providing the reserve volume for the oil expansion and foam.

The oil tank capacity is calculated as a sum of the following volumes:

1) The volume  $V_1$  required for consumed oil capacity

$$V_1 = K \frac{C_M \cdot N_e \cdot \tau}{\rho_M},$$

where  $C_M$  — specific oil consumption;  $N_e$  — engine power;  $\tau$  — maximum possible engine operation time without oil servicing;  $\rho_M$  — oil den-

sity;  $K$  — safety factor, which takes into account random oil emission, incomplete filling up, etc  $K \approx 1.15...1.2$ .

2) The volume  $V_2$  that is the amount of oil left in the oil tank at the end of flight. This volume provides the acceleration of the system (to provide normal oil temperature at advance the throttle control lever, and in case of the delay in opening the cooler shutters) the lubrication in case of oil outflow in the casing, etc.

$$V_2 \approx (2...5) \frac{N_e}{100} + (5...10)[L].$$

Value  $V_2$  is usually specified by the engine manufacturer and is specified in maintenance manual.

3) The volume  $V_3$  required for a lubrication system filling up.

Thus, the oil capacity, which is filled in the system is determined as  $V_{LS} = V_1 + V_2 + V_3$ .

The total oil tank capacity is  $V_T = K_1 \cdot V_{LS}$ , where  $K_1$  — the tank capacity safety factor at the oil-into-foam expansion  $K_1 = 1.15...1.25$ .

### 6.5.2. Calculation of Altitude Performance of Lubrication System

Altitude performance of lubrication system is the maximum flight altitude up to which the pressure pump provides the required oil flow at minimal allowable pressure.

Diameter of suction pipeline is determined as a result of the calculation. Parameters of the pressure pump, the maximum pressure valve in the tank, etc. are also checked. The scavenge line usually is not calculated. Diameter of the scavenge pipeline is assumed equal to  $d_s = (0.8...1)d_p$ .

Condition of lubrication system calculation is the required oil flow through the engine. Oil pressure in the pressure pump inlet is accordingly determined.

Calculation of oil pressure in the pressure pump inlet is carrying out similarly to the calculation of fuel system

$$P_{IN} = P_H + \Delta P_T + \rho_o g(y_T - y_I) - \Delta P_{IN} - \Delta P_{hid} \geq P_{IN.min},$$

where  $P_H$  — atmospheric pressure at the altitude;  $\Delta P_T$  — overpressure in the tank;  $\rho_o$  — oil density;  $(y_T - y_{IN})$  — relative height of the tank over the pump inlet level;  $\Delta P_{IN}$  — inertial pressure losses;  $\Delta P_{hid}$  — hydraulic pressure losses at the oil motion;  $P_{IN.min}$  — minimal allowable oil pressure in the pump inlet.

The hydraulic losses are determined by the known formula

$$\Delta P_{hid} = \left( \lambda \frac{l}{d} + \sum \xi_{form} \right) \frac{\rho_o V_o^2}{2}.$$

However, the flow mode is assumed as laminar, i. e.  $\lambda = K \cdot 64/Re$ . The factor  $K$  takes into account the increase of the friction losses because of oil cooling (according to the viscosity increase) near the pipeline surface.  $K = 1.1$  — for heated pipelines;  $K = 1.3$  — for non-heated ones.

One more difference is that the overpressure in the oil tank is created not by external sources, but due to allocation of air from the air–oil emulsion. This overpressure is determined by the formula, which is analogous to one while calculating the fuel system venting:

$$\Delta P_T = \left( \lambda \frac{l_v}{d_v} + \sum \xi_{form.v} \right) \frac{\rho_{air} V_v^2}{2},$$

where  $\lambda$ ,  $l_v$ ,  $d_v$ ,  $\xi_{form.v}$ ,  $\rho_{air}$ ,  $V_v$  — friction drag coefficient, length, diameter of the pipeline, its factors of form losses, density and speed of air in the vent system, respectively.

Overpressure in the oil tank is usually maintained constant by the installation of the proportional valve in the oil tank outlet to the venting pipeline. This pressure should be minimally required, as it increases mass of the oil tank. It is approximately  $\Delta P_T = 10...25$  kPa.

The minimal allowable pump inlet pressure  $P_{IN.min}$  is determined by cavitation pump characteristics (Fig. 6.9). To calculate them, we should do the following.

1. Determine required oil flow through the engine by the condition of rejection of required heat quantity.
2. Lay off the value of this flow on the ordinate axis (point **a**).
3. Draw a horizontal (**ab**) upto an interception with the cavitation pump characteristic corresponding to the assumed contents of air in oil.
4. Drop a perpendicular from the obtained intersection point (point **b**) to the abscissa axis (point **c**).

If altitude performance of the lubrication system is not provided, the following measures can be used: increase of overpressure in the oil tank, the booster pump installation, the increase of pipeline diameter or use of another lubrication system scheme.

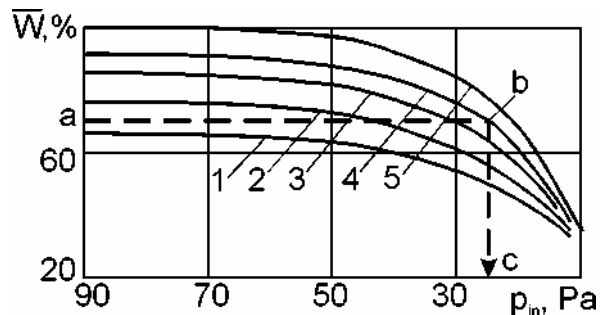


Fig. 6.9. Finding Minimal Allowable Pump Inlet Pressure

## 6.6. Oil Cooling. Coolers

Coolers are devices intended for transferring heat from the cooled medium to the cooling one. Coolers are named by the medium: cooling-cooled: air-oil, fuel-oil, water-oil coolers.

Air-oil coolers are usually applied on airplanes with piston or turbo-prop engines. In this case, fuel-oil coolers are not used, as far as the heat flow into oil is considerable, but fuel consumption is low. Thus fuel can not cool oil down to acceptable temperature.

Fuel-oil coolers are usually used on airplanes with (bypass) turbo-jets. It provides:

[+] Elimination of power consumption intended for overcoming the external drag of the cooler installation;

[+] Preheating of fuel before supplying it to the engine;

[+] Automatic regulation of oil temperature while changing the engine power, i.e. the fuel consumption.

Besides on supersonic airplanes, the air stagnation temperature can exceed the oil temperature, thus, preventing the cooling process. Therefore for such airplanes fuel-oil coolers will be used only.

According to the design we can distinguish two types of coolers: honeycomb and finned.

**Honeycomb coolers** (Fig. 6.10) consist of a set of copper or brass pipes of 250...300 mm long, with the wall thickness of 0.1...0.2 mm and the 4...5 mm in diameter, concluded in a steel casing.

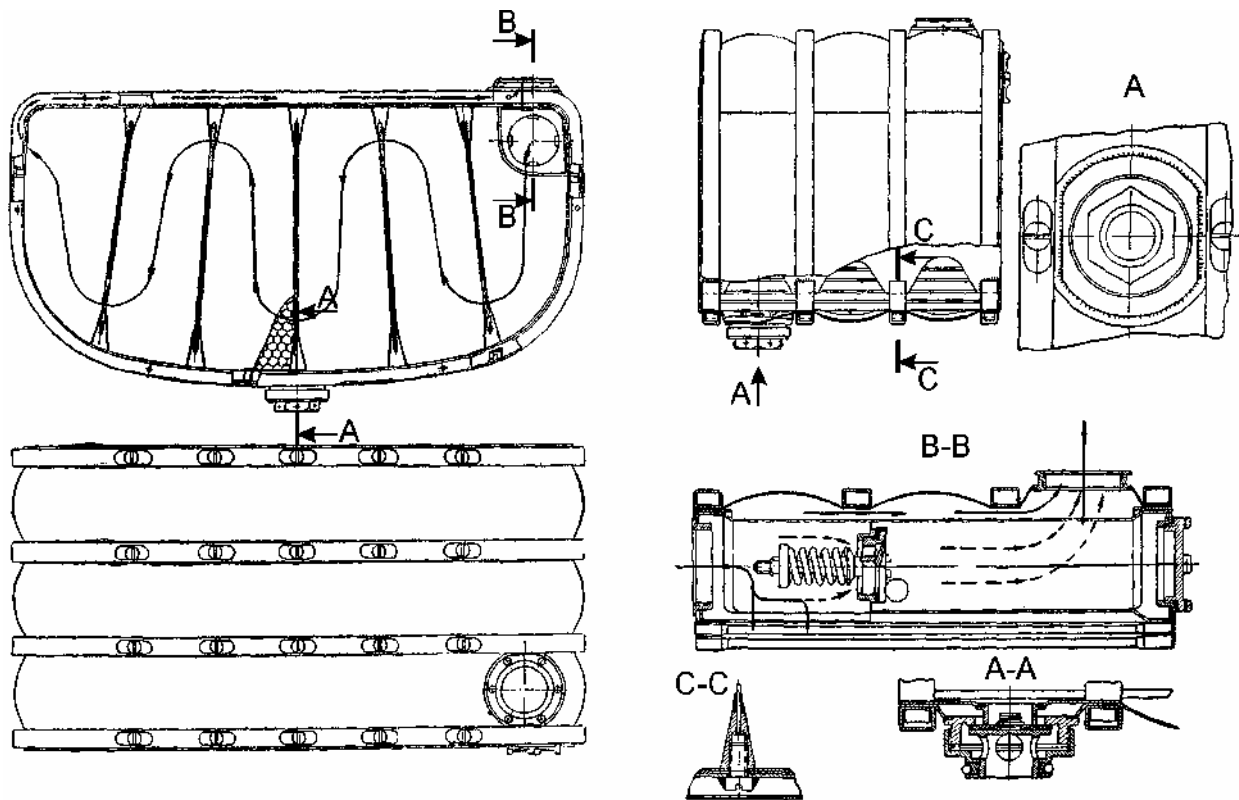
The pipe tails at the  $l = (2...3)d$  segment are flared to a hexahedron and hermetically soldered. Thus there are slots among the pipes, through which the hot oil passes. The cooling medium flows inside the pipes. To increase the oil path and to decrease the cooler sizes, bulkheads are installed in the cooler. Besides, these bulkheads provide strength and stiffness of the cooler structure.

The honeycomb coolers are convenient in service. In case of tube leakage, it is possible to unsolder it and replace or solder both ends.

The shape of coolers is selected in compliance with arrangement on the airplane: round, oval, horse-shoe.

The other type of coolers is **finned coolers** (Fig. 6.11). They consist of a set of streamlined pipes, made of aluminum alloys and welded to the perforated plates. The oil flows inside these pipes, and the air passes in the space among the pipes.

Finned coolers have approximately 25% smaller mass, they are stronger than honeycomb ones. Their tightness does not depend on quality of welding. The finned coolers have stable heat conduction characteristics. Their only disadvantage is that it is impossible to repair a finned cooler in field conditions.



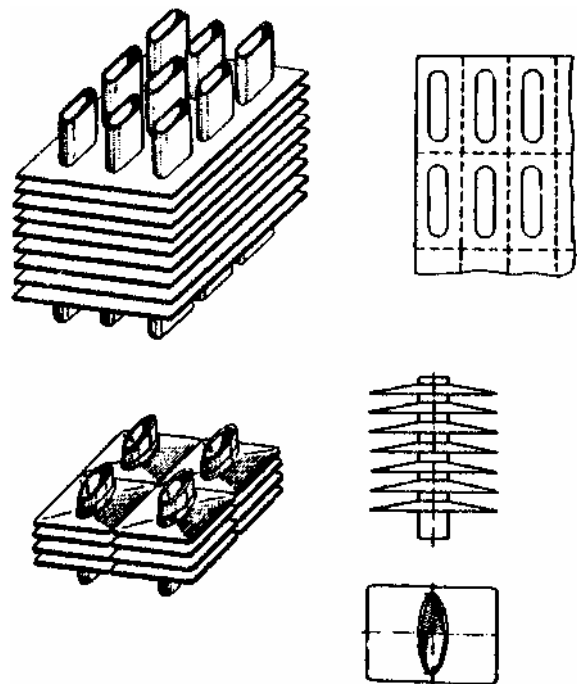
**Fig. 6.10. Design of Honeycomb Air-Oil Cooler**

The coolers are attached by steel attaching straps with damping rubber plates, analogously to the stiffen fuel tanks attaching (ref. Fig. 5.36).

### 6.7. Oil Tank Structure

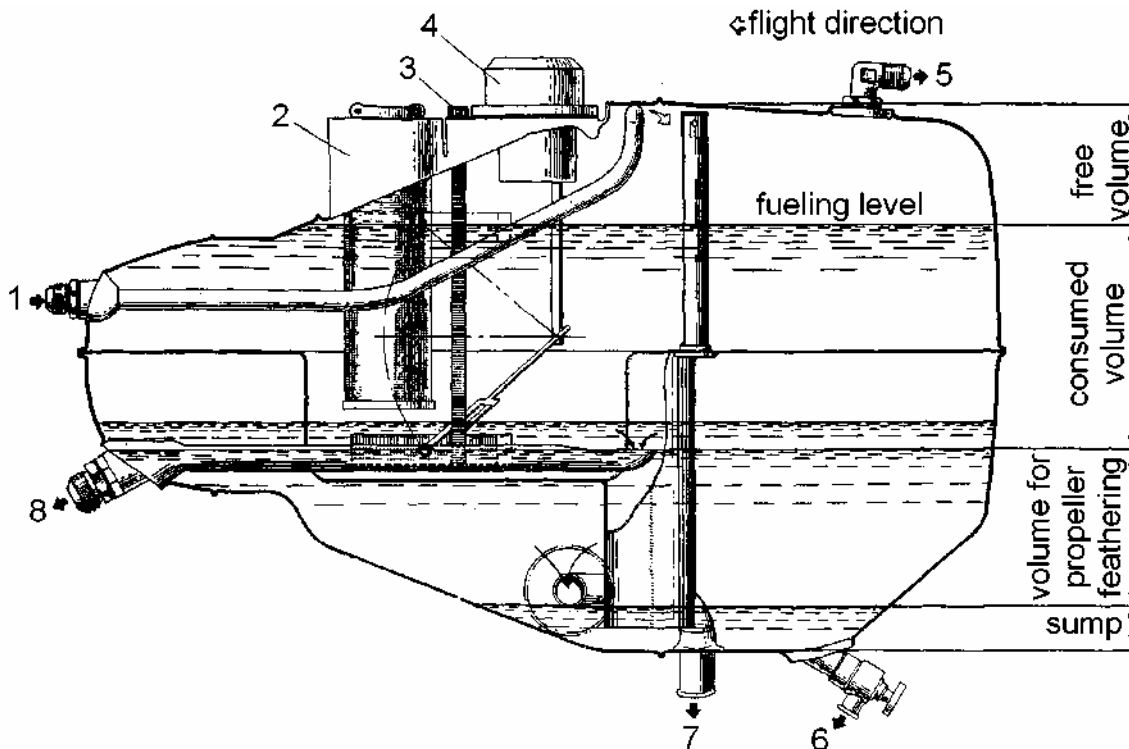
The structure of oil tanks (Fig. 6.12) is very similar to that of fuel tanks. Now the stiffen oil tanks are usually used. They are made of aluminum-magnesium or aluminum-manganese allows. The main difference between oil and fuel tanks consists in the installation of the air separation devices and the pipe connection for return the oil to oil tank.

There are various types of de-aerators. For instance let us consider the pan de-aerator and the centrifugal de-aerator-well.



**Fig. 6.11. Design of Finned Air-Oil Cooler**



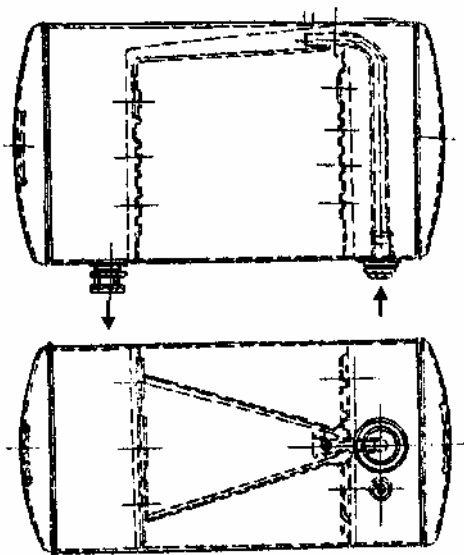


**Fig. 6.12. Oil Tank Structure of Turboprop Airplane:**  
**1 — Air–Oil Emulsion Input from Engine; 2 — Filler Neck; 3 — Measuring Bar; 4 — Oilmeter Sensor; 5 — Venting; 6 — Drain; 7 — Oil Outlet for Propeller Feathering; 8 — Oil Outlet to Engine**

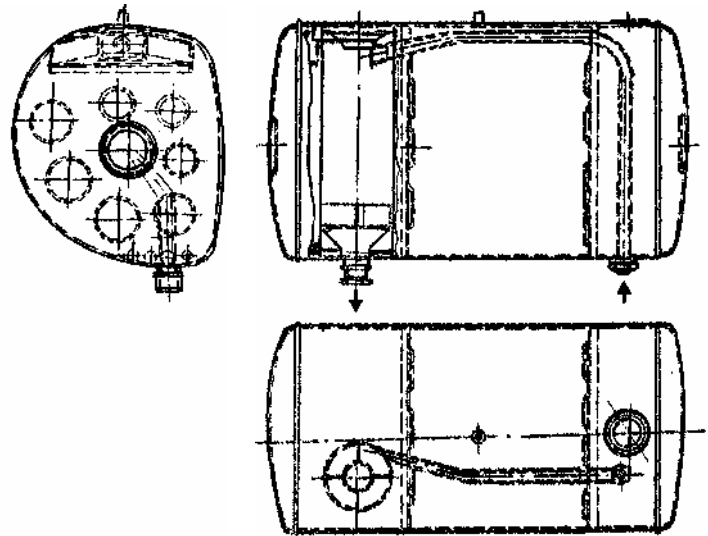
In the **pan de-aerator** (Fig. 6.13) a principle of sediment is used. After entering into the oil tank, the oil stream is directed to the inclined surface—pan, extending to the bottom. Sometimes a grid is installed instead of a pan. The oil freely drains through such grid, but the surface tension prevents air bubbles from passing through it. Such de-aerators are usually used in flat tanks up to 350...400 mm height.

If the oil tank is high, the **centrifugal de-aerator–well** is usually used (Fig. 6.14). Oil gets into the de-aerator tangentially to the well wall and leaves it in the same way or along the axis of the well. In the first case, the oil is pressed to the well wall by the centrifugal force. In the second case, the straightening cross is installed below in places of the oil exit. The separated air leaves the de-aerator through special openings above.

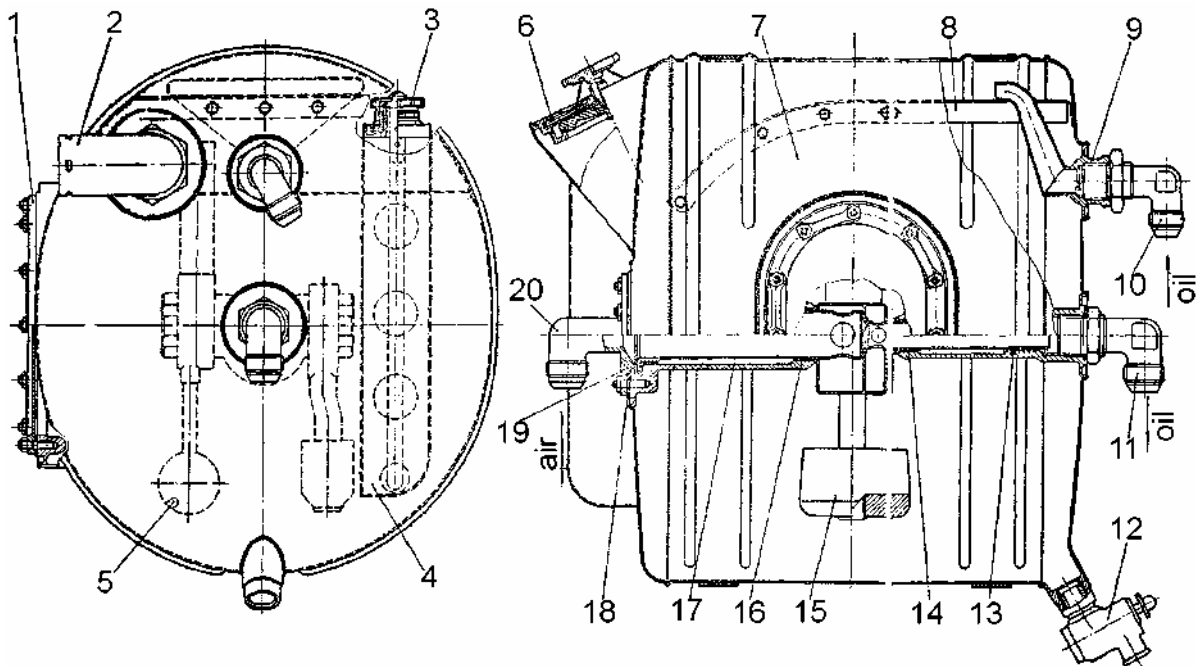
The pipe connection for the oil return should be installed in the top of the oil tank. Thus duration of oil availability in the oil tank increases and oil in full capacity is circulated. The venting connection disposes in the top of the oil tank. If the oil tanks are wide or long, it is recommended to make two venting connections. To provide reliable operation of lubrication system at acrobatic and maneuverable aircraft under decelerations, special devices are installed inside the oil tanks. One example of the devices is shown in Fig. 6.15.



**Fig. 6.13. Pan De-Aerator**



**Fig. 6.14. Centrifugal De-aerator-Well**



**Fig. 6.15. Oil Tank with Device Ensuring Oil to Engine Supply at Decelerations:**

- 1 — Technological Hatch Covering; 2 — Pipeline Connection;
- 3 — Measuring Bar; 4 — Measuring Pocket; 5 — Air Intake Counterweight;
- 6 — Filler Neck; 7 — Tank; 8 — Pan De-Aerator; 9 — Pipe Supplying Oil on the Pan De-Aerator;
- 10 — Air-Oil Emulsion Pipeline Connection; 11 — Oil Outlet;
- 12 — Drain Valve; 13 — Hydraulic Seal Ring; 14 — Bushing;
- 15 — Oil Intake with Counterweight; 16 — Bushing; 17 — Axis;
- 18 — Washer; 19 — Ring; 20 — Air Outlet

## Theme 7. INTERNAL AERODYNAMICS OF POWER PLANTS

### 7.1. General Questions and Problems of Internal Aerodynamics

The section of applied aerodynamics studying the air flow inside an aircraft is called **internal aerodynamics**. Respectively, external aerodynamics studies flow around aircraft by a counter airflow.

The most important problem of the internal aerodynamics is organization of airflow inside a power plant, which determines the engine cooling, power losses on cooling, engine thrust, etc.

Air ducts located inside the airplane should have the minimal pressure losses. In some cases the uniform field of speed in front of the air consumer is also required. It is essential for turbojet compressors and coolers.

Besides, it is required to organize the input, and sometimes the output of airflow so, that the related increase of aerodynamic drag is minimal. It is necessary to note, that requirements of internal aerodynamics often come into conflict with those of external aerodynamics.

### 7.2. Standard Air Duct Segments. Pressure Losses During Air Motion

There are air ducts of various overall dimensions and designs. However the main air ducts segments are always the same.

To provide the visualization of this statement we shall consider an axial-symmetric jet engine air duct (Fig. 7.1, a) and an air-oil cooler air duct (Fig. 7.1, b).

The **H** cross-section corresponds to the undisturbed airflow. In theory it lies on the infinite distance far from the **e** cross-section.

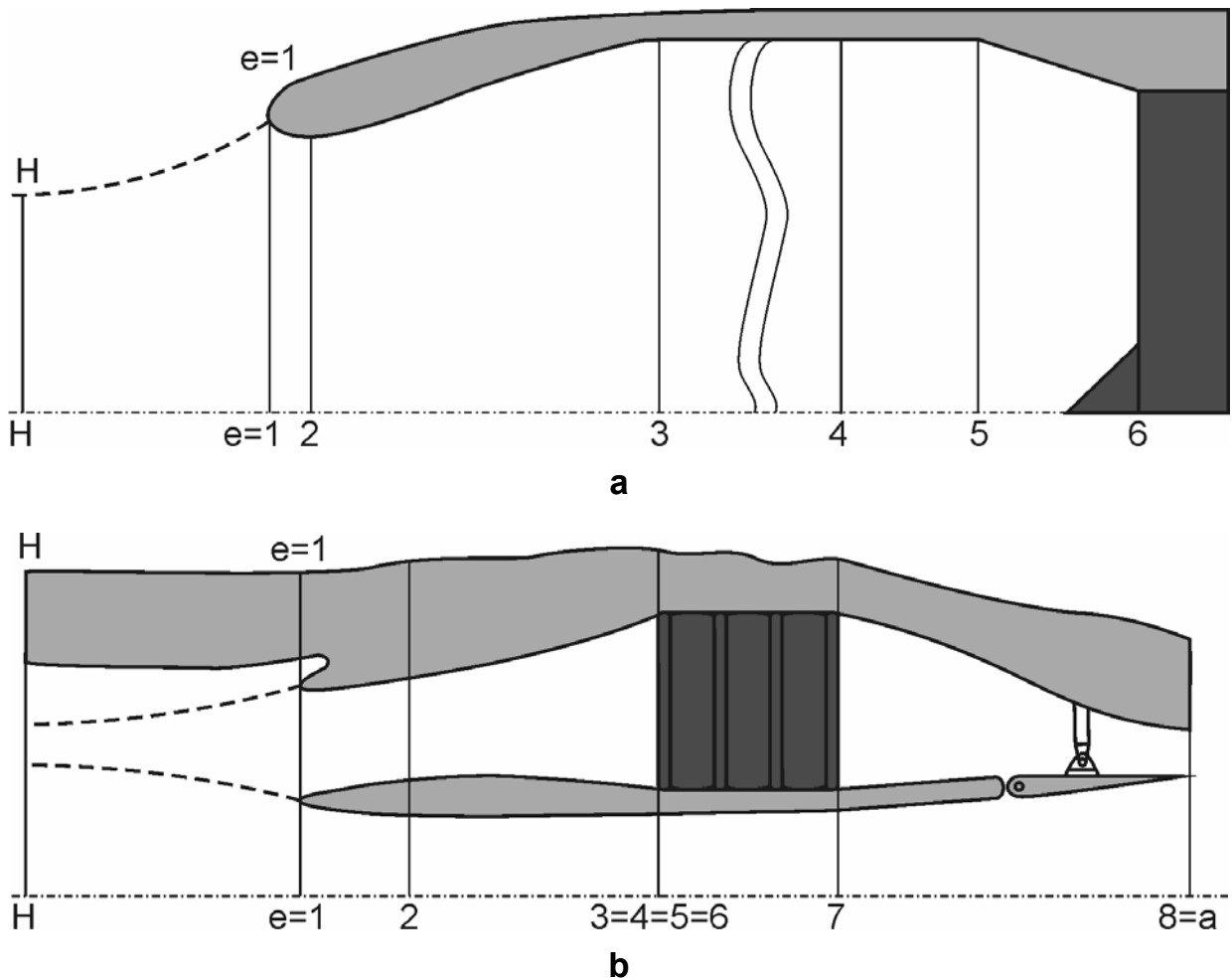
**The affluence segment (H-e)** of the sucked air is located in front of the air intake. The purpose of this segment is to brake the airflow and increase pressure. That's why it is also called the external diffuser.

**The input segment (1-2)** is designed in the form of confuser to ensure the small airflow compression, in order to avoid the airflow separation in the air intake at the big angles of attack or slipping. This segment is far shorter than the following one.

**The diffuser (2-3)** is a widening duct in which the airflow speed decreases, and the pressure increases.

**The air duct channel (3-4)** is required if the air intake lies far from the air consumer for the design reasons. Sometimes, this segment has a curved axis according to the onboard airplane arrangement.

**Segment 4-5.** In this segment, the axis of channel 3-4 is aligned with the axis of the air consumer. This segment is required, only if these axes do not coincide. The segment is  $(0.5...1)d$  long.



**Fig. 7.1. Standard Air Duct Segments**

**Segment 5–6** represents a confuser. In this segment airflow speed slightly increases to obtain more uniform field of speed in front of the air consumer.

**Segment 7–8** presents only in cooler ducts. In this segment the airflow is accelerated again to decrease external drag:

$$X_{P,P} = m_A (V_H - V_a),$$

where  $m_A$  — mass second air consumption, through the channel;  $V_H$ ,  $V_a$  — ram airflow and airflow speeds in a cross-section. Apparently, if speed  $V_a$  is less, than  $V_H$ , additional drag appears.

Pressure losses at air motion in the air duct are conditionally subdivided into two kinds: friction losses and losses in diffusers.

Friction losses are calculated by the usual formula

$$\Delta P_{fr} = \sum_{i=1}^n \lambda_i \frac{l_i}{d_i} \frac{\rho V_i^2}{2},$$

where  $i$  — a number of segment having approximately constant area and perimeter of the channel cross-section;  $d_i = 4F/P$  — a hydraulic diame-

ter of the channel cross-section;  $F$ ,  $P$  — the area and perimeter of the channel cross-section;  $\lambda_f$  — the coefficient of friction, which is taken from reference books; other parameters are obvious.

### 7.3. Diffuser. Losses in Diffusers

Diffuser is a channel, at motion along which speed of flow decreases. It is known that there are subsonic and supersonic gas flows. Respectively, we subdivide the diffusers into subsonic and supersonic. In this section we shall consider designing of subsonic diffusers. Designing of supersonic diffusers will be considered in the «Supersonic air intakes» section. It is known that subsonic flow brakes in the widening duct. Thus a subsonic diffuser is a widening duct, along which axis the flow speed decreases, and pressure increases.

The formula for calculation of losses at sudden expansion (impact), (Fig. 7.2, a) is known from the hydraulics:

$$\Delta P_{impact} = \frac{\rho}{2}(V_1 - V_2)^2 = \frac{\rho V_1^2}{2} \left(1 - \frac{V_2}{V_1}\right)^2 = \frac{\rho V_1^2}{2} \left(1 - \frac{F_1}{F_2}\right)^2,$$

where  $V_1$ ,  $V_2$  — speeds before and after sudden expansion of the channel;  $F_1$ ,  $F_2$  — cross-sectional areas of the channel before and after expansion.

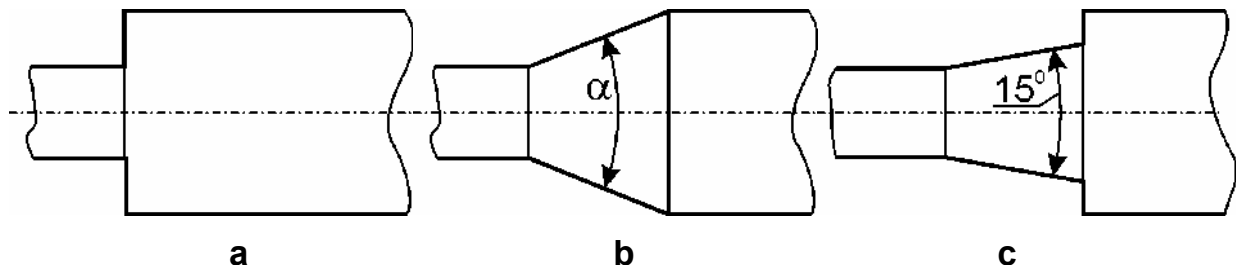


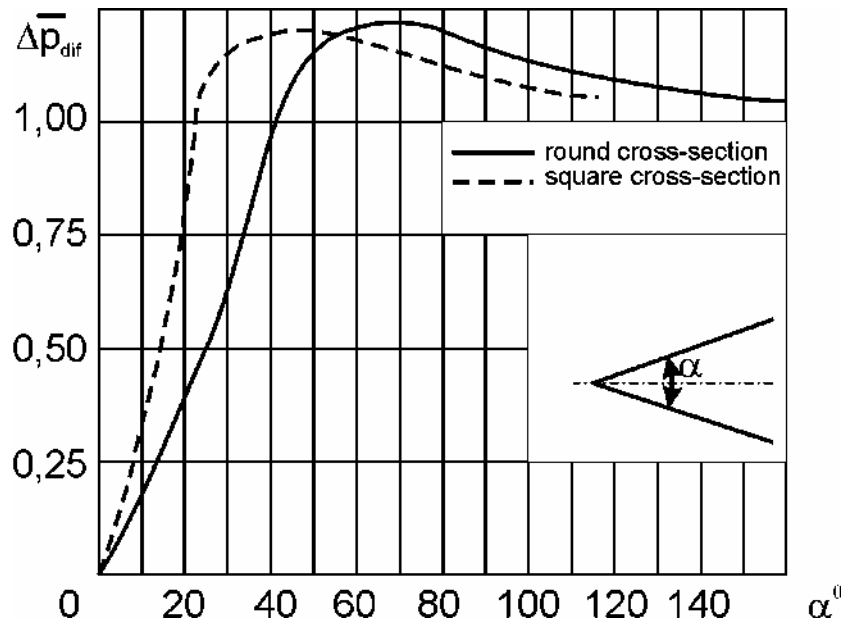
Fig. 7.2. Types of Diffusers with Straight-line Generatrix

In calculations the pressure losses in the diffuser are usually compared with the losses at impact, using the impact softening factor:

$$\overline{\Delta P}_{dif} = \frac{\Delta P_{dif}}{\Delta P_{impact}}.$$

### 7.4. Types of Diffusers and Their Shaping

The simplest type of the diffuser is the diffuser with straight-line generatrix and constant expansion angle (Fig. 7.2, b). The relation of the impact softening factor from the diffuser expansion angle for diffusers with straight-line generatrix is shown in Fig. 7.3.



**Fig. 7.3. Relation of Impact Softening Factor from Diffuser Expansion Angle for Diffusers with Straight-Line Generatrix**

Apparently, even if the angle is  $40^\circ$  ( $25^\circ$  for square cross-section), losses in the diffuser become equal to losses at impact. The optimal expansion angles  $\alpha$  (angles at which the sum of friction losses and expansion losses is close to minimum) are  $6...10^\circ$  for round diffuser cross-section and  $5...8^\circ$  for square diffuser cross-section. This results in the long length of diffusers (and as a result big mass), which is undesirable, because of the growth of friction losses and complications of their arrangement onboard an aircraft.

The main pressure gain and losses in the diffuser are concentrated in the beginning part of the diffuser (where the speed is high). Hence, it is necessary to provide good performance of the diffuser in this section. Even if thus, performance of a tail section where the speed is low will worse. It is the main idea of designing of diffusers of small length.

The stepwise diffuser is a simple realization of this idea (ref. Fig. 7.2, c).

Another implementation of the idea is the application of **diffusers with curvilinear generatrix**, which have the small expansion angles in the high-speed section and large in the low-speed field. Besides, the curvature of the walls determines a positive pressure gradient from the wall to the axis, which reduces the danger of the airflow separation. The most successful are curvilinear diffusers with a constant length pressure gradient ( $dP/dx = \text{const}$ ) and with a constant length velocity gradient ( $dV/dx = \text{const}$ ) (Fig. 7.4, 7.5). Fig. 7.4 shows three diffusers with the same pressure losses and Fig. 7.5 shows three diffusers with the same

length.

First from them is the best if the short ducts with the large rate of braking are designed. Usually they are used for inlet ducts of cooler tunnels and cowlings of piston engines.

Diffusers of the second type are widely used for jet engines air intakes. As they have smoother outlines and provide more uniform field of speed in the jet engine inlet. Thus, they usually result in greater length.

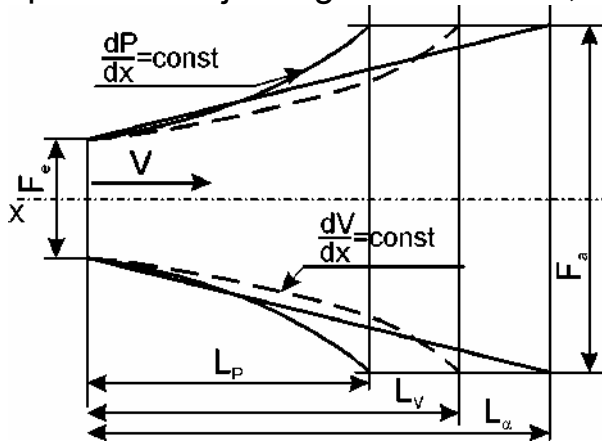


Fig. 7.4. Diffusers with Curvilinear Generatrix

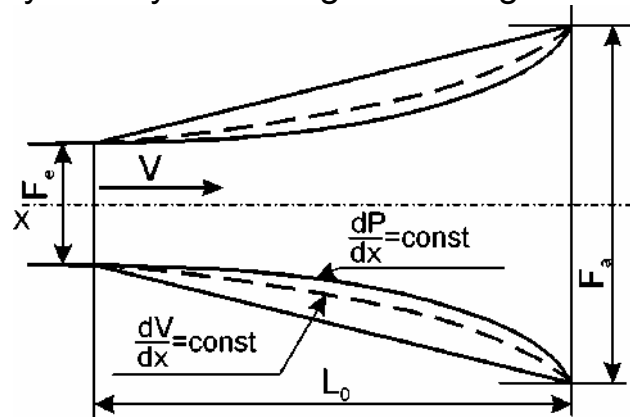


Fig. 7.5. Diffusers with Curvilinear Generatrix

Besides the considered questions, when designing the channels of air ducts, a lot of other problems are also solved, such as: the stream compression to provide uniformity of the field of speed, the stream turns, struggle against boundary layer at entry of air intake, provision of rational stream exit from the cooler ducts, the provision of the air intakes performance at the angles of attack and slipping, the pressurization of air duct channels.

## Theme 8. AIR SUCTION SYSTEM. SUBSONIC INLETS

### 8.1. Air Suction System, its Purpose, Requirements and Components

**Air suction system is intended** to intake and supply the required quantity of air to air consumer at all operation modes. Thus, it is required to provide the transformation of the stream kinetic energy into the pressure potential energy with minimal losses.

The following **requirements** are made to the **air suction system**:

1. Supply the engine by the required quantity of air at all its operational modes.
2. Provide the minimal losses of impact air pressure when it brakes from the flight speed up to 40...60 m/s at the engine inlet.
3. Minimal external drag.

4. Provide the minimal mass, sufficient strength and stiffness, good manufacturability.

5. Provide the operational stability without stalls and pressure fluctuation.

6. Provide the sufficient uniformity of speed and pressure fields at the air consumer inlet.

7. Ensure the protection against icing and ingestion of foreign matter.

**The air suction system consists** of the inlet (air intake), the air consumption control device and devices intended to protect the engine against ingestion of foreign matter.

## **8.2. Jet Engines Overall Compressor Pressure Ratio. Impact Pressure Recovery Factor**

The jet engine overall compressor pressure ratio  $\pi$  is determined as the ratio of the impact air pressure behind a compressor to the undisturbed airflow static pressure by the formula

$$\pi = \frac{p_c^*}{p_H} = \frac{p_{in}^*}{p_H} \frac{p_c^*}{p_{in}^*} = \pi_{in} \pi_c^*,$$

where  $p_c^*$  — the impact pressure behind a compressor;  $p_H$  — the air pressure at the  $H$  altitude;  $p_{in}^*$  — the impact air pressure in front of the jet engine compressor;  $\pi_{in}$  — the air intake pressure ratio;  $\pi_c^*$  — the jet engine compressor pressure ratio.

At subsonic flight speed, the air pressure rises mostly in the jet engine compressor. At supersonic speed, the pressure is increased mostly in the air intake.

When airflow braking, the friction and vortex formation causes pressure losses, and the pressure losses in shock waves appear in supersonic airflow. Air braking efficiency in the air intake is determined by air pressure losses at braking of airflow and the friction losses in an air intake duct. The losses, caused by the air compression in the air intake, are estimated by the impact pressure recovery factor  $\sigma_{in}$ . This factor represents the ratio of the impact pressure in front of the compressor to the impact pressure of the isentropic broken stream:

$$\sigma_{in} = \frac{p_{in}^*}{p_H^*}.$$

Decreasing the  $\sigma_{in}$  value leads to the pressure decrease at the compressor inlet  $\left( \pi_{in} = \sigma_{in} \frac{p_H^*}{p_H} \right)$ , the thrust decrease and the increase of



fuel consumption. (For example, decreasing  $\pi_{in}$  by 30% at  $M=2.5$  causes the 45% engine thrust decrease and 15% increase of the specific fuel consumption). Therefore, one of the main requirements made to inlets is to provide the greatest possible  $\sigma_{in}$ .

### 8.3. Air Intake Classification

The type of air intake depends on the design  $M$  number, the required  $M$  number range, the type of engine, its arrangement, and other factors.

According to the flight speed, air intakes are divided into subsonic, transonic and supersonic (Fig. 8.1). According to the arrangement on aircraft, the air intakes can be nose, lateral, ventral, under-wing.

**Nose air intakes** are located in the nose of a fuselage or in the front of an engine nacelle. The advantages of these air intakes include:

- [+] design simplicity;
- [+] uniformity of the entry field of speed;
- [+] low hydraulic pressure losses;
- [+] absence of entry boundary layer.

**Lateral air intakes** are located on each side of a fuselage. They can be semicircle, flat with vertical ramp, flat with horizontal ramp (scoop). Such air intakes have usually specific features for example: high hydraulic pressure losses and the entry boundary layer. However, thus the fuselage volume is used much better for the arrangement of fuel and equipment.

**The ventral and under-wing air intakes.** The deficiency of these air intakes consists in the bad operation at the big negative angles of attack. At the same time such arrangement of air intakes is favorable for supersonic aircrafts. As the supersonic airflow can be broken in a shock system of the airplane, it is possible to apply the subsonic air intakes at supersonic aircraft (F-16, F-18). If the supersonic air intake is used, the aircraft lift is increased by the pressure growth in front of it, as the pressure is applied to the lower surface of the fuselage or the wing.

Besides, air intakes are classified by the cross-section shape, the boundary layer control method and degree of mechanization.

### 8.4. Designing Model and Design Algorithm of Subsonic Air Intakes

The initial data at designing the air intakes include: the flight speed  $V_H$ , the flight altitude  $H$ , the engine diameter by the compressor  $D_{en}$  and the air consumption  $G_a$  [kg/sec] for this flight mode.

Within the subsonic air intake design, four problems should be solved: the inlet area determination; the external and internal outlines shaping and the air intake lip forming.

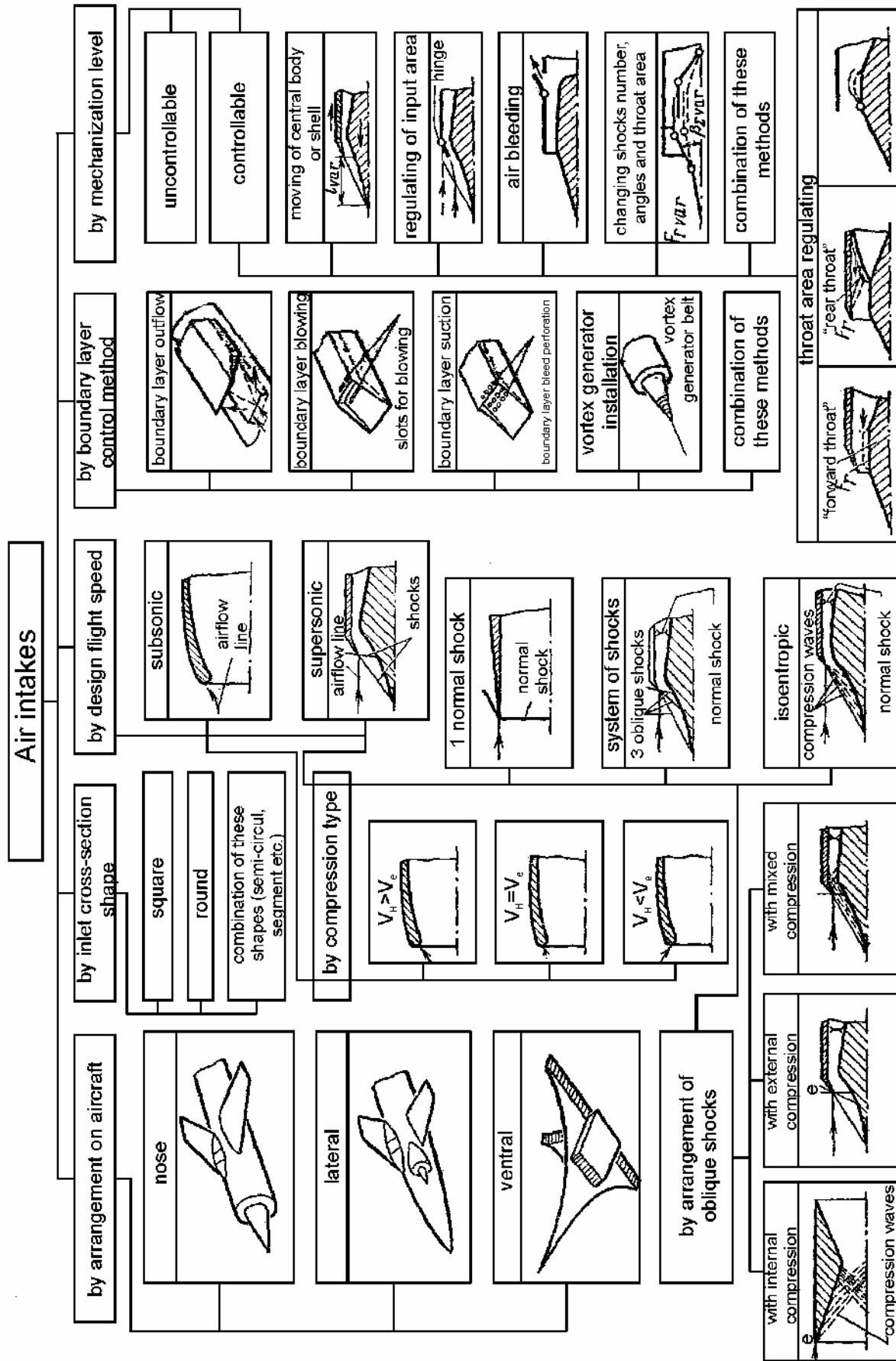
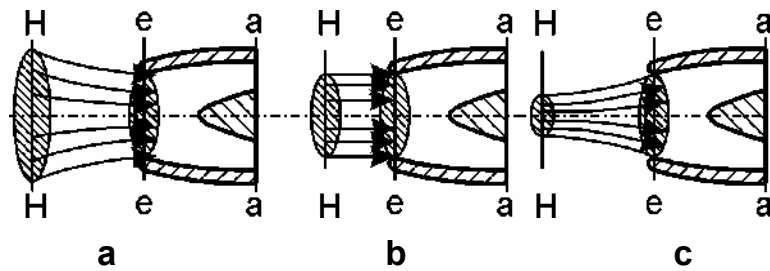


Fig. 8.1. Air Intake Classification

The airflow in front of the air intake depends on the ratio of the flight speed  $V_H$  and air speed  $V_e$  into the inlet. There are possible three flow modes (Fig. 8.2).



**Fig. 8.2. Operational Modes of Subsonic Air Intake**

At  $V_H < V_e$ , the airflow accelerates in front of the air intake (Fig. 8.2, a). At  $V_H = V_e$ , the air stream enters the diffuser without changing the shape (Fig. 8.2, b). At  $V_H > V_e$ , the airflow decelerates in front of the air intake (Fig. 8.2, c), and the pressure losses decrease in the air intake. The last case corresponds to the design mode of the air intake operation.

The **inlet area** is determined by the formula

$$F_e = \frac{G_a}{V_e \rho_{com}},$$

where  $F_e$  — the inlet area,  $m^2$ ;  $V_e$  — the air speed into inlet,  $m/s$ ;  $\rho_{com}$  — the air density into the inlet taking into account compressibility,  $kg/m^3$ :

$$\rho_{com} = \rho_H \left[ 1 + 0.5 M_H^2 (1 - \bar{V}_e^2) \right],$$

where  $\rho_H$  — the air density at the flight altitude,  $kg/m^3$  (is taken from the International Standard Atmosphere);  $\bar{V}_e = V_e / V_H$  — the relative speed into inlet  $\bar{V}_e = 0.2...1$ . In order to provide good external airflow about, it is recommended to assume  $\bar{V}_e = 0.5...0.7$ .

If the air consumption through the engine at design flight mode is unknown, it can be determined by the formula

$$G_a = \frac{\alpha \cdot K \cdot C_p(M, H) \cdot P(M, H)}{3600},$$

where  $\alpha$  — quantity of air which is theoretically required for complete combustion of 1 kg of kerosene,  $kg/kg$ ;  $K$  — the excess air coefficient;  $C_p(M, H)$  — the specific fuel consumption at the design mode,  $kg/N \cdot hour$ ;  $P(M, H)$  — engine thrust at the design mode,  $N$ .

The **external outlines and lip** are shaped by condition of minimum aerodynamic drag. That's why it is necessary to obtain radius of curvature in mid-section as greater as possible and smooth lengthwise growth of the radius of curvature.

Elliptical outlines meet these requirements. In this case, the design begins with the «skeletal» line, in the form of ellipse. Then, an external outline will be put aside from this «skeletal» line, in the form of ellipse too (Fig. 8.3).

The location of the «skeletal» line is determined by the affluence angle  $\beta$ , determined by the formula, rad:

$$\beta = \frac{\pi \cdot K_{\beta}}{180} \sqrt{\frac{1}{V_e} - 1}.$$

The  $K_{\beta}$  coefficient depends on the shape of the air intake in the front view:  $K_{\beta} = 22$  — for round air intakes;  $K_{\beta} = 26$  — for the rectangular air intakes.

To design the «skeletal» line, we use the equation of an ellipse, which major semiaxis equals to  $L + L_0$ , and the minor semiaxis equals to  $D_{M0}/2$ :

$$\frac{x^2}{(L + L_0)^2} + \frac{y^2}{(D_{M0}/2)^2} = 1.$$

The external outline is shaped by the ellipse with semiaxes  $L_{\delta}$  and  $\delta$ . The ordinates of the ellipse are put aside normally to the «skeletal» line

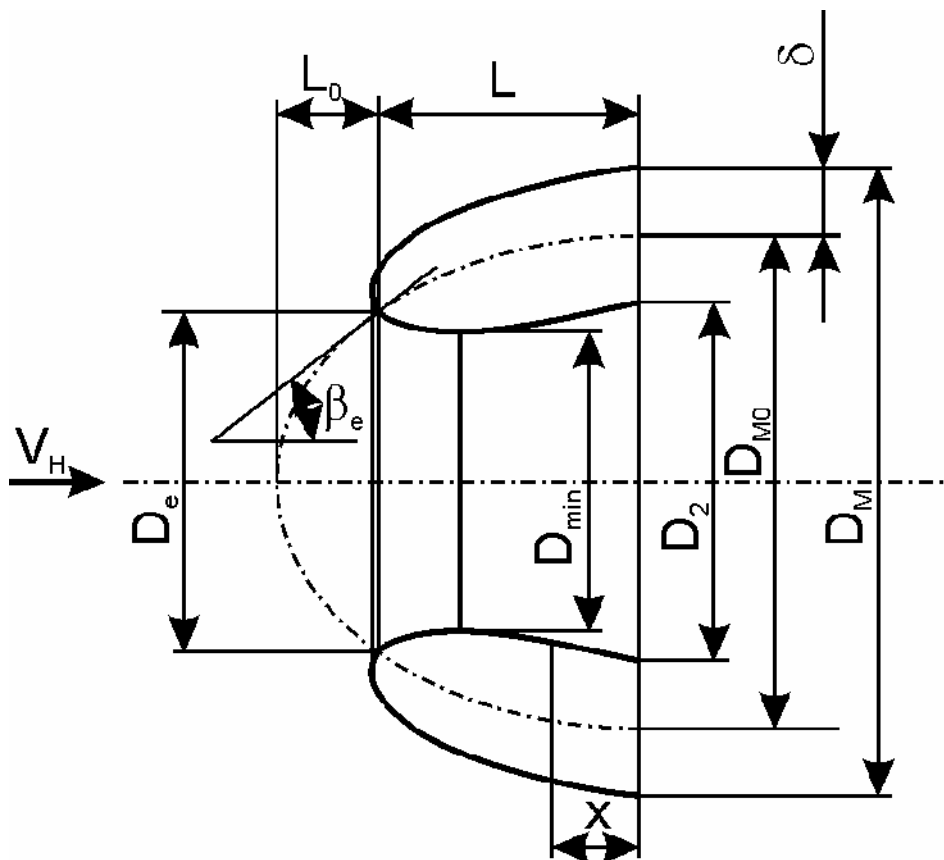


Fig. 8.3. Designing Model of Subsonic Air Intake

$$\frac{x_{\delta}^2}{L_{\delta}^2} + \frac{y_{\delta}^2}{\delta} = 1.$$

The radius of curvature of air intake lip is approximately assumed:

$$\rho_{\min} = K_1 \sqrt{F_e}, \quad K_1 = 0.04 \dots 0.05.$$

The engine nacelle mid-section diameter is assumed equal to:

$$D_M = K_2 \cdot D_{en}, \quad K_2 = 1.2 \dots 1.3.$$

The semiaxes  $L_{\delta}$  and  $\delta$  are determined by the ratios:

$$\delta = \frac{\rho_{\min}}{\bar{\delta}}, \quad L_{\delta} = \frac{\delta}{\bar{\delta}}, \quad \bar{\delta} = 0.1 \dots 0.25.$$

The semiaxes of «skeletal» line ellipse are determined by the formulas

$$D_{M0} = D_M - 2\delta, \quad L = \frac{D_{M0}^2 - D_e^2}{2D_e \operatorname{tg}\beta}, \quad L_0 + L = \frac{L}{\sqrt{1 - D_e/D_M}}.$$

To provide the low aerodynamic drag of the external outlines, it is expedient, that

$$\frac{L}{D_M} \geq (1.5 \dots 3) M_H^2,$$

where  $M_H$  — the M flight number.

To provide the good operation of the air intake on the stop, the area of the minimal cross-section is assumed equal to:

$$F_{\min} = K_4 F_e, \quad K_4 = 0.8 \dots 0.9.$$

Thus, the radius of curvature of the lip is obtained equal to:

$$R_l = \frac{D_e - D_{\min}}{2(1 - \sin\beta)}.$$

Finally we assume the greater value of the lip radius of curvature:  $R = \max\{\rho_{\min}, R_l\}$ .

Now all parameters are specified, and it is possible to design the «skeletal» line by the ellipse equation:

$$y = \frac{D_{M0}}{2} \sqrt{1 - \frac{x^2}{(L + L_0)^2}}.$$

Then the ordinates of the ellipse of external outline are put aside normally to the «skeletal» line:

$$y_{\delta} = \delta \sqrt{1 - \frac{x_{\delta}^2}{L_{\delta}^2}}.$$

The **internal outlines** are shaped by one of the previously considered laws:  $dP/dx = \text{const}$  or  $dV/dx = \text{const}$ . In the first case, the follow-

ing formula is used:

$$F(x) = \frac{F_2}{\sqrt{\frac{x}{L - R \cos \beta} \left( \frac{F_2^2}{F_{\min}^2} - 1 \right) + 1}},$$

in the second case: 
$$F(x) = \frac{F_2}{\frac{x}{L - R \cos \beta} \left( \frac{F_2^2}{F_{\min}^2} - 1 \right) + 1},$$

where  $F_2 = K_5 \frac{\pi D_{en}^2}{4}$  — the cross-sectional area of the channel at the end of the diffuser,  $K_5 \geq 1$  — the confuser factor.

### 8.5. Subsonic Air Intake Design

Air intakes of subsonic aircrafts of round cross-section are designed as fuselage-type aircraft assemblies (Fig. 8.4). That is, they consist of transversal frame and longitudinal stringer sets, together with the outer and inner skins. Certainly, the inner skin is riveted by the countersunk rivets to reduce pressure losses.

The air intake lip is usually supplied with the hot air deicing system (Fig. 8.5).

Sealing in air intakes design is reached by using of rubber or other seals in detachable joints in the form of hollow profiles.

The air intakes elements are made of aluminium alloys and high-strength composite materials (fiberglass plastic, boron plastic, black-reinforced plastic).

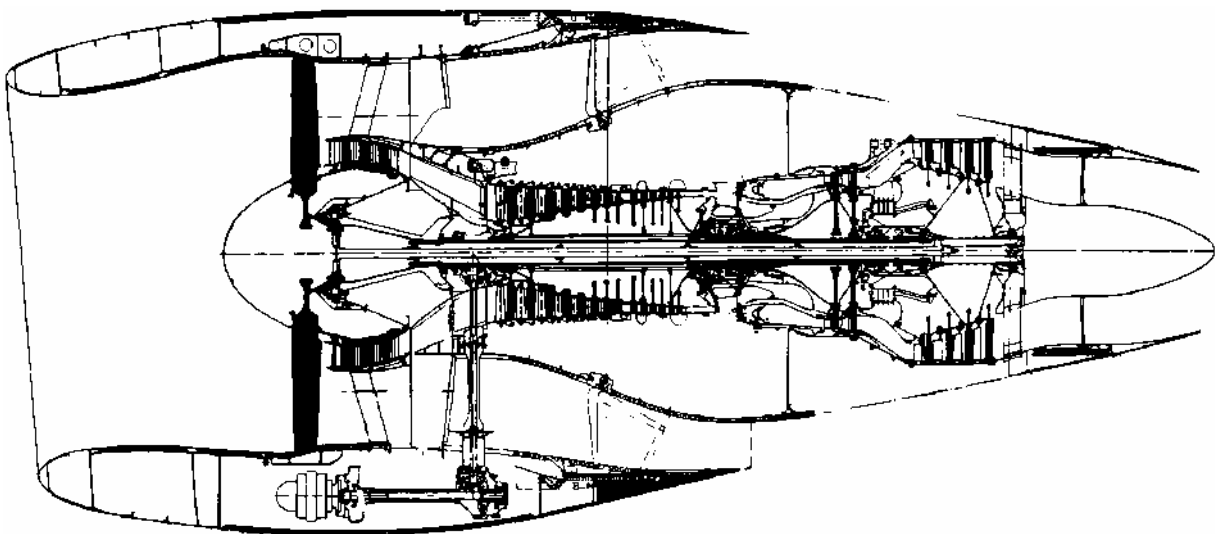
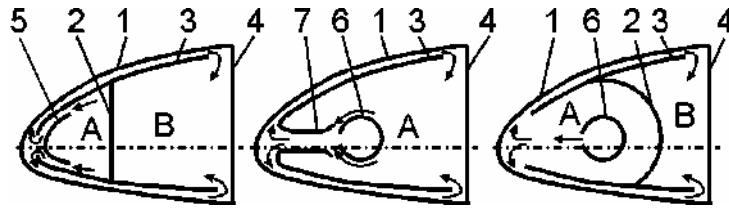


Fig. 8.4. Subsonoc Bypass Turbojet Air Intake



**Fig. 8.5. Hot Air Deicing System of Air Intake Lip:**  
**A — Hot Air Supply Channel; B — Used Air Channel; 1 — Skin; 2 — Web;**  
**3 — Goffer; 4 — Spar; 5 — Shield; 6 — Distributing Pipe; 7 — Mixing Chamber**

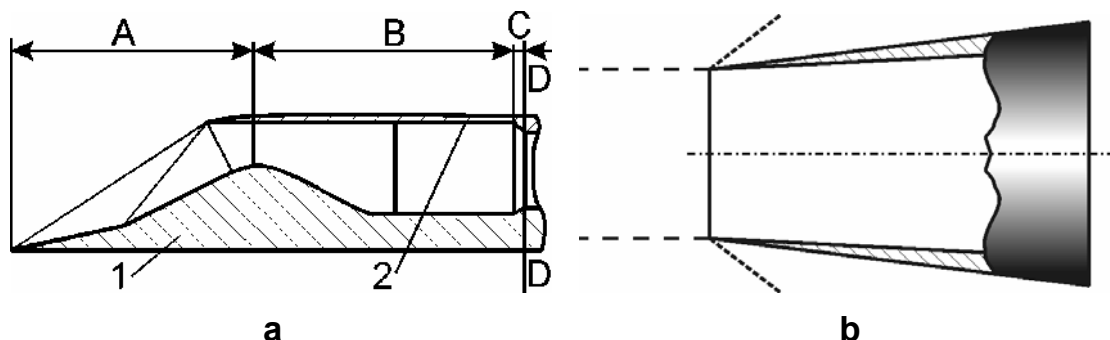
It is necessary to note, that the quality of internal surface of air intakes must meet the high requirements, because of the outstanding rivet heads and ledges in the air duct deteriorate the internal aerodynamics of air intakes.

## **Theme 9. SUPERSONIC AIR INTAKES**

### **9.1. Shocks. Shock Creation Methods.** **Types of Supersonic Air Intakes**

The supersonic air intake (Fig. 9.1, a) usually represents supersonic (A) and subsonic (B) diffusers connected together; being practically an inverted Laval nozzle. Structurally the supersonic air intake consists of a central body 1 and shell 2.

In ideal gas, the transition from supersonic to subsonic flow occurs in the diffuser throat. In such ideal diffuser the non-viscous stream is braked isentropically without formation of shocks and boundary layer in the converging section of the channel, till the sonic speed in the throat ( $M = 1.0$ ), so it operates without impact and static pressure losses.

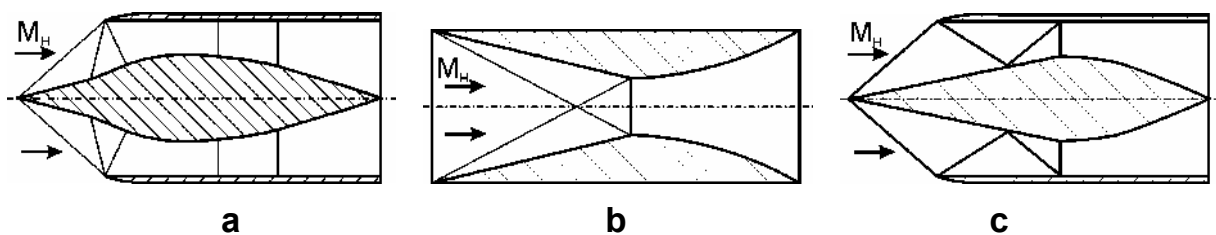


**Fig. 9.1. Scheme of Supersonic Air Intake**

In real conditions it is possible to provide the transition from supersonic to subsonic flow with impact pressure losses (the entropy growth) in a normal (direct) shock. It is implemented in the supersonic air intake. The subsonic stream is further broken in the extending subsonic diffuser.

In order to reduce the impact pressure losses, the supersonic flow is broken by a various systems of oblique shocks, which are finished by a normal shock. These oblique shocks provide the small Mach number before the normal shock, which reduces impact pressure losses. Such systems of shocks are efficient at the high supersonic speed. Up to  $M = 1.5$ , there is no effect from using this type of system of shocks. That is why at such speeds, transonic air intakes with one direct shock are used (Fig. 9.1, b).

According to the location of this system of shocks in the air intake, the following air intakes are available: the air intakes with external compression (Fig. 9.2, a), the air intakes with internal compression (Fig. 9.2, b) and the air intakes with mixed compression (Fig. 9.2, c).



**Fig. 9.2. Schemes of Supersonic Air Intakes by Arrangement of Shock System**

The type of compression is chosen depending on the cruising flight speed, flight speed range and purpose of the aircraft.

The system of shocks with **external compression** is usually formed by the system of turns. It can result in the total turn angle exceeding the limit turn stream angle for the specified Mach flight number (Fig. 9.3). In this case the system of shocks (Fig. 9.4, a) will be destroyed; and the detached head wave is formed in front of the air intake (Fig. 9.4, b). This wave has much greater drag and the higher impact pressure losses, than the system of shocks.

The system of shocks with external compression has the minimal impact pressure recovery factor among three considered schemes. Its advantage consists in the low boundary layer thickness and regulation simplicity. These air intakes are applied for supersonic aircrafts with sharply varying parameters of external airflow, maneuverable, with wide altitude and speed ranges.

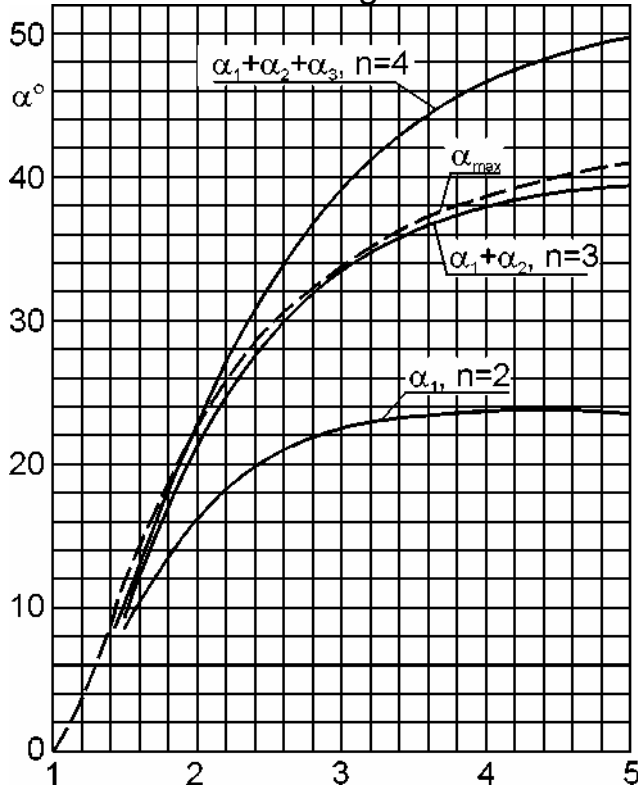
The system of shocks with **internal compression** is arranged completely inside the air intake. As a result, the boundary layer has the maximum impact to the system of shocks here. Such air intakes are more difficult to regulate and start. However, this type of air intake has the highest impact pressure recovery factor among the considered schemes.

The systems of shocks with **mixed compression** are formed by combination of above considered systems. Such systems take intermedi-

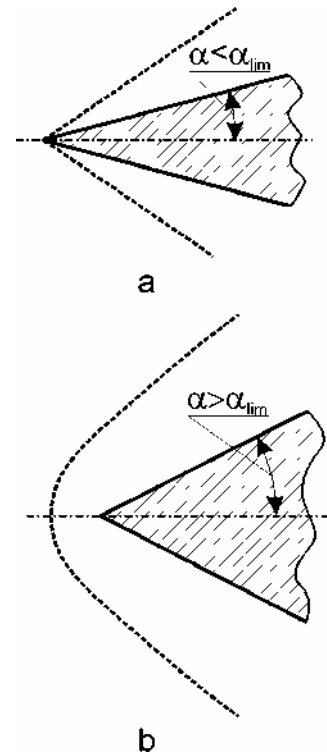


ate place between earlier considered ones by the impact pressure recovery coefficient, by regulation simplicity and by boundary layer impact. It usually has no problem with limit turn stream angle.

By cross-section shape, supersonic air intakes are divided into flat and axially-symmetrical. Flat air intakes have smaller angles of stream deflection and are easier regulated.



**Fig. 9.3. Total Turn Stream Angles  $\alpha_i$  for Optimal External Compression Diffusers with Number of Shocks  $n$  and Limit Stream Turn Angles  $\alpha_{max}$  Dependence on Mach Flight Number**



**Fig. 9.4. Detached Head Wave (a) and Attached Oblique Shock (b)**

## 9.2. Dependence of Impact Pressure Recovery Factor on M Flight Number and Number of Shocks

Optimal supersonic air intake designing is based on the work [37], in which two problems of optimization of system of shocks for flat air intakes have been defined and successfully solved. The first problem has been defined as follows: for the given speed and flight altitude, determine number and arrangement of the oblique shock system with closing direct shock, which provides the maximum impact pressure behind the system of shocks. The second problem: at the same conditions, provide maximum static pressure behind the system of shocks. The first problem has the greatest practical value.

It is necessary to note, that the impact pressure recovery factor of system is maximal, if all impact pressure recovery factors are equal among themselves.

Results of this work are given in tables or graphs of dependence of the maximum impact pressure recovery factor on the  $M$  flight number and number of oblique shocks before the direct one (Fig. 9.5).

Supersonic air intakes are designed on the basis of this work. Designing is carried out for the design mode (usually it is close to maximum flight speed).

### 9.3. Design Model and Design Algorithm of Supersonic Air Intakes

When designing the supersonic air intake, the following problems should be solved:

1. Selection of the number of shocks.
2. Calculation of the inlet area.
3. Shaping the central body.
4. Shaping the shell.
5. Shaping the internal air duct.

Let's consider the design algorithm of the flat air intake with external compression. The initial data for calculation are:  $m_a$  — the air consumption through the engine, kg/s;  $M_H$  —  $M$  flight number;  $H$  — the flight altitude.

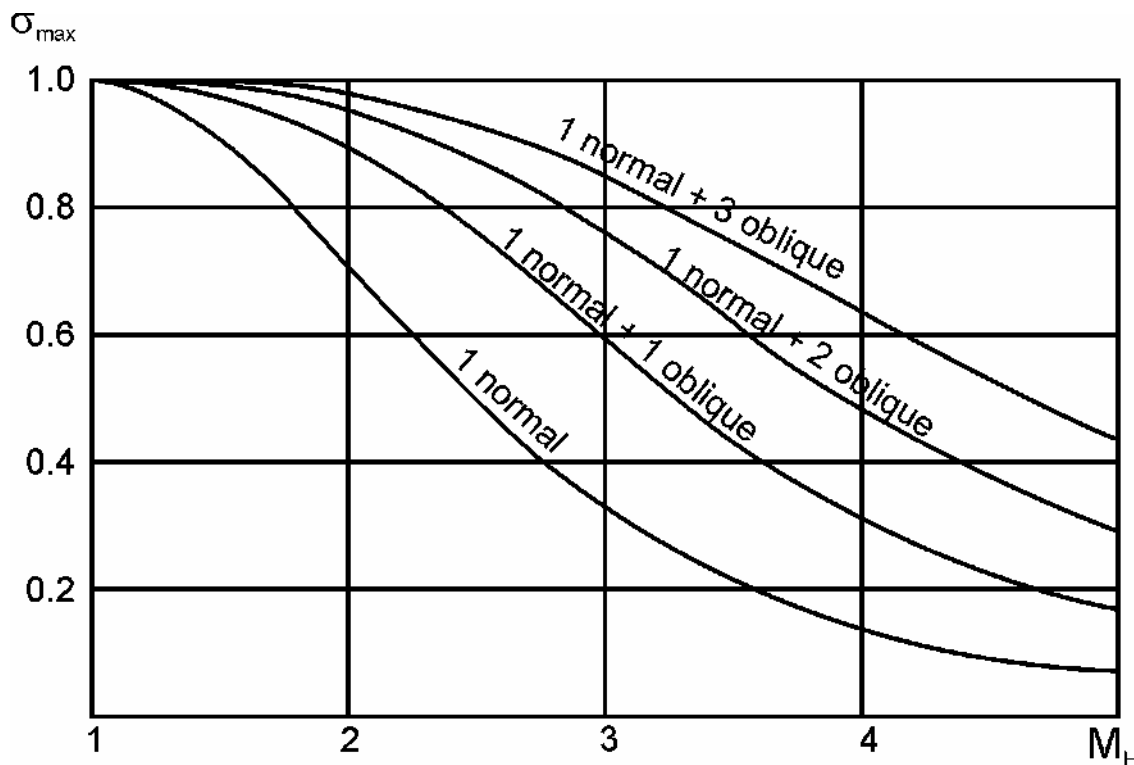


Fig. 9.5. Dependence of Impact Pressure Recovery Factor on  $M$  Flight Number and Number of Oblique Shocks for Optimum Systems of Shocks

It is necessary to note, that the air intake is designed for the specified flight mode, for which its parameters are optimum. At the other flight modes, the impact pressure losses can appear inadmissible, that is the air intake regulating is required. One more reason for regulating the supersonic air intake is complication of reaching of the design mode.

In view of the air intake arrangement, the  $M$  number is determined before it as follows:

- for nose air intakes:  $M_{des} = M_H$ ;
- for under-wing air intakes:  $M_{des} = M_H - (0.04...0.05)$ ;
- for lateral air intakes:  $M_{des} = M_H - 1.05$ .

**Number of shocks  $n$**  is chosen on the base of the graph (Fig. 9.5). The impact pressure recovery factor at each shock is determined as  $n$ -th power root of the impact pressure recovery factor taken from the graph (Fig. 9.5):

$$\sigma_i = \sqrt[n]{\sigma_{max}}$$

**Inlet area  $F_e$**  is determined by the condition when the maximum air consumption at the design flight mode is provided:

$$F_e = \frac{m_a}{\rho_H V_H}$$

where  $\rho_H$  — the air density at the design altitude;  $V_H$  — the flight speed.

For **shaping the central body**, it is necessary to know the stream turn angles at each shock  $\alpha_i$  (Fig. 9.6). These calculations are carried out on the base of the oblique shock theory. The impact pressure recovery factor for the oblique shock is given by the relation:

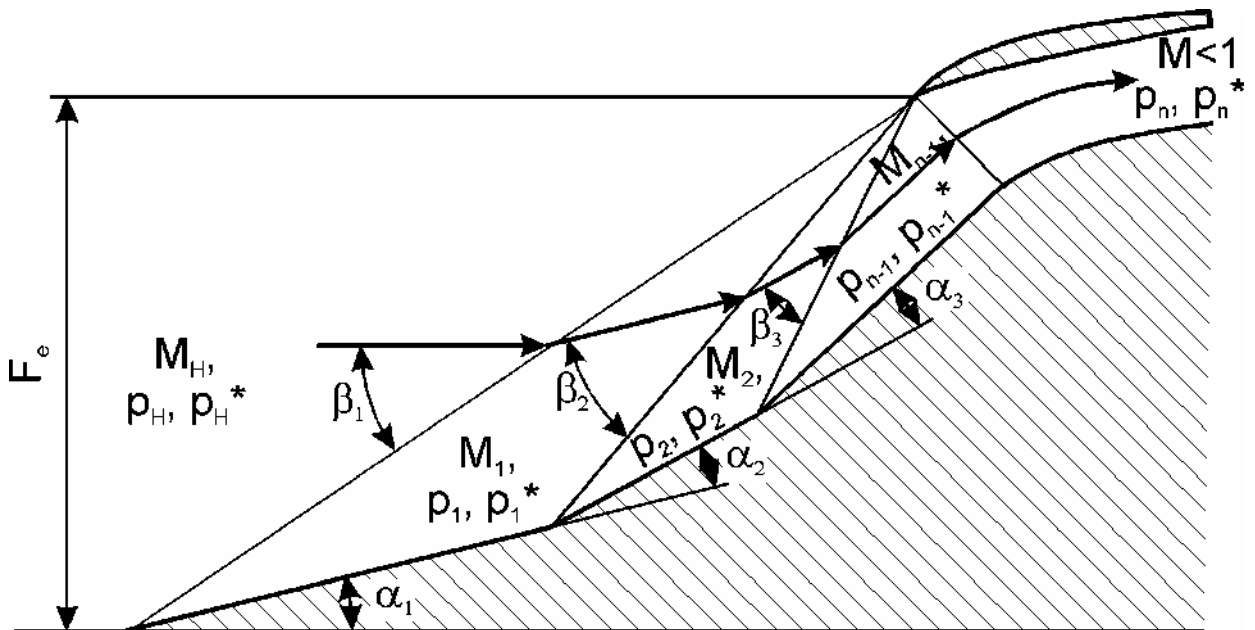


Fig. 9.6. Design Model of Supersonic Air Intake

$$\sigma_i = \left( \frac{k+1}{2} \right)^{\frac{k+1}{k-1}} \left( \frac{M_i^2 \sin^2 \beta_i}{1 + \frac{k-1}{2} M_i^2 \sin^2 \beta_i} \right)^{\frac{k}{k-1}} \frac{1}{\left( k M_i^2 \sin^2 \beta_i - \frac{k-1}{2} \right)^{\frac{1}{k-1}}},$$

where  $k = 1.4$  — ratio of the air thermal capacity at constant pressure to the thermal capacity at constant volume;  $M_i$  — M number before  $i$ -th shock;  $\beta_i$  — the shock front deflection angle from the initial direction of the stream.

Let's denote  $x = M_i^2 \sin^2 \beta_i$ . As the impact pressure recovery factor at  $i$ -th shock is already known, then it is possible to find  $x$  by any numerical method. For example, we can use iteration method from the equation:

$$x = \frac{\sigma_i^{2/7} (1 + 0.2x)(1.4x - 0.2)^{5/7}}{1.2^{12/7}}.$$

Expressing angles  $\beta_i$  by using them, we obtain:

$$\beta_i = \arcsin \frac{\sqrt{x}}{M_i}.$$

Then, with known  $\beta_i$  angle and M number before shock  $M_i$ , the M number behind the shock  $M_{i+1}$  is given by the expression:

$$M_{i+1} = \sqrt{\frac{5 + M_i^2}{7 M_i^2 \sin^2 \beta_i - 1} + \frac{5 M_i^2 \cos^2 \beta_i}{5 + M_i^2 \sin^2 \beta_i}}.$$

From last two equations angles  $\beta_i$  and M numbers behind shocks  $M_{i+1}$  are consistently determined.

Stream turn angles at each shock  $\alpha_i$  are determined by the equation:

$$\alpha_i = \beta_i - \arctg \left[ \left( \frac{1}{6} + \frac{5}{6 M_i^2 \sin^2 \beta_i} \right) \operatorname{tg} \beta_i \right].$$

For axially-symmetrical air intakes, there is no exact solution. Therefore, for their designing, at first the flat air intake is designed, and then we pass to the cone angles using the  $\alpha_{con}(\alpha_{wed})$  diagram. Otherwise, the axially-symmetrical air intake can be designed using nomogram from reference books.

When **shaping the shell** it is necessary to satisfy two conditions: 1) to organize air flow on the inner shell surface, it is necessary to provide the inner shell angle  $\theta$  equal to the total stream turn angle in the shell lip and 2) to create the attached oblique shock on the outer shell surface, it is

necessary to provide the outer shell angle  $\kappa$  less than limit stream turn angle  $\alpha_{lim}$ .

For external compression air intakes, the first condition is formulated as follows: to create closing direct shock between the wedge and the shell, it is necessary to provide the inner shell angle  $\theta$  equal to the total stream turn angle on the wedge  $\alpha_{\Sigma}$ .

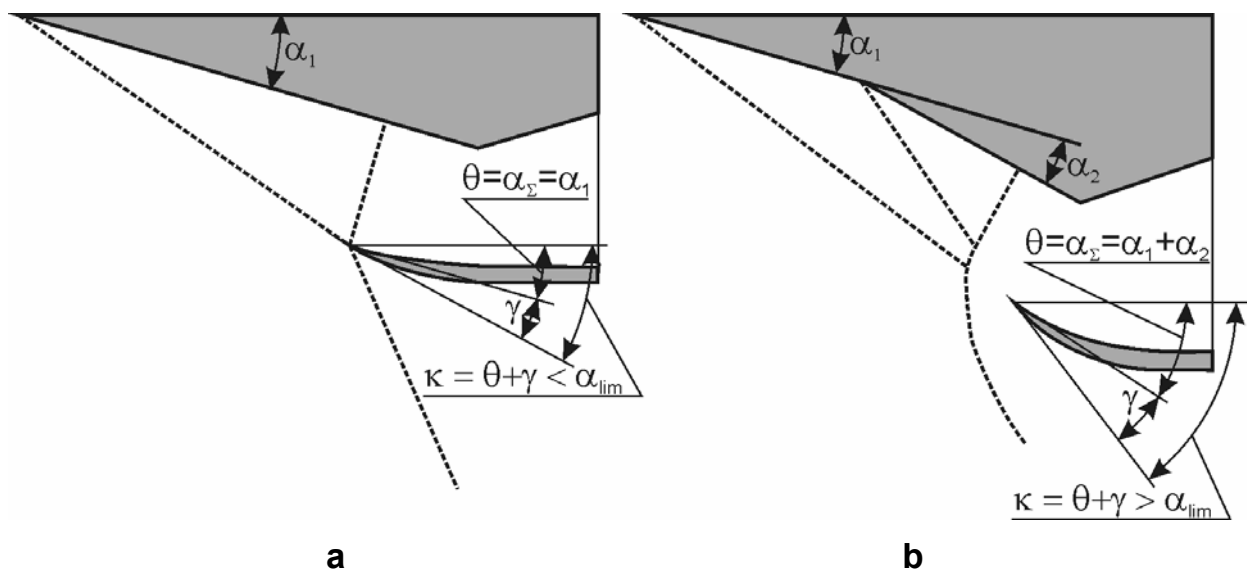
It is easily to satisfy the first condition separately. For this purpose, it is enough to set:

$$\theta = \alpha_{\Sigma}.$$

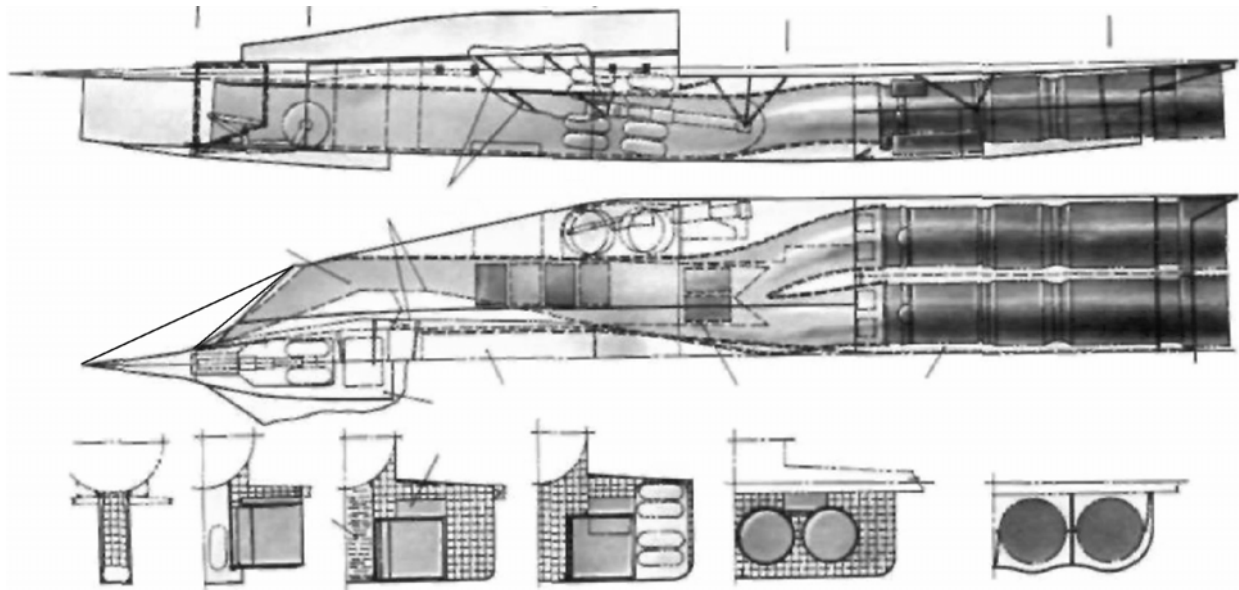
Fig. 9.3 shows, that the total stream turn angle for the optimal shock systems with  $n \leq 3$  is less than limit one. But the outer shell angle  $\kappa = \theta + \gamma$ . And  $\gamma$  angle cannot be less than  $(4...8)^{\circ}$ , by the strength and stiffness conditions. Thus already for  $n = 3$ , we obtain  $\kappa = \alpha_{\Sigma} + \gamma > \alpha_{lim}$  (ref. Fig. 9.3). And the direct (detached) shock is created (Fig. 9.7, b) instead of oblique (attached) one (Fig. 9.7, a). It results in the big impact pressure losses and external drag increase.

Thus, to meet the second requirement with  $n > 2$ , it is appropriate mixed compression air intakes application.

The **internal duct** is shaped in the following way. Behind the direct shock, the internal duct is designed converging (Fig. 9.8). In such converging duct, the subsonic flow is accelerated and becomes supersonic again in the critical section. Further the duct is expanded. In such duct, the supersonic flow is accelerated. After that, the second direct shock is located. The reason of this second direct shock is the counterpressure in the duct in front of the engine. This supersonic airflow area is required to prevent the engine disturbances from affecting the system of shocks and to increase operational stability of the air intake.



**Fig. 9.7. Air Intake Shell Designing**



**Fig. 9.8. Supersonic Air Intake Duct**

At this step, the slot and throat areas are calculated by the gas dynamics formulas. When shaping the duct, these sections are usually connected by a smooth curve of the second order.

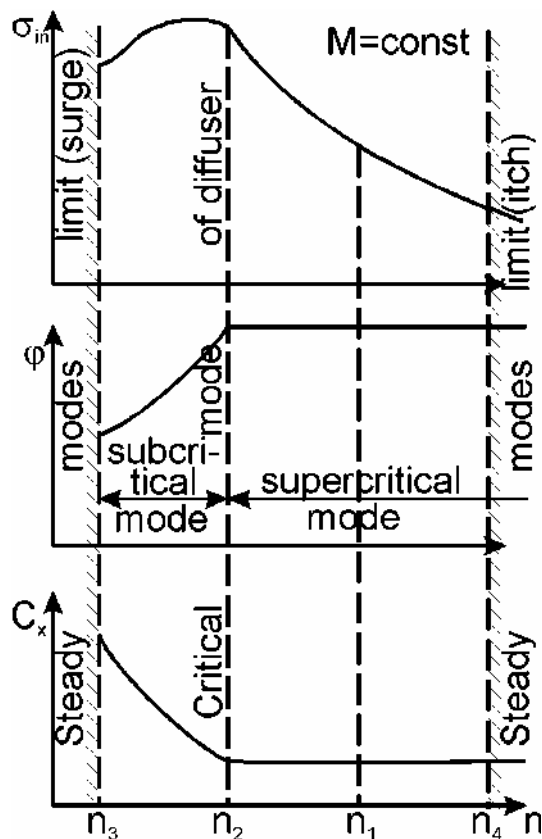
#### **9.4. Operation of Air Intakes at Off-design Conditions**

The air intake operational mode is characterized by the ratio of required  $m_{req}$  and available  $m_{av}$  air consumption, the impact pressure recovery factor  $\sigma_{in}$  (the ratio of the impact pressure behind the air intake to the impact pressure in front of it), the flow coefficient  $\varphi = m_{av} / m_{max}$  (the ratio of the available air consumption  $m_{av}$  to the maximum possible one  $m_{max}$ ) and external drag coefficient  $C_x$  (Fig. 9.9).

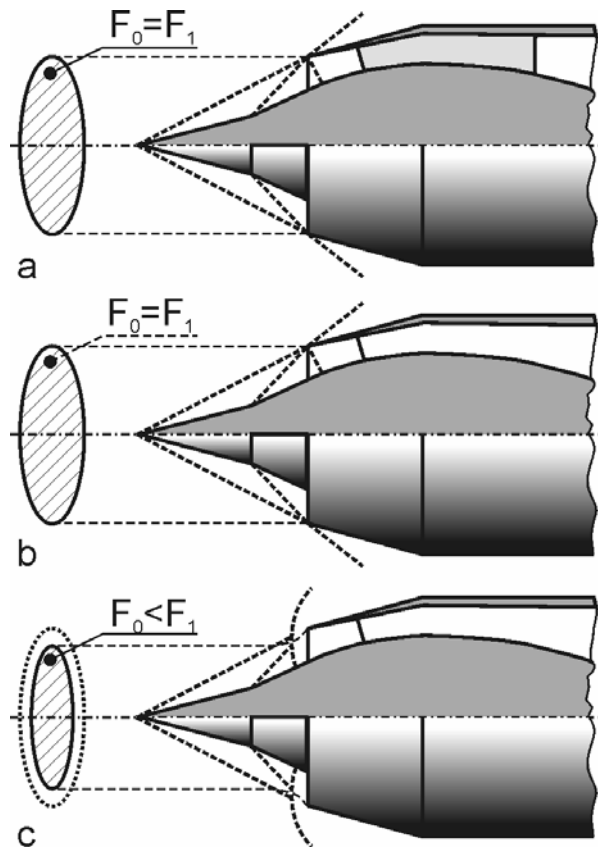
The air intake operational modes can be **steady** and **unsteady**. At steady modes, the flow parameters do not change during the time. Unsteady modes are characterized by the fast variation of flow parameters during the time.

The **critical mode** of operation is a particular case of a steady mode at which the required air consumption is equal to available one (Fig. 9.10, b).

At decreasing  $m_{req}$ , the air intake gets to the **subcritical operational mode** (Fig. 9.10, c). The available air consumption through the air intake decreases due to the stream overflow in front of the shell. In this case, the flow coefficient decreases sharply and becomes less than one. This mode is characterized by the head wave at inlet. The air intake drag coefficient intensively increases. The impact pressure recovery factor variates a little. The air flow is subsonic everywhere inside the air intake duct.



**Fig. 9.9.  $\sigma_{ex}$  (a),  $\phi$  (b) and  $C_x$  (c) Coefficients Dependence on (Bypass) Turbojet Reduced Number of Revolutions**



**Fig. 9.10. Supercritical (a), Critical (b) and Subcritical (c) Operational Modes of Supersonic Air Intake**

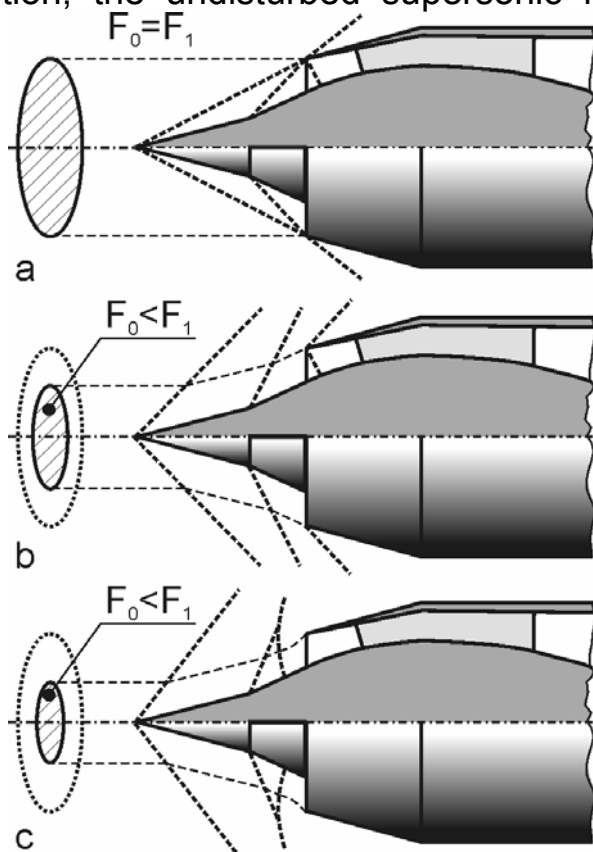
At increasing  $m_{req}$ , the air intake gets to the **supercritical operational mode** (Fig. 9.10, a). In this case, the shock–wave system remains constant; but air pressure before the engine decreases (because of the available air consumption through the air intake cannot farther increase). Matching of air consumptions becomes possible, when the normal shock, which is formed behind the throat, moves downstream. The pressure losses inside the air intake increase. In this case, the air pressure before engine is decreased exactly so much, that required and available air consumption becomes equal. In this case, the impact pressure recovery factor falls, which leads to the engine thrust decrease.

If the **Mach numbers are less than designed one (Fig. 9.11)**, the shock deviation angles  $\beta_i$  increase, and the shocks are not focused in the shell lip. Therefore the flow coefficient decreases, and the additional drag appears. In addition, the ram pressure of the undisturbed airflow sharply decreases, which leads to the pressure and air density decreasing inside the air intake throat. That is, when Mach number decreases, both the shock–wave system air consumption, and throat air consumption de-

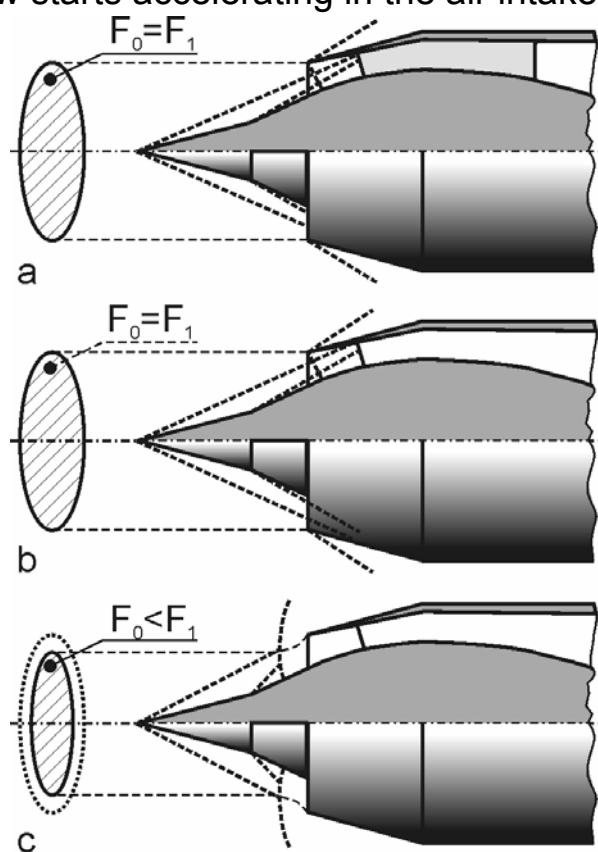
crease simultaneously. But the latter decreases considerably faster. If the throat area is selected on the basis of the design operational mode of the air intake, then if the Mach numbers are less than designed, the throat cannot pass the total air consumption, which passed through the oblique shock-wave system. As a result, the head wave is formed in front of the inlet (Fig. 9.11, c); this head wave provides decreasing of the available air consumption due to air overflow, and the drag coefficient considerably increases.

If the throat area is selected greater, than it is necessary for the design mode, then the head wave formation can be prevented in a narrow range of Mach numbers which are smaller than the design one (Fig. 9.11, b).

If the **Mach numbers are greater than designed one (Fig. 9.12)**, the shock deviation angles  $\beta_i$  decrease. The shocks hit not in the lip of the air intake shell, but on its internal surface, and the airflow inside the duct becomes very complex, as a result of mirroring of these shocks. In addition, the undisturbed supersonic flow starts accelerating in the air intake



**Fig. 9.11. Flow Modes at Mach Number: Equal to Designing (a) and Less than Designing (b — Throat Is More than Required; c — Throat Is Less than Required)**



**Fig. 9.12. Flow Modes at Mach Numbers, Bigger than Designing One: a — Supercritical; b — Critical; c — Subcritical**



part free from oblique shocks, when getting into it. The throat pressure and fields of speed become sharply irregular. The impact pressure losses in the shock–wave system increase due to increase of shocks intensity and specified irregularity.

At the **subcritical operational modes** (Fig. 9.12, c), the head wave is always formed. In this case, with the **Mach number increase**, impact pressure recovery factor decreases, due to shocks intensity increase. The flow coefficient also decreases, as a result of air overflow behind the head wave. The drag coefficient increases due to the formation of the strong head wave.

Let's consider the **unsteady operational modes** of air intakes. The interaction of the boundary layer with the system of shocks results in essential difference between the real system of shocks and designing one, which is obtained without regard for the viscosity and stream separation zones. Thus at the subcritical and supercritical modes, the unsteady oscillations of the flow parameters can appear, due to the unsteady separation of boundary layer. These phenomena are known as a surge and an itch.

The **surge** is a complex self–oscillating process, caused by stalled instability of a boundary layer, resulting in the periodic oscillations and the configuration changing of the shock system. It results in the oscillations of the air consumption and pressure (the oscillation frequency at the surge is usually 5...10 Hz).

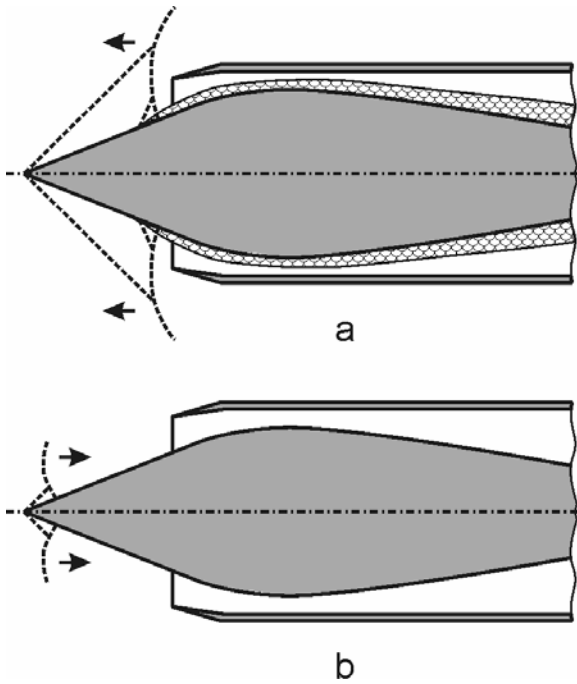
The surge can be provoked by the sharp decreasing of the required air consumption through the engine (Fig. 9.13, a). At the moment of the greatest decreasing of the air consumption, the system of shocks turns in the «knock» head wave located ahead of the air intake (Fig. 9.13, b). Air intake gets to the subcritical mode. After that the air consumption grows. The flow conditions become close to design ones. The system of shocks is rebuilt. The closing shock is placed in zone of possible separation. The boundary layer is restored with the same parameters. Then there is a new separation, and the process is periodically repeated.

At the substantial growth of the required volumetric consumption through the engine the shock moves behind the diffuser throat. The supercritical mode is observed. The closing shock intensity can cause the separation of the boundary layer (Fig. 9.14). This stream separation is accompanied by high–frequency pressure oscillations. This mode is known as an **itch**. (In this case, the oscillation frequency makes 300...400 Hz).

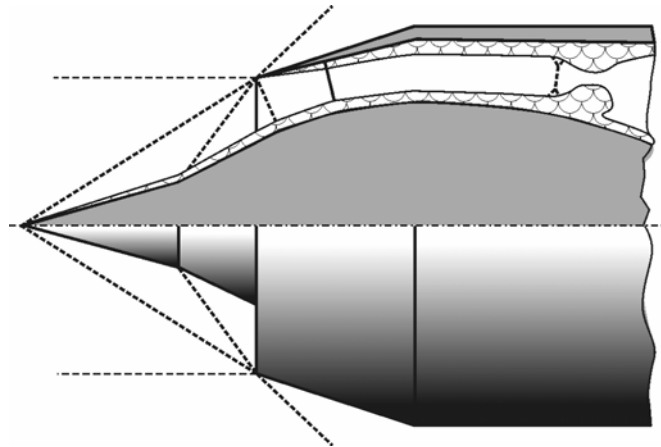
It is necessary to note, that the air intake operation at the unsteady modes is extremely undesirable.

## 9.5. Methods of Supersonic Air Intake Regulation

The main problem concerning to the regulation of the air intakes, when flight speed and the engine power are changed, is how to keep the



**Fig. 9.13. Surge**



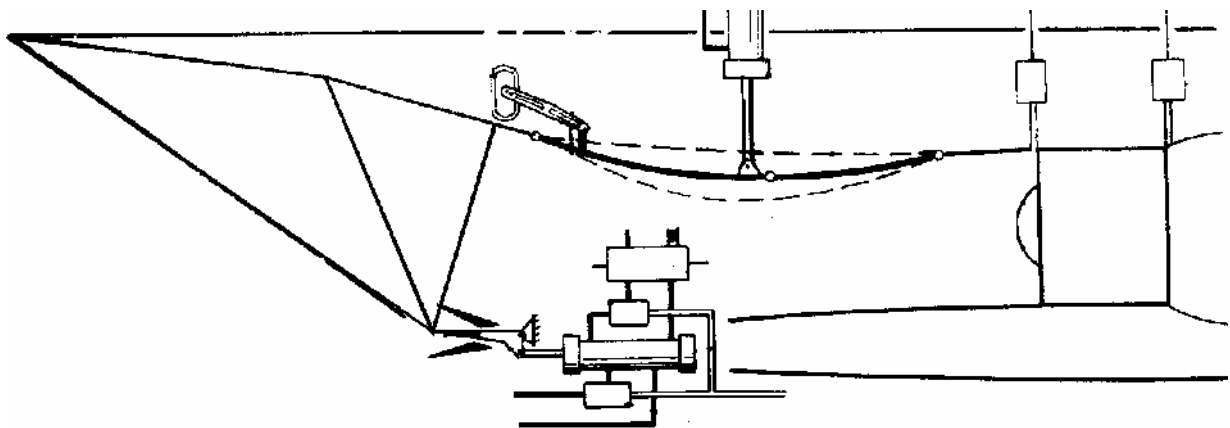
**Fig. 9.14. Itch**

required and available air consumption equal. It provides to save the system of shocks in the specified position.

The regulation can be carried out by:

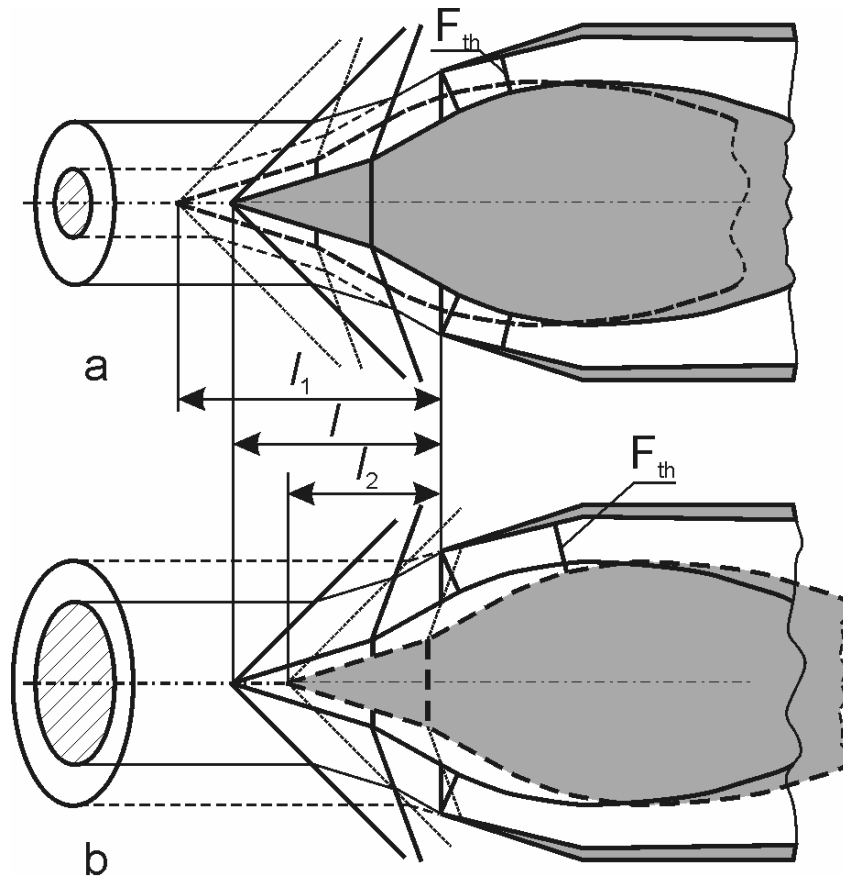
- 1) Changing the inlet area;
- 2) Moving the central body or the shell;
- 3) Changing the throat area;
- 4) Air bleeding through by-pass valves.

**Changing the inlet area** can be provided by movable forward part of the flat air intake shell (Fig. 9.15).



**Fig. 9.15. Air Intake Regulation by Inlet Area Changing (Simultaneously with Throat Area Changing)**

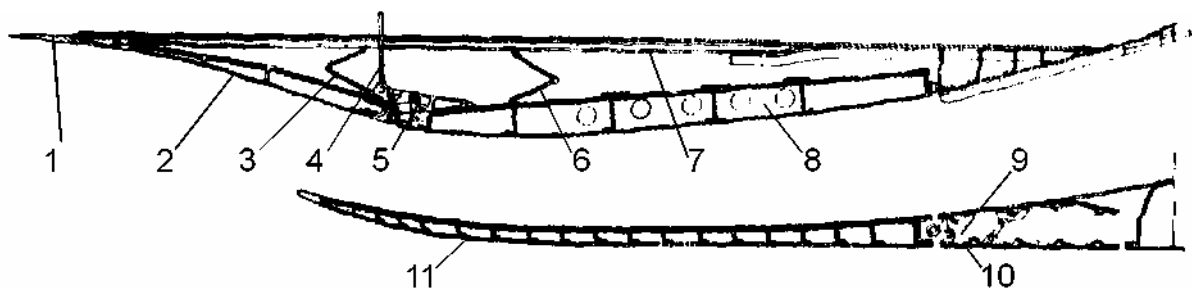
**Moving the central body** allows to change the inlet air consumption (Fig. 9.16). With the flight speed increase, the central body is extended forward. Thus oblique shocks remain focused in the shell lip. At the speed



**Fig. 9.16. Air Intake Regulation by Moving of Central Body:**  
**a — Forward; b — Back**

decrease, the process occurs in the opposite direction. The throat area varies simultaneously. It is achieved by special shaping of the internal shell surface.

The **direct change of the throat area** include the relocation of the mobile panels or shaped inserts (Fig. 9.15, 9.17), the blowing of stream onto the central body surface in the throat zone, changing the central body diameter.



**Fig. 9.17. Air Intake Regulation by Relocation of Mobile Panels:**  
**1 — Forward Fixed Wedge Part; 2 — Forward Shatter; 3 — Forward Cur-**  
**tain; 4 — Actuator; 5 — Link; 6 — Rear Curtain; 7 — Inner Panel; 8 — Rear**  
**Shutter; 9, 10 — Inner and Outer Take-Off Shutters; 11 — Outer Side Panel**

**By-pass valves** (anti-surge doors) also serve as the regulating elements. The available and required air consumption are kept equal by their opening.

Regulating can be continuous or discrete.

### 9.6. General Arrangement of Air Intakes

The design of the air intake is similar to the fuselage design. It is necessary to make only a few remarks.

1. Flat walls should be avoided. Such walls are strongly loaded because of the high pressure difference. If necessary, they must be reinforced by frames, embossings, sections.

2. All riveted joints should be sealed.

3. All detachable joints must be reliably locked.

4. It is necessary to provide controllability in service for the condition of the air intake internal surface.

5. It is necessary to take into account the impulse loadings appearing during surge. Otherwise these loadings can cause the air intake damage.

6. It is necessary to avoid the resonance oscillations of structural elements during surge.

Structural materials for supersonic air intakes are practically the same materials, as for subsonic ones. However titanium alloys and sometimes steel are more widely used here, because of small element thicknesses and big loadings.

Providing high efficiency and reliability of power plant makes requirements to the **air intake arrangement onboard an airplane**. That is, it is necessary to take into account interference of the air intake and aircraft, air intake and engine. First of all, the air intake inlet conditions are the most important: the local speed of stream, the speeds non-uniformity and the local downwash angles.

When the air intake is arranged in a **braking zone**, it is possible to increase the impact pressure recovery factor and reduce the inlet area and drag.

The increase of **speeds non-uniformity** results in decreasing of the impact pressure recovery factor and decreasing of the surge margin.

The large **local downwash angles** (caused, for example, by the fuselage) contort the inlet theoretical conditions and disturb the design operation mode.

**Flat and semicircular** air intakes **attached to the fuselage** are better integrated to it, structurally are easily integrated in sheaves of two or four air intakes, provide the greater regulating range of the flow area and have better characteristics at the large angles of attack.

The advantage of the **axially-symmetrical** air intake consists in the smaller mass and length, but they are difficult to be regulated in wide consumption range.

Flat air intakes, in comparison with the axially-symmetrical ones, are usually characterized by the large change of the optimal wedge angle, than the change of the central cone angle, depending on the M flight number. It results in the greater range of deviations in the actuation system and increases its mass.

In view of above stated, **three types of air intakes** are widely used in the modern aviation: nose, attached to fuselage (lateral, ventral, dorsal) and wing (under-wing, over-wing).

**Nose air intakes**, round or oval cross-section, are used in the case of the arrangement of engines in fuselage or engine nacelle.

[+] Other airplane units practically do not affect to their operation.

[-] They have long and complicated air duct (the mass of 1 m<sup>2</sup> of the supersonic air intake duct surface makes approximately 8...12 kg).

[-] This air duct occupies the payload volume of the fuselage and makes the fuel storage as well as the arrangement of the landing gear and other equipment more complicated.

The last disadvantage is eliminated if the engine is located in the nacelle.

The cross-section of **lateral air intakes attached to fuselage** can be of various shapes (rectangular with rounded corners, semicircular, semi-elliptical, a quarter of a circle, etc.).

[+] The air intakes location on each side of the fuselage shortens the air duct, reduces the hydraulic pressure losses, and frees place for the radar equipment.

[-] One of them is usually shaded by the airplane during the slip maneuver;

[-] The boundary layer, formed on the fuselage, can affect the air intake operation.

**Dorsal and ventral air intakes** are a variation of the lateral ones and they are characterized by the reduced compression efficiency at the large angles of attack, because of the fuselage impact.

**Under-wing and wing flat air intakes** are most widely used nowadays on the heavy supersonic aircraft, despite of the great effect of engines to flow about the wing and cause a decline of its characteristics.

In this case two versions of arranging the braking surface are probable: horizontal and vertical.

The airplanes with horizontal arrangement of the braking surface (Tu-144, Concorde), are characterized by the increase of the aircraft lift with the increase of the angle of attack and higher lift-to-drag ratio. These air

intakes are less sensitive to the change of the angles of attack.

Air intakes with a vertical wedge (XB-70A, T-100) have essentially smaller (approximately 20%) length, due to the equalization of the inlet field of speed. They are less sensitive to lateral washes.

The wing and under-wing arrangement of the air intakes is expedient in the zones of minimal washes, that is as closer as possible to the aircraft plane of symmetry. Thus, it is necessary to provide the operation stability of the neighboring air intakes if the surge occurs in one of them.

## Theme 10. EXHAUST UNITS OF POWER PLANTS

### 10.1. Exhaust Units, Their Purpose, Requirements and Structure

Exhaust units of power plants are **intended** for transformation of the thermal and potential energy of gases into kinetic energy of outflowing stream.

**Requirements** to exhaust units are:

1. Minimal losses of effective thrust and high efficiency of power plant;

2. Operation reliability;

3. Simple design;

4. Minimal mass;

5. Minimal external drag.

Exhaust units can **include**:

1. Jet nozzle;

2. Extension pipes for gas passing to nozzle;

3. Thrust reverse device;

4. Thrust vectoring device;

5. Air-supply system for the structural elements cooling;

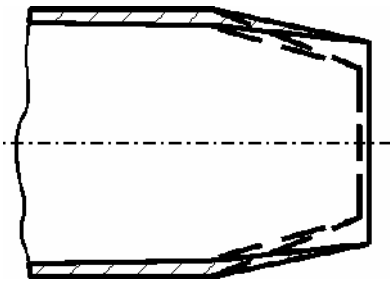
6. Noise abatement devices.

Depending on the exhaust velocity, jet nozzles are divided into subsonic and supersonic.

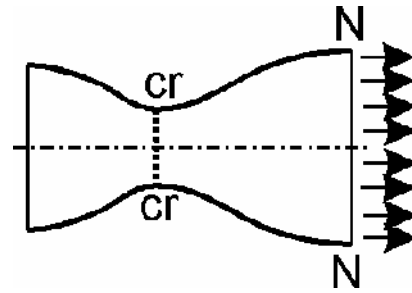
For subsonic and small supersonic flight speed, converging nozzles are used (Fig. 10.1). These nozzles are simple in design and have a small mass.

For supersonic flight speed, Laval nozzles are applied (Fig. 10.2). In the critical section of a Laval nozzle, speed reaches the local sound speed. In the divergent section, the gas pressure further drops and the exhaust speed grows. As is known from gas dynamics, the thrust force of jet engine is calculated by the following formula

$$F = m_g(V_N - V_\infty) + S_N(p_N - p_\infty),$$



**Fig. 10.1. Subsonic Nozzle**

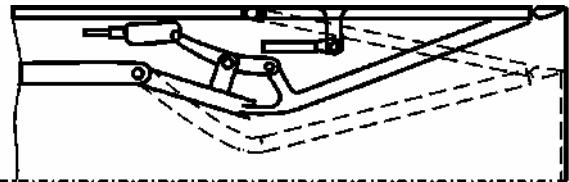


**Fig. 10.2. Laval Nozzle**

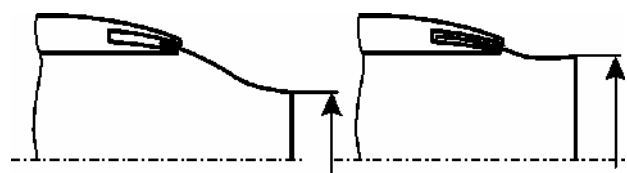
where  $m_g$  — gas consumption at the exhaust nozzle exit (it is equal to the sum of the air and fuel consumption);  $V_N$ ,  $V_\infty$  — speed of gas in the exhaust nozzle exit and the aircraft flight speed;  $S_N$  — nozzle cross-sectional area;  $p_N$ ,  $p_\infty$  — gas pressure in the exhaust nozzle exit and the air pressure at the flight altitude.

It is necessary to note, that the Laval nozzle is designed for the specified M flight number. And the off-design operation of the nozzle causes considerable losses. Therefore, fully-variable variable-area Laval nozzles are applied. Fig. 10.3 shows the scheme of variable-area Laval nozzle.

A lot of up-to-date multimode airplanes execute most of the task at transonic flight speed. Here during afterburning, it is necessary to increase the area of critical section and the exhaust nozzle exit in 1.5...2 times. For such airplanes it is sometimes, expedient to use «iris» nozzles (Fig. 10.4).



**Fig. 10.3. Variable-Area Laval Nozzle**



**a**  
**b**  
**Fig. 10.4. «Iris» Nozzle**

This is a variation of a Laval nozzle. The position of shutters for maximum mode (a) and for power argumentation (b) is shown in Fig. 10.4. The kinematics of the «iris» nozzle shutters provides obtaining the converging nozzle in non-afterburning modes, with the shutters moving back. At the supersonic flight speed, the «iris» nozzle has considerable losses owing to the gas underexpansion.

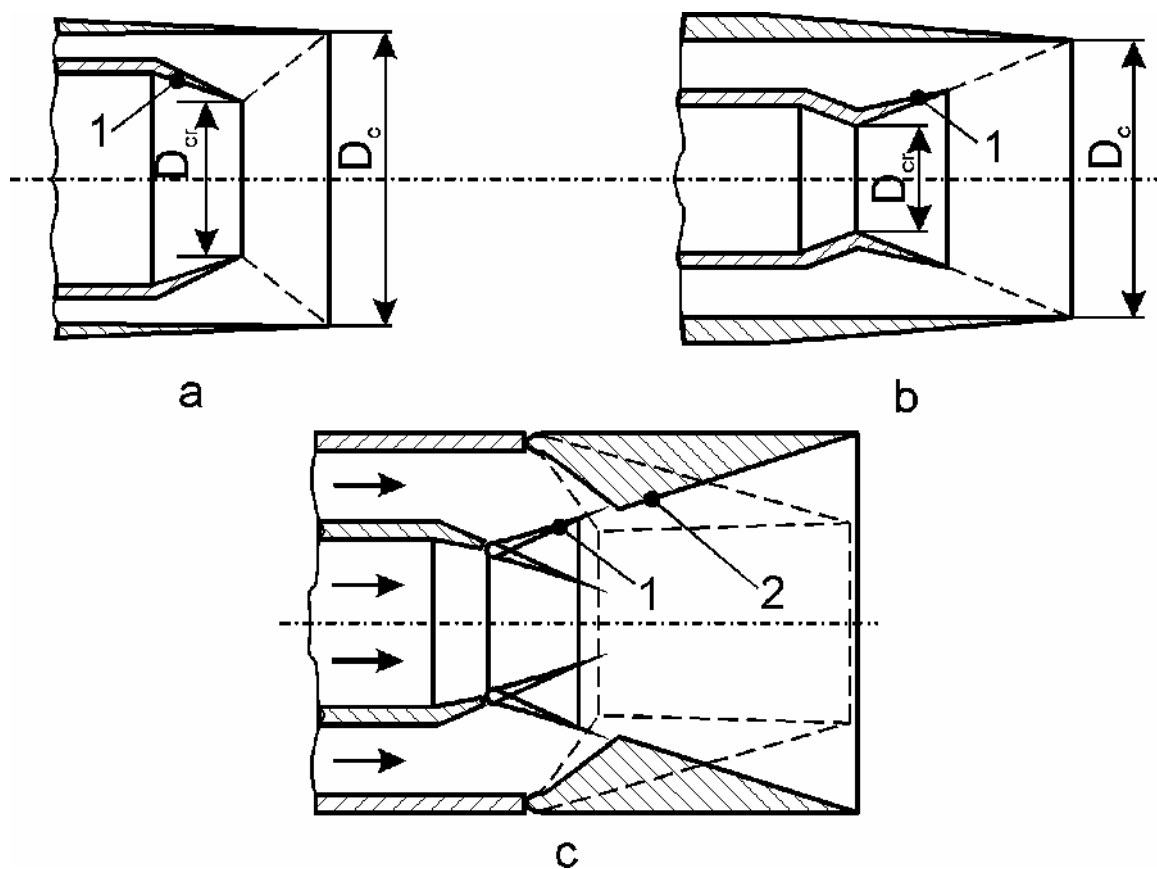
Fully-variable variable-area Laval nozzles are very complicated in design. The ejector nozzles shown in Fig. 10.5 are simpler. The ejector nozzles have the supersonic section of the nozzle in full or in part replaced by boundary line of free supersonic stream. Thus supply of induced air in

the nozzle is carried out. Ejector nozzles consist of the primary nozzle 1 and shell 2. Turned shutters can also be placed on the shell. Regulating of ejector nozzles is easier, than Laval's ones.

## 10.2. Thrust Reverse Devices, Their Purpose and Requirements. Types of Thrust Reverse Devices

Up-to-date airplanes require long runways. The landing run of airplane can be essentially reduced by means of thrust reverse. That is by veering of exhaust stream to  $100...180^\circ$ . Thrust reverse devices have the widest application in transport and passenger aviation.

Thrust reverse devices are **intended** for decreasing of landing run at landing, and also for increasing maneuverability at ground.



**Fig. 10.5. Ejector Nozzles**

The following **requirements** are made to the thrust reverse devices.

1. When the reverse is actuated the negative thrust should be not less than  $35...45\%$  of the thrust developed by the engine in stand conditions.

2. The reverse should be switched on/off in one–two seconds.

3. Keeping constant engine power when reversing.

4. Switching on the reverse device should not impair to stability and



controllability of airplane.

5. The exhaust stream should not cause overheating of airplane surfaces, when hitting them.

6. The thrust asymmetry should be avoided during thrust reverse actuation when the airplane has several engines.

7. The thrust reverse device should not affect the operation of power plant air intakes.

8. Increase of engine nacelle drag due to installation of reverse should be minimal.

9. Noise should not exceed permissible limits at reverse switched on.

10. The design of thrust reverse device should be simple, reliable, convenient and safety in service, have minimal mass and overall dimensions.

Nowadays scoop and turning vane thrust reverse devices are widely used. On bypass turbojets without mixing, these devices can be installed in both contours. Fig. 10.6 shows the schemes of typical designs of thrust reverse device: a — a turning vane device and b — a shuttered type device. The thrust reverse device applied to a bypass turbojet without mixing is shown in Fig. 10.6, c.

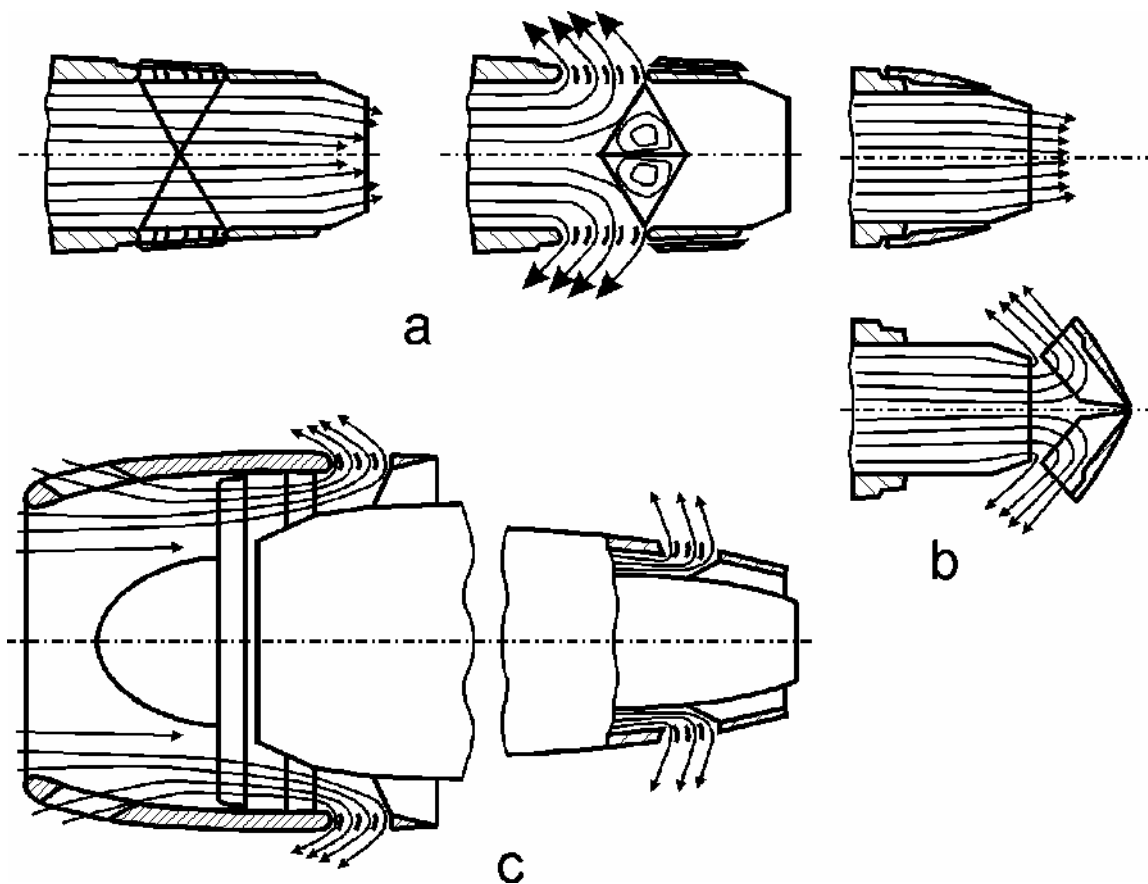


Fig. 10.6. Schemes of Thrust Reverse Devices

### 10.3. Aircraft Noise Sources. Noise Abatement Procedures

Power plant is the main source of noise onboard an airplane. On airplanes with (bypass) turbojets, the noise is produced by a jet stream and compressor. At turboprop airplanes it is produced by jet stream and a propeller. The higher the bypass ratio is, the lower the gas exhaust speed becomes. The amount of noise reduces accordingly.

The compressor, fan and turbine noise is caused by the interaction of the rotor blades and guide vanes. In the compressor noise, the discrete frequencies dominate. They are perceived as a howl or squeal when engines operate at small throttle at landing.

The turbine noise is limited and is partially masked with noises produced by the jet stream. The main components of the propeller noise are the rotational and whirlwind noises. First is caused by periodic effect of propeller blades on the environment. Second is generated by the vortex sheet, which is formed behind blades by outflow separation.

Noise abatement is carried out by the following ways.

1. Modification of existing and creation of new quiet engines.
2. Rational arrangement of power plants onboard the airplane.
3. Increasing of lift-to-drag ratio of airplane.
4. Using of the optimum take-off and landing modes.

**Design measures of noise abatement** are:

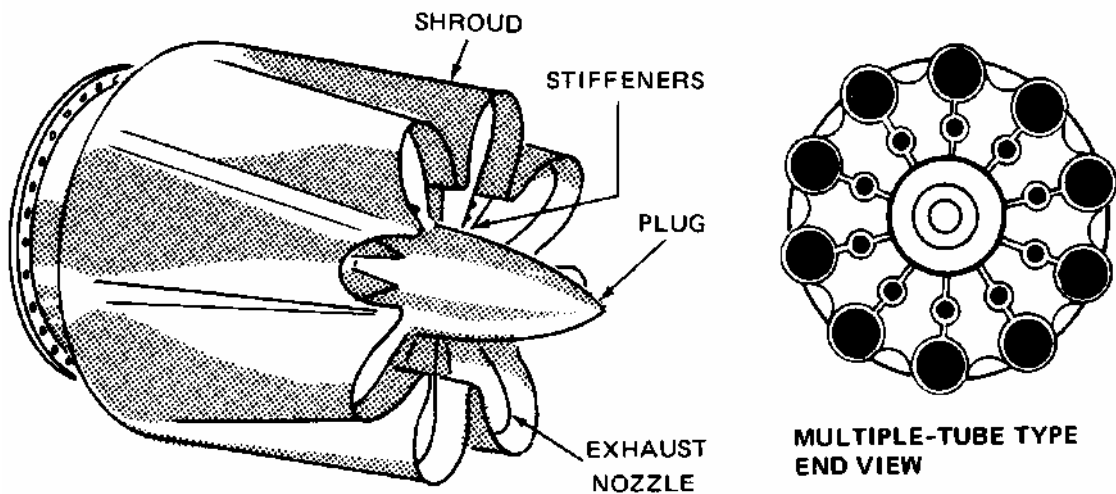
- 1) The application of the noise absorbing nozzles (for turbojets and bypass turbojets with small bypass ratio) (Fig. 10.7);
- 2) The application of the devices designed for mixing the first and second gas contours (for bypass turbojets with bypass ratio less than 4);
- 3) The decrease of the number of fan stages;
- 4) The removal of the inlet guide vanes in order to eliminate the interaction between the vane trailing edges and the rotor blades;
- 5) The optimization of the rotor and stator blades number (number of stator blades should exceed number of rotor blades not less than twice);
- 6) Regulating the rotor blade angles;
- 7) Changing the inlet configuration (installation of compressor bleed ports in front of and behind the fan and installation of lamp-shaped body along duct axis);
- 8) The regulating of jet nozzles;
- 9) The provision of the air bleeding between contours;
- 10) Covering the surface of turbojet ducts with the sound-absorbing materials.

The most effective method of the noise abatement is to reduce the exhaust speed. However, if the bypass ratio becomes higher than 7, the noise abatement is not further obtained. Thus raise of fan noise takes

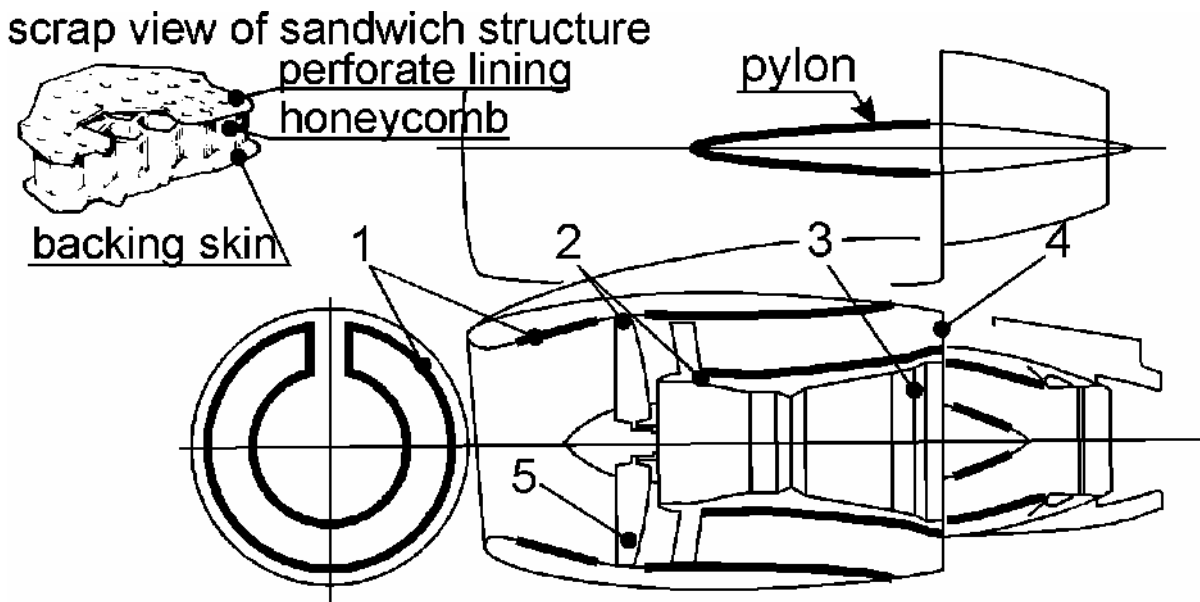
place.

The further noise abatement can be achieved by the acoustical covering of the nacelle and the extension of the air intake and nozzle (Fig. 10.8).

It is necessary to keep in mind, that design measures intended for noise abatement result in increase of structural mass and growth of the engine nacelle drag, and also reduce the engine thrust. Therefore, it is necessary to apply compromise settlement to economically sound application of these measures.



**Fig. 10.7. Noise Suppression Unit in Turbojet Nozzle**



**Fig. 10.8. Ways of Noise Abatement for Bypass Turbojet:**

- 1 — acoustical covering of elements of power plant (thick lines);
- 2 — optimum clearances; 3 — optimum number and blade configuration;
- 4 — optimum nozzle position for decreasing exhaust velocity; 5 — fan without inlet guide vanes

## **Theme 11. AIRCRAFT POWER PLANT OUTLOOK**

Innovative design concepts in the new generations of airplanes and helicopters have a profound effect on the development of aircraft power plants. The success or failure of a new aircraft design often depends on the ability of its power plant to meet some specific requirements, while conforming to Aviation Regulation [3].

We have decided to analyse all basic categories of modern AC: strategic bombers, front-line bombers, attack aircraft, fighters, cargo and passenger airplanes. Our analysis covers the current state and prospects for each AC category. Although the focus is on design features and trends in the development of aircraft power plants, those are viewed from a broader perspective that takes into account the current evolution in AC design concepts as a whole. By way of illustration, we mostly use the US aircraft; however, one should keep in mind that practically all of them have Soviet (later — Russian) analogs.

Our attention to military AC is only partially caused by their importance for every nation's defence; more importantly, defence industry creates manifold engineering solutions which are later applied to civil transport. For example, an anti-explosion system designed to protect the fuel tanks of the Republic F-105 Thunderchief fighter-bomber is used in the fire protection systems of Boeing 707 and Boeing 727 passenger airliners.

### **11.1. Aircraft Classification**

For the purposes of our further discussion, it is essential to define the terms to be used. There are different ways of AC classification, e.g.:

- according to the power plant design features (engine type, engine number, etc.);
- according to the AC function (Fig. 11.1); this type of classification is best suited to the purpose of analysing power plant design concepts.

The AC function is interpreted as a system of operations that the aircraft is designed to perform and for the fulfilment of which it is optimized. Later in the aircraft service life, other versions of the basic AC model may be designed and other functions added. However, to study the development of design concepts, we will concentrate on the initial AC function, which determined the optimisation of its power plant.

First of all, AC are divided into two large groups — military and civil aircraft. Military AC are subdivided into strike aircraft, fighters and a multifarious AC group named military special-purpose aircraft (tankers, reconnaissance planes, aircraft for electromagnetic countermeasures, etc). Civil AC are subdivided into transport aircraft (cargo and passenger) and civil

AIRCRAFT	Military aircraft	Strike aircraft	Strategic (or long-range) aircraft		
			Front-line (or tactical) aircraft		
			Attack aircraft		
		Fighter	Front-line fighter		
			Fighter-interceptor		
			Patrol fighter-interceptor		
			Escort Fighter		
		Military special-purpose aircraft (examples)	Air refueller		
			Reconnaissance aircraft		
			Electronic countermeasures aircraft		
			Early-warning aircraft		
		Civil aircraft	Cargo aircraft	Super-heavy (strategical) aircraft	
	Heavy (strategical) aircraft				
	Medium-payload or medium-capacity (tactical) airplanes				
	Light (operational) airplanes				
	Transport aircraft		Cargo-passenger aircraft, Cargo-passenger convertible aircraft		
			Passenger aircraft	Trunk-route airliners	Long-range trunk-route airliners (more than 6000 km)
					Medium-range trunk-route airliners (2500–6000 km);
					Short-range trunk-route airliners (1000–2500 km);
			Airplanes of local airlines (up to 1000 km) Regional passenger airplanes (500-1500 km)		
General aviation (passenger capacity up to 30 persons)					
Civil special-purpose aircraft (examples)	Fire extinguishing aircraft				
	Ambulance airplanes				
	Crane helicopters				

**Fig. 11.1. Aircraft Classification according to function**

special-purpose aircraft (ambulance planes, planes for fire-prevention, ice patrol, etc). Brief characteristics of these AC classes are as follows.

Firstly, AC intended for ground and above-water target destruction were named **bombers** because their main armaments were bombs. Later torpedo carriers, anti-submarine AC, missile carriers, etc. were designed for this purpose. Sometimes the term **strike aircraft** is used to refer to all these AC types. Many AC-design specialists continue to use the term «bombers», implying that modern bombers can be armed with bombs, torpedoes, nuclear bombs, thermonuclear bombs, missiles, cruise missiles, etc.

Bombers, or strike aircraft, are subdivided into **strategic** (or long-range) aircraft (Table 11.1), **front-line** (or tactical) and **attack** aircraft (Table 11.2).

They are intended to destroy high-priority targets located far from the front line. Usually those are objects that provide long-time support to military operations or do serious damage to the operator's forces. The USSR-USA Strategic Armament Limitations Treaty (SALT) gives the following definition of the medium range strategic weapons: these weapons are used to destroy objects located at 1,000–5,500 km from the front line; by contrast, strategic intercontinental weapons are used to destroy objects at distances of more than 5,500 km. Thus, a strategic attack aircraft should have a range of more than 11,000 km (taking into account the possibility of in-flight refueling on condition that the AC power plant is accordingly equipped). However, since air refueling above the enemy territory is impossible, the AC has to fly this route segment without refueling.

**Attack aircraft** (strafers) are intended to destroy small-size movable ground (over-water) targets and troops, both on the battlefield and in the rear, by bombs, missiles and artillery guns. Attack aircraft are designed for operation in the air over the battlefield and in the army<sup>1</sup> rear to support the land and naval forces from the air. The AC purpose determines a set of requirements formed for this category of aircraft:

1) ability to be used against small-size (point) movable targets (usually tanks) and high-level precision of aim. It follows that the AC should fly at low altitude («low-altitude flight»). As any kind of weapons may be used against the AC, from personal guns and sub-machine-guns to anti-aircraft artillery and anti-aircraft missile systems (both stationary and portable), it has to have strong armoring, high level of invulnerability and viability.

2) fast response, i.e. the AC ability to get to the battlefield soon after it receives a «call», which means that the aircraft may be based on natural-surfaced airfields with short runways.

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<sup>1</sup> Here the term "army" is used to designate a group of forces consisting of several divisions that operate jointly.

Table 11.1

## Strategic Bombers and their alternatives

Aircraft type	Aircraft						Engine						
	Aircraft design bureau (Aircraft manufacturer)	First flight date/ Total number	AC length, m	Wing span, m	Maximal take-off mass, t	AC empty mass, t /Maximal payload, t	Max. range, km / Full load range, km	Maximal speed, km/h / M number	Engine number x engine type	Engine model / Engine design bureau (Engine manufacturer)	One engine thrust, / in afterburner, kN	Engine mass, t	Length / diameter, m
B-52H	Boeing	1960 102	48	56.4	222	78.6 30	12070	960	8x bypass	TF33-P-3 Pratt & Whitney	76.5	1.7	3.43 1.35
B-58	Convair	1956 86	29.5	17.3	73.9	33.6 10	3860	2230	4xturbojet, afterburner	J79-GE-5B General Electric	44.5 65.8	1.65	5.14 0.97
XB-70	North American	1964 2	57.6	32	250	108 12.2	10200 *	3200	6xturbojet, afterburner	YJ93-GE-3 General Electric	138	2.36	6.02 1.33
FB-111	General Dynamics	1967 76	23.0	21.3 10.3	53.9	21.6 17	6000 1900	2330	2xbypass, afterburner	TF 30-P-412A Pratt & Whitney	93	1.8	5.99 1.29
B-1B	Rockwell	1984 100	44.8	23.8 41.7	216.4	87.1 34	10400	1.25	4xturbojet, afterburner	F101-GE-100 General Electric	133	1.814	4.6 1.4
pilotless AGM-86	Boeing	1982 3000	6.3	3.66	1.45	0.12	2400	0.8	1x bypass	F107-WR-100 Williams Research	2.7	0.064	0.94 0.31
Kh-55	MKB Raduga	1982 N/A	6	3.1	1.25	0.41	2500	840	1x bypass	R95-300 Soyuz AMNTK	3.5	0.095	0.85 0.32
B-2	Northrop	1989 21	21	52.4	168	45.4 22.6	12230	760	1xbypass	F118-GE-100 General Electric	84.4	N/A	N/A N/A
pilotless X-45A	Boeing+ DARPA	2002 2	8.08	10.3	5.53	0.4	920	850	1xbypass	F124-GA-100 Honeywell	28	N/A	N/A N/A
pilotless X-47B	Northrop Grumman	2009 —	11.6	18.9	20.25	1.8	>2500	890 *	1xbypass	F100 PW-220U Pratt & Whitney	72.6	1.12	3.23 1.16

Refer notes p. 152

Table 11.2

## Front-Line Bombers, Attack Aircraft and their alternatives

Aircraft type	Aircraft										Engine			
	Aircraft design bureau (Aircraft manufacturer)	First flight date/ Total number	AC length, m	Wing span (rotor diameter, m)	Maximal take-off mass, t	AC empty mass, t /Maximal payload, t	Max. range, km / Full load range, km	Maximal speed, km/h	Engine number x engine type	Engine model / Engine design bureau (Engine manufacturer)	One engine thrust, kN (power, kW)	Engine mass, t	Length, m diameter, m	
IL-28	Ilyushin AK	1948 6316	17.6	21.4	23.4	12.9 3	2400	900	2×turbojet	VK-1 Klimov corp.	26.5	0.87	2.64 1.27	
A-5	North American	1961 79	22.3	16.1	28.1	1.8 3.6	3700	2228	2×turbojet afterburner	J-79-GE-8 General Electric	48.5 75.6	1.67	5.3 0.97	
A-10	Fairchild Republic	1976 713	16.3	17.5	21.4	10.5 7.2	930	722	2×bypass	TF34-GE-100 General Electric	41.1	0.68	2.54 1.33	
helicop. AH-64	Hughes Helicopter	1984 1086+	14.7	14.6	9.4	4.66 0.8	460	287	2×turbo-shafts	T700-GE-700 General Electric	1130	0.19	1.17 0.58	
helicop. RAH-66	Boeing Hel. + Sikorsky	1995 2	13.2	11.9	7.6	4.2 2.3	556 222	315	2×turbo-shafts	T800-LHT-801 LHTEC	1165	0.15	0.84 0.55	
Harrier GR.1	Hawker Siddeley	1967 78	14.5	7.7	12	11.7 2.6	370	1160	1×bypass	Pegasus Mk 103 Rolls-Royce	97.5	1.39	2.51 1.22	
Yak-38	Yakovlev OKB	1970 193	15.5	7.32	11.3	7.48 2	680 240	1042	1×turbojet 2×turbojet	R27-V-300 Soyuz + 2×RD36-35FV Rybinsk	61 31	1.35	3.71 1.01	
Yak-141 (fighter)	Yakovlev OKB	1987 2	18.3	10.1	19.5	11.7 2.6	2100 690	1850	1×bypass 2×turbojet	RD-79V-300 Soyuz + 2×RD-41 Rybinsk	153 2×40	2.75	5.23 1.72	
F-117	Lockheed	1981 59	20.1	13.2	23.8	13.4 2.2	1450	933	2×bypass	F 404-400 General Electric	49.9	1.04	4.03 0.88	
pilotless Taifun	STN→ATLAS Elektronik	1999	2.08	2.26	0.15	0.05	250	270	1×piston	N/A	33	N/A	N/A	

Refer notes p. 152



### NOTES to Tables 11.1 – 11.5:

1. When compiling tables the public data from various authoritative sources were used, therefore some inconsistencies and inaccuracies in aircraft parameters (especially for future-technology vehicles) can take place.
  2. The character \* is used to mark design variables been not validated during flight test.
  3. In case two different parameters (one under another) are shown in cells of table column for the aircraft involved, the parameter names are given in the column head using / character as delimiter. In case one from two parameters according to the aircraft type is shown in the table column (for example, engine power for aircraft equipped with piston and turboprop engines and thrust for aircraft equipped with turbo-jet and dual-flow turbojet engines), the name of parameter of less prevalence is given in parentheses in the column head.
  4. In case there is no information about the parameter involved (for example, it is a confidential data of design company), the «N/A» (NO DATA) characters are shown in the corresponding cell of the table. In case the parameter is nonsense for aircraft, the dash ( — ) is shown in the corresponding cell of the table.
  5. .When showing the total amount of aircraft produced (in the third column), the character + is used to mark aircraft been in production to present day.
  6. As times goes the firm names are changed. The modern firm names are used both in the text and in tables.
  7. The brief names of aircraft-building and engine building firms are used both in the text and in tables. The corresponding full firm names are given in the Appendix.
- 

3) ability to hit numerous and various targets, which means that the AC has to have its own high-power artillery guns, small-caliber bombs (usually bomb cartridges), numerous and various arms usually located on external suspension brackets.

**Front-line or tactical bombers** (strike aircraft) are intended to act against enemy troops and to support own troops during tactical and operative-tactical actions on the enemy territory (in the enemy rear), i.e. to fulfil combat missions in the interests of the front (several armies operating jointly).

Attack aircraft are intended to destroy the following standard targets: tanks, armoured troop-carriers, enemy army units, fortified defence objects, artillery and anti-aircraft batteries, motor torpedo-boats and missile boats. Front-line bombers aim the following standard targets: moving tank groups, headquarters, launchers of tactical rockets, fuel stores, ammunition depots, railway stations, front airfields, etc.

The development of these two AC categories was parallel and similar.

**Fighters** (Table 11.3) are operational AC intended for shooting down man-controlled and pilotless aircraft. In this classical definition the main mission of fighters is stated; fighters are optimized to execute this mission at the highest level of efficiency. But modern fighters, especially if

Table 11.3

## Fighters Examples

Aircraft type	Aircraft							Engine					
	Aircraft design bureau (Aircraft manufacturer)	First flight date/ Total number	AC length, m, m	Wing span, m	Maximal take-off mass, t	AC empty mass, t /Maximal payload, t	Combat radius, km / Ferry range, km	Maximal speed, km/h / M number	Engine number x engine type	Engine model / Engine design bureau (Engine manufacturer)	One engine thrust, / in afterburner, kN	Engine mass, t	Length, / diameter, m
YF-12	Lockheed	1963 2	32.7	16.9	77.1	27.2 N/A	2400	3220	2xturbojet, afterburner	J58-4 Pratt & Whitney	151	2.93	N/A 1.27
MiG-25	ANPK MiG	1964 ~600	23.3	13.5	36.7	19.6 2.18	950 1800	3000	2xturbojet, afterburner	R-15B-300 Soyuz AMNTK	76 112	2.59	6.65 1.64
Tu-128	Tupolev ANTK	1961 189	26	20	43	26 2	1300 5000	1665	2xturbojet, afterburner	AL-7F-2 Luyka-Saturn	66.7 99	2.1	6.65 1.3
Su-27	AVPK Suchoi	1977 N/A +	21.9	14.7	30	16.3 4	1500 4000	2430	2xbypass, afterburner	AL-31F; Luyka-Saturn	123	1.50	4.95 1.22
F-15	McDonnell-Douglas	1979 1415+	19.4	13.1	30.8	12.7 7.26	750	2650	2xbypass, afterburner	F100-PW-100 Pratt & Whitney	111	1.29	4.85 1.19
F-14	Grumman	1970 784	19.1	19.5 11.7	33.7	18.0 6.58	3220	2550	2xbypass, afterburner	TF 30-P-412A Pratt & Whitney	93	1.8	5.99 1.29
Su-37	AVPK Suchoi	1996 1	22.2	14.7	34	8	1600 3600	2500	2xbypass, afterburner	AL-31FP; Luyka-Saturn	125	1.57	4.99 1.28
MiG-29	ANPK MiG	1977 N/A +	17.3	11.4	17.7	10.9 3	650 2100	2400	2x bypass, afterburner	RD-33 Klimov corp.	81.4	1.05	4.3 1.1
F-16	General Dynamics	1984 N/A +	15.0	9.45	19.2	8.66 5.44	1000	2145	1x bypass, afterburner	F100-PW-100 Pratt & Whitney	111	1.29	4.85 1.19
F-22	Lockheed-Martin	1990 N/A +	19.6	13.1	26.3	15.4	1500	2130	2x bypass, afterburner	F119-PW-100 Pratt & Whitney	139	1.78	5.16 1.1
F-35B	Lockheed-Martin	2000 N/A +	15.6	10.7	28.8	13.9 10.1	833	1.6	1xbypass, fan afterburner	F-135 Pratt & Whitney + fan Rolls-Royce	159	1.45+	1.1 1.04

Refer notes p. 152.

they operate in the front-line zone, are able to perform (perhaps at a lower level of efficiency) additional missions, e.g. the destruction of ground and above-water targets, aerial reconnaissance, radio counter-measures. Thus, the concept of the **front-line fighter** assumes its ability to destroy both air targets (non-maneuvrable bombers and transport aircraft, armoured attack aircraft, enemy high-maneuvrable fighters, low-altitude airplanes, unmanned aircraft and helicopters, etc.) and ground targets.

The performance and design solutions of a **fighter-interceptor** and its power plant are different from those of a fighter. Interceptors are intended to strike low-maneuvrable air targets (e.g. strategic bombers) over their own territory far from the front line; this aircraft form a part of the anti-aircraft defence system. The basic concepts underlying the fighter-interceptor class are as follows:

a) concept of a fighter-interceptor that operates «on request» (operation in the mode of battle duty on the ground);

b) concept of a **patrol fighter-interceptor** that operates under «air intercept» conditions (operation in the mode of battle duty in the air at a certain altitude and speed).

Fighter-interceptors defend important industrial, military and public facilities located, as a rule, in large cities. This AC usually operates as follows: the anti-aircraft defence ground system detects an enemy AC at a long-distance; the fighter-interceptor takes off, climbs, attacks and shoots down the enemy AC and lands at the main or diversion airfield.

Power plant design in fighter-interceptors is determined by the following specific operational conditions:

- fighter-interceptors operate over their own territory beyond the zone of enemy anti-aircraft defence systems and enemy fighters;
- fighter-interceptors are not used in conditions of manoeuvrable air fight or against ground targets;
- fighter-interceptors attack single or not numerous air targets;
- fighter-interceptors are based on hard-surfaced airfields.

These concepts were implemented in the design of the Su-9, F-102, F-106, Su-15 supersonic fighter-interceptors equipped with advanced air-to-air missiles: their power plants have high-level thrust-to-weight ratio and rate-of-climb characteristics. Understandably, high-level invulnerability and cost effectiveness are not critically important for these AC.

**Transport aircraft** are intended for the air transportation of different kinds of freight and, as the need arises, passengers;

**Cargo aircraft** (Table 11.4) are transport AC intended for transporting cargoes, mail, technical equipment with attendants;

**Passenger aircraft** (Table 11.5) are civil AC intended for transporting passengers, their luggage and mail; in addition, they usually carry some cargoes, but cargo transportation is not their main function;

Table 11.4

## Reconnaissance and Cargo Aircraft Examples

Aircraft type	Aircraft						Engine						
	Aircraft design bureau (Aircraft manufacturer)	First flight date/ Total number	AC length, m	Wing span (rotor diameter, m)	Maximal take-off mass, t	AC empty mass, t / Maximal payload, t	Max. range, km / Full load range, km	Cruise speed, km/h	Engine number x engine type	Engine model / Engine design bureau (Engine manufacturer)	One engine thrust, kN	Engine mass, t	Length, m / diameter, m
pilotless RQ-4A	Teledyne Ryan	1998 N/A +	13.5	35.4	11.6	4.05 / 0.91	22200	639	1xbypass	AE 3007H Rolls-Royce + Allison	33.8	0.717	2.71 / 1.11
pilotless Воротей	NII PFM «KhAI»	2010 N/A +	0.6	1.05	2.5·10 <sup>-3</sup>	2·10 <sup>-3</sup> / 3·10 <sup>-4</sup>	90	90	1×electric motor	SIL-3026 Scorpion Power System	0.35	2·10 <sup>-4</sup>	0.08 / 0.04
Black Widow	Aero Vironment	1996	0.15	0.15	42·10 <sup>-6</sup>	4·10 <sup>-5</sup> / 2·10 <sup>-6</sup>	2	65	1×electric motor	N/A	8·10 <sup>-3</sup>	N/A	N/A
An-124	ANTK ANTONOV	1982 / 55	69.1	73.6	405	150	4500	850	4xbypass	D-18T ZMKB Progress	230	4.1	5.4 / 2.94
An-225	ANTK ANTONOV	1988 / 1	84	88.4	600	250	14700 / 4500	850	6xbypass	D-18T ZMKB Progress	230	4.1	5.4 / 2.94
helicop. Mi-26	Mil MVZ N/A +	1978 N/A +	12	32	56	20	480	295	2xturbo-shafts	D-136 ZMKB Progress	8500	1.05	3.96 / 1.67
A-300-600ST	Airbus	1994 / 4	56.2	44.8	153	45	813	750	2xbypass	CF 6-80 General Electric	27.9	3.98	4.24 / 2.36
MV-22B	Bell-Boeing	1989 N/A +	19.1	14.0 / Ø11.6	27.4	15.0 / 9.07	3340	510	2xturbo-shafts	T406-AD-400 Allison	4586	0.44	1.96 / 0.67
An-74 ≈An-72	ANTK ANTONOV	1983 / 121 +	28.1	31.8	33.5	19.9 / 10	2000 / 1000	720	2xbypass	D-36 ZMKB Progress	63.7	1.1	3.32 / 1.71
An-70	ANTK ANTONOV	1994 / 2 +	39.9	44.1	130	47 / 30	7500 / 4300	780	4xturbo-propfan	D-27 ZMKB Progress	10440	1.65	4.20 / 1.37

Refer notes p. 152

Table 11.5

## Passenger Aircraft Examples

Aircraft										Engine			
Aircraft type	Aircraft design bureau (Aircraft manufacturer)	First flight date / Total number	AC length, m	Wing span, m	Take-off mass / AC empty mass, t	Pass. capacity / Payload, t	Max. range / Full load range, km	Cruise speed, km/h / M number	Engine number × Engine type	Engine model / Engine design bureau (Engine manufacturer)	One engine thrust, kN (power, kW)	Engine mass, t	Length / diameter, m
IL-62M	Ilyushin AK	1970 / 194	53.1	42.5	161.5 / 69.4	186 / 23	11500 / 8800	870	4×bypass	D-30KY PMKB	108	2.65	5.7 / 1.45
Concorde	Aerospatiale + BAC	1969 / 16	62.1	25.6	181.5 / 79.9	128 / 12.7	6550	2150 / 2.02	4×bypass afterburner	Olympus 593 Rolls-Royce+SNECMA	169 / 172	N/A	3.81 / 1.22
S-21	AVPK Suchoi + Gulfstream	—	40.5	19.9	35 / 15	12 / N/A	N/A / 6500	2336 / 2-2.2	2× bypass afterburner	Rolls-Royce or Lyulka-Saturn	N/A	N/A	N/A
Boeing 747-400	Boeing	1988 / 694	70.7	64.3	395 / 177.7	660 / 65	15390 / 12700	939	4×bypass	JT9D Pratt & Whitney	252	4.12	3.26 / 2.47
IL-96	Ilyushin AK	1988	55.4	57.7	216 / 117	300 / 40	11000 / 9000	900	4×bypass	PS-90A PMKB	157	2.8	5.33 / 1.9
A 350-800	Airbus	2013	66.9	64	265 / HД	366 / HД	HД / 15000	903 / 0.85	4×bypass	Trent XWB Rolls-Royce	387	N/A	N/A / 3.00
A 380-800	Airbus	2004	72.8	79.8	590 / 277	150	14800	930 / 0.89	4×bypass	Trent 900 Rolls-Royce GP7200 Pratt&Whitney	311	HД	HД / 2.79
Boeing 777	Boeing	1994 / >630	37.8	28.4	51.7	124	8920	905	2×turbofan	PW 4084, PW 4098 Pratt & Whitney	393 / 444	6.77 / 7.48	4.87 / 2.87
Boeing 787	Boeing		56.0	58.8	205.4	200	14445	N/A	2×turbofan	Pratt & Whitney or General Electric	290	N/A	N/A / 2.92
An-148	ANTK ANTONOV	2004	29.1	28.9	36.4	80	7000 / 2200	870	2×bypass	D36-5AF ZMKB Progress	64	1.1	3.32 / 1.71
An-140	ANTK ANTONOV	1997	22.5	24.5	21.5 / 12.8	52 / 6	3700 / 1300	537	2×turbofan	TV3-117VMA-CBM1 Klimov corp. + Progress	1838	0.57	2.06 / 0.65

Refer notes p. 152.

**Cargo-passenger aircraft** are intended for transporting passengers and cargo at the same time; cargoes are arranged in under-floor sections or in special zones in the cabin; such AC are equipped with cargo doors of necessary size, strengthened floors, cargo attachment points and cargo handling equipment.

An effective conceptual solution to the problem of cargo-passenger transportation is designing a **convertible cargo-passenger aircraft**; its cabin or a section of the cabin is easily adjusted to transporting cargoes or passengers according to the current need. The design of some cargo-passenger aircraft makes it possible to load the under-floor cargo containers to the AC carrying capacity if there are no passengers aboard. If this design is practicable, which is not always the case, there is no need for convertibility. Cargo-passenger and convertible AC are designed to increase the efficiency of AC operation, to use aircraft carrying capacity to the utmost and round the clock (allowing for seasonal, day/night and directional variations in passenger/cargo flow on certain airlines, or one-time surges in the demand for passenger transportation to the venue of a major sporting event).

Transport AC are subdivided into airplanes, helicopters and dirigibles.

Our analysis of transport AC development prospects concerns aircraft designed specifically for air transportation purposes; AC converted from outmoded bombers, cargo and military transport aircraft and passenger versions of cargo AC are not considered.

Cargo aircraft are sometimes classified as military or civil transport according to their purpose, but usually an aircraft can be used in both versions or can be converted quickly from one version to another. According to their take-off weight and payload weight, cargo AC are grouped into light, medium-payload (or medium-capacity), heavy and super-heavy aircraft (in military terms: operational, tactical and strategic aircraft). This classification is only relative, as it reflects the current state of AC engineering; for example, the An-22 Антей airplane, which was originally considered a super-heavy AC, now belongs to the class of heavy cargo airplanes.

When designing power plants for cargo aircraft, it is necessary to take into account the following features:

1. To provide take-off/landing from/on natural-surfaced runways, a cargo airplane is usually a high-wing monoplane (this aerodynamic configuration helps to prevent foreign objects from getting into air intakes).
2. A large cargo cabin with a constant or nearly-constant cross-section (without narrow spots and crosspieces) makes it impossi-

ble to locate the engine in the fuselage.

3. Availability of an auxiliary power plant and other special features enhance maintenance autonomy and minimizes turnaround servicing time.
4. Augmented stability is crucial in situations when there is an abrupt change in the centre-of-gravity position, short-term and long-term disturbed motion during landing operations, e.g. low-altitude airdropping of military equipment.

The **passenger airplane** class comprises civil transport AC designed for transporting passengers and their luggage, mail and cargo.

Passenger airplanes have to meet the following main requirements:

- passenger capacity and range as needed; the variability of these parameters (range increase with simultaneous passenger capacity decrease in a further design modification);
- reliability and flight safety: standby systems and units, high operational viability, incombustible materials, emergency fuel discharge, arrangement of fuel tanks far from the passenger cabin, etc;
- comfort for passengers: a large pressurized cabin with normal or almost normal air pressure inside, air conditioning and individual air cooling, low level of noise, all-weather flying, etc;
- environmental friendliness — minimal outboard noise, low level of harmful emissions, etc.

Passenger airplanes may be classified according to their range (related to the airplane size, carrying capacity, and passenger capacity) as follows:

- airplanes of local airlines (range up to 1,000 km) or regional passenger airplanes (range of 500-1,500 km);  
(Note: general aviation craft with passenger capacity up to 30 persons, such as training, administrative, sports airplanes, gliders, air-taxi, etc., are also members of this numerous and structurally stable class, which is not analysed here. Aircraft piston engines and piston power plant units (radiators, carburetors, etc.) are still designed and manufactured for these airplanes.)
- trunk-route airliners usually subdivided into:
  - a) short-range trunk-route airliners (range of 1,000–2,500 km);
  - b) medium-range trunk-route airliners (range of 2,500–6,000 km);
  - c) long-range trunk-route airliners (range from 6,000 km up to more than 11,000 km). (Note: as the Earth's radius is 6,378 km and its half-perimeter is 20,038 km, the range of ~16,000 km means nonstop flights between the most remote points on the Earth's surface, e.g., Kiev — Sydney, New-York — Tokyo, Novosibirsk — Brasilia).

## 11.2. Power Plants for Super-Heavy Aircraft: Design Philosophy

Designing **super-heavy cargo airplanes** is one of the most promising areas of cargo AC development, where the Ukrainian Aviation Scientific-Technical Complex named for O.K. Antonov is the rightful leader. The heaviest commercial cargo airplane An-124 Руслан (maximum payload: 150,000 kg, Fig. 11.2, ref. Table 11.5) and the heaviest experimental cargo airplane An-225 Мрія (Fig. 11.3, ref. Table 11.5, maximum payload: 250,000 kg) are unsurpassed aircraft. According to the well-known square-cube law, each step in their development was stipulated by a multi-dimensional complex program addressing a whole set of interconnected problems in the areas of aerodynamics, strength, material science, structural design, production techniques, electronics and avionics, control systems and, undoubtedly, power plant design.

Designing the An-124 and An-225 involved the following major innovations in power plant design:

- New turbofan engines were developed with a high rate of by-pass ratio ( $\sim 6$ ), take-off thrust of 23,000-25,000 kg, fuel rate of no more than 0.57-0.59 kg of fuel / kg of thrust  $\times$  hour at the cruising rating; the engines provide steady work and acoustic stability as well as high cycle parameters (compression ratio  $\pi_K = 27$  and temperature  $T = 1600$  K);

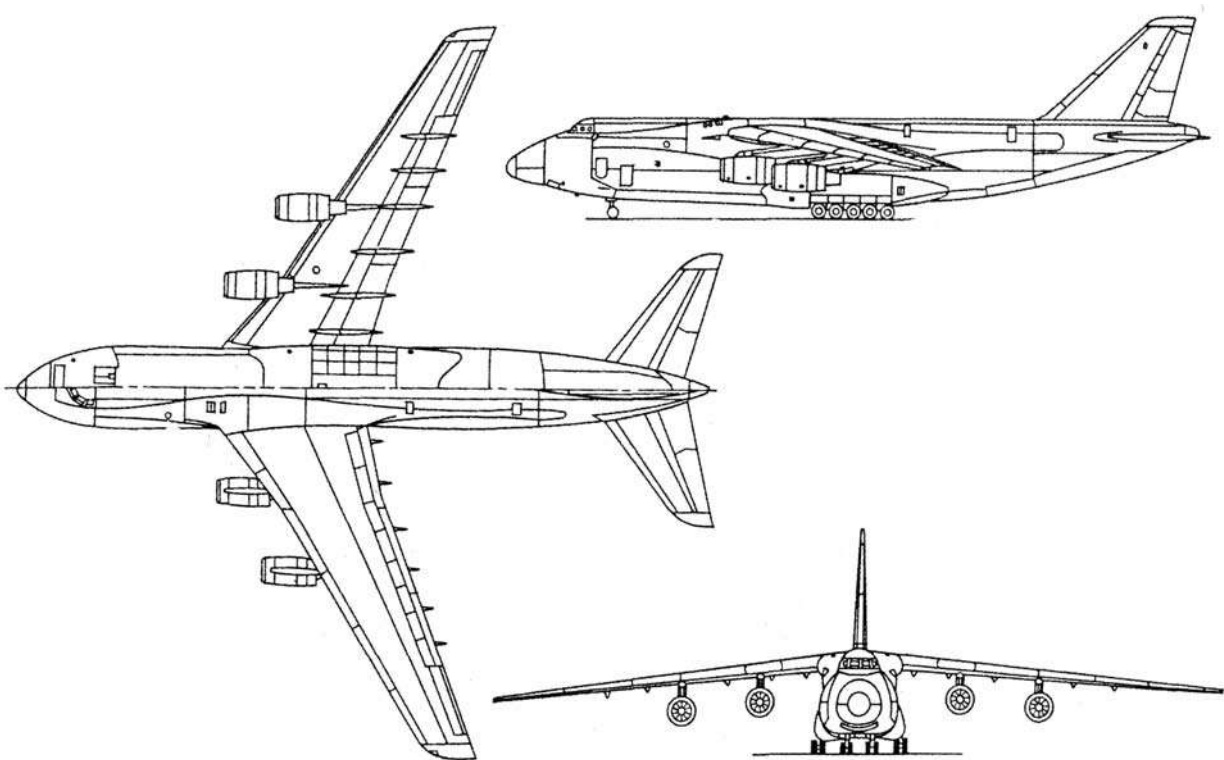
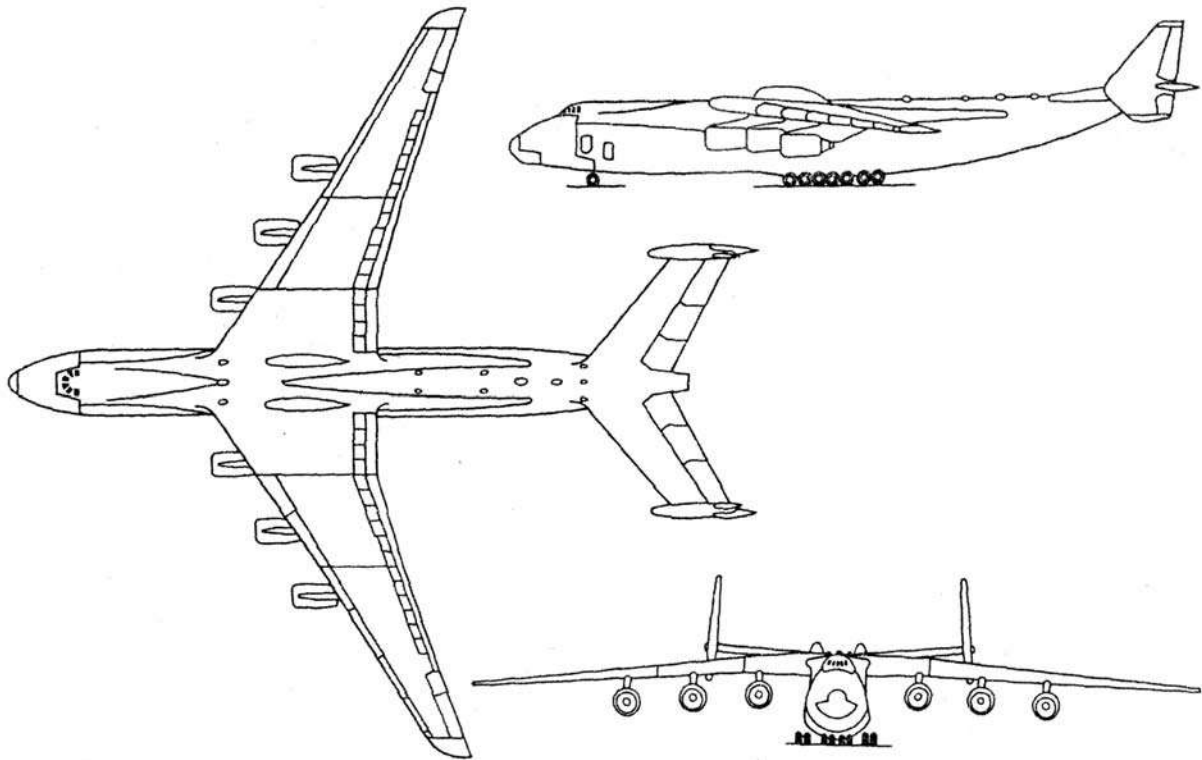


Fig. 11.2. Heaviest Commercial Cargo Airplane: An-124 Руслан





**Fig. 11.3. Heaviest Experimental Cargo Airplane: An-225 Mpiя**

- Analytical, design and arrangement operations were developed by the Central Aero-Hydrodynamic Institute to minimize lift/drag ratio losses caused by the power plant and wing-power plant interference; more than 60 models of inlets, fan fairings, gas generator fairings, pylons and «wing + power plant» layouts were tested; the wind tunnel experimental testing time totaled 1,500 hours;
- A number of power plant design solutions were worked out to design AC with the back range of centre-of-gravity positions and low static longitudinal stability; these solutions lowered lift/drag ratio losses for balancing and increased the lift-to-drag ratio  $\Delta K = 0,5$ , simultaneously reducing the structure mass by 2,800 kg;
- AC maintainability was substantially enhanced: the pre-flight preparation time was reduced to maximum 2 hours, the time of replacement of faulty parts — to 30 minutes, and maintenance work hours — to 25 man-hours per hour of flight;
- AC power plants were protected against damage from collisions with birds and lightning strikes, to ensure their invulnerability in operation;
- A computer-aided design system for super-heavy long-range transport aircraft was developed and implemented; on its basis 540 versions of super-heavy long-range transport AC layout were analysed; tension, deformation and resource parameters of their main structural

elements (such as engine attachment points and engine pylons-to-wing joints) were studied for all versions.

### 11.3. Power Plants for Large-size Cargo Transportation Aircraft: Design Philosophy

Only specially designed AC can meet the challenge of conveying large-size (bulky) cargoes which cannot be delivered by road and railway transport. The Beluga cargo airplane (Fig. 11.4, ref. Table 11.4) is intended for the transportation of large-size cargoes such as fuselage sections, e.g., sections of Airbus Industrie European Consortium wide-body passenger airplanes.



a



b

**Fig. 11.4. A-300-600T Beluga Transport Airplane of Airbus Corporations for Bulky Cargo Transportation:  
a — overall view; b — airplane off-loading**

Sometimes items to be transported are so large that they can be conveyed by cargo airplanes only as external stores; in this case the aerodynamic arrangement and power plant of cargo airplanes has to meet special requirements. For example, the Soviet Буран aerospace aircraft was transported by the An-225 Мрія super-heavy cargo airplane as an external load (Fig. 11.5). Earlier, wings of the An-124 Руслан airplane manufactured in Tashkent were transported to Kiev for assembling by the An-22 Антей airplane. The helicopters provide the excellent capabilities for large-size cargo transportation as external stores (Fig. 11.6).

The size and carrying capacity of the An-225 Мрія make it possible to use the airplane for launching earth-to-orbit vehicles (Fig. 11.7). The advantages of this air launch are beyond any doubts. Although the separation of two vehicles with different lift-to-drag ratios, at altitudes and speeds close to the airplane-carrier maximum values, is a matter of great difficulty, the ANTK Antonov solved this problem when developing the An-225 aerodynamic layout and power plant.

The low-altitude parachute-free airdropping of military equipment creates additional challenges for the control systems and power plants of cargo AC. Modern tanks are capable of withstanding the impact of landing under conditions of contour flying at the altitude of 20 m and minimum speed. Thus the AC designers faced and managed to answer the question of how to provide flight safety when the center-of-gravity position is changed abruptly at a low altitude.



**Fig. 11.5. Transportation of Soviet Буран Aerospace Aircraft as External Load Fastened to the Fuselage of the An-225 Мрія**



**Fig. 11.6. Heaviest Commercial Cargo Helicopter: Mi-26**



**Fig. 11.7. Earth-to-Orbit Vehicle  
Launching by An-225 Mriya  
Super-Heavy Cargo Aircraft (project)**

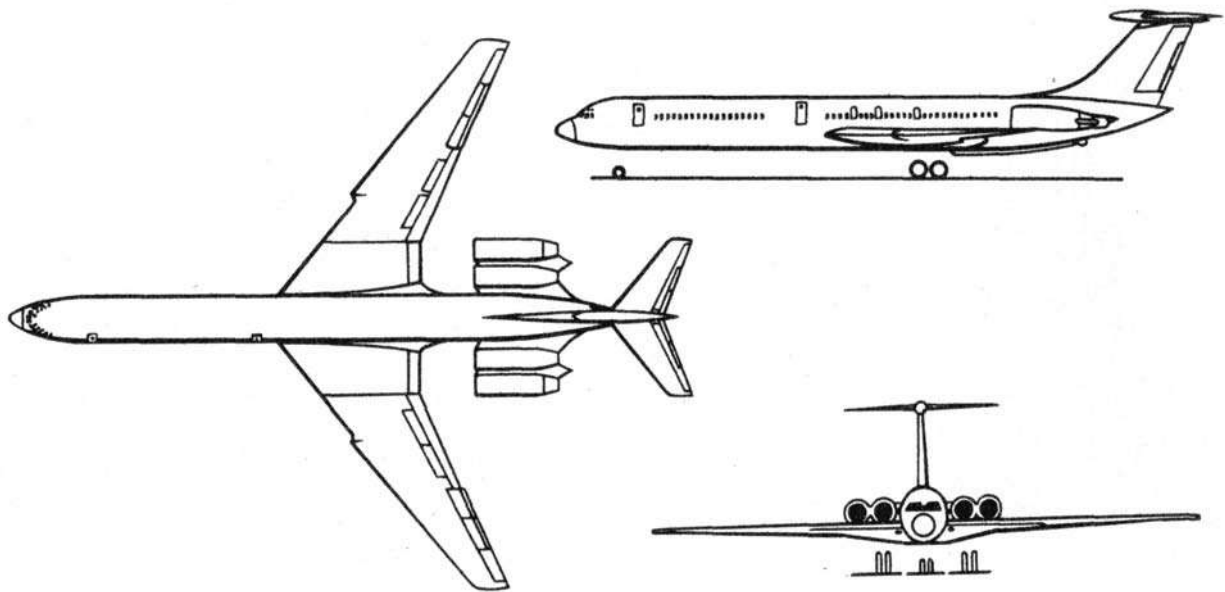
## 11.4. Ultra-Economical Power Plants for Long-Range Aircraft: Design Philosophy

Designing power plants for the AC which are both super-heavy and long-range poses extraordinary challenges. Long-range airliners have always been technological leaders and acted as aviation progress indicators. The era of mass passenger conveyance by air began when the first jet airliners, such as «Comet», Tu-104, Boeing 707, were put into operation. Since then air conveyance has been a strong contender to railway, road and sea transport. At the beginning, however, the comfort of air travel and, as a result, its competitive ability suffered from the low efficiency of AC power plants: transatlantic flights were only possible with 2-3 intermediate landings.

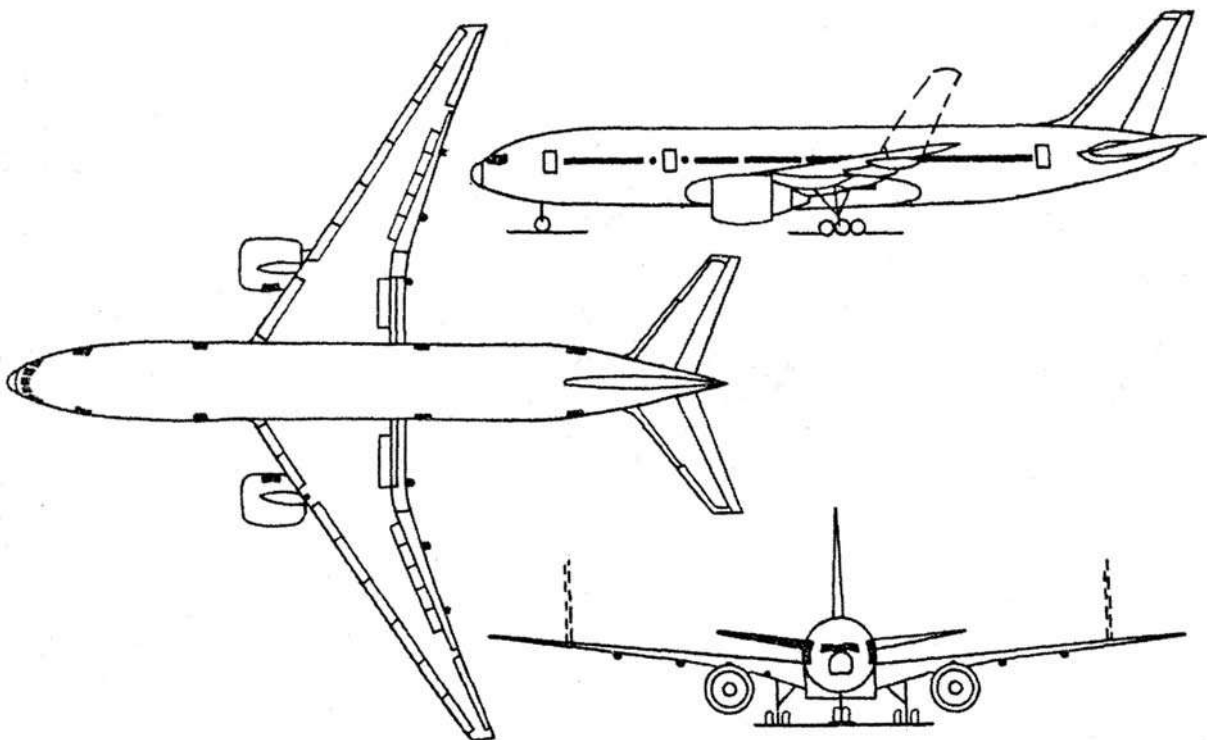
Later generations of airliners, equipped with more powerful and efficient power plants, began to dominate transatlantic conveyance, having forced out ocean-liners. In the USSR the class was represented by the following airliners: first, the unique Tu-114 high-speed turbo-prop, later, the turbojet IL-62 (Fig. 11.8, ref. Table 11.5). In those years, all airliners had to have four engines in compliance with the Airworthiness Requirements. Following the significant achievements in aircraft engine design and the enhanced reliability of power plants, in 1991 a new ETOPS certification came into effect, which removed all range limitations (including those of oversea flight) for certified twin-engined airliners. As a result, two of the three long-range airliners (A330; A340, Boeing 777 — Fig. 11.9, ref. Table 11.5) designed in 1991–1994 were twin-engined craft. Twin-engined airplanes surpass four-engined ones in efficiency and dominate the current airplane production.

At present, there are two development concepts of long-range air conveyance elaborated by the two world airplane market leaders: Airbus Common European Consortium and Boeing Corporation (USA).

According to the concept of air conveyance consolidation (**hub-to-hub concept**), air transportation service should be organized as follows: long-range conveyance is carried out between large airports (hubs) by airplanes of large passenger capacity; passengers are ferried to these airports by short-range and medium-haul airliners. Designing the largest passenger airplane, A 380 (Fig. 11.10, ref. Table 11.5), was a logical step in the implementation of this concept (the project was started in 2000, the first flight took place in April 2005). It is a truly giant airliner: its basic version (A 380-800) maximum take-off weight is 560,000 kg, length: 72.75 m, fuselage diameter: 7.14 m, passenger capacity: 555 seats. However, this «fatty's» cruising speed is 900 km/h, and its range capacity with



**Fig. 11.8. IL-62 Long-Range Trunk-Route Airliner**



**Fig. 11.9. Boeing 777 Long-Range Trunk-Route Widebody Aircraft**

555 passengers is 14,800 km. The passengers are accommodated on two decks. Interestingly, the first customers gave preference to specially arranged advanced comfort 400-450-passenger variants, with a pool, berths, cinema, etc., over the basic three-class versions. The plane has four turbofan engines of 36,300 kg thrust.



**Fig. 11.10. A 380 Largest Passenger Aircraft**

Boeing Corporation engineers are strong opponents of the hub-to-hub concept who author the idea of air transport fragmentation (the point-to-point concept). They argue that there are too few airports whose ground infrastructure meets the A 380 aircraft category (category F, according to ICAO classification) requirements. In early 2005, only four airports in the world answered the purpose; and ten more were ready to spend several million dollars on their updating. In contrast, the Boeing 777-300 airplane is able to take-off from and land on hundreds of airports all over the world. Boeing considers the airplane's current passenger capacity to be optimal and is not going to increase the AC passenger capacity, especially since there were no orders for the one-class and 550-person version of Boeing 777-300. Nevertheless, Airbus had a store of 127 orders from 16 operators as of June 2005 (none from the USA).

The **point-to-point concept** presupposes passenger conveyance from one point to another by direct flights; in this case the greatest possible number of cities is connected by direct air routes. Boeing experts rightly assume that nobody enjoys changing planes in overcrowded hubs, but everyone would prefer to fly non-stop from A to B even if A and B are not very large cities. Therefore they suggest developing durable, low-maintenance airplanes with the passenger capacity of short-range/medium-range trunk-route airliners and range capacity of long-range airliners. For this purpose, Boeing has started designing the Boeing 787 Dreamliner (Fig. 11.11, ref. Table 11.5) wide-body, low passenger capacity, long-range airplane. Its base version is the Boeing 787-800 intended for transporting 223 passengers within 15,700 km at the cruising





**Fig. 11.11. Boeing 787 Dreamliner  
Long-Range Trunk-Route Wide-Body Aircraft**

speed of 900 km/h; it is priced at \$ 120 million, which is a reasonable figure for small AC operators. There are other versions in the Boeing 787 family: the Boeing 787-900 «stretched» (lengthened) fuselage airliner, seating 256 passengers in three classes with a range of 15400 km, and the Boeing 787-300 290-seat, two-class, medium-range airliner with a range of 6,500 km. In 2005 Boeing had orders for 255 airplanes from 18 operators (mainly from the USA and China).

At first Airbus experts criticized this concept as low-profitable, but later they admitted its advantages and declared that the future would see both concepts used together. Airbus has even started to work on the A 350 wide-body low passenger capacity, long-range airplane on the basis of the A 380 design solutions. It is noteworthy that all Airbus planes have common maintenance procedures; flight and technical service personnel do not have to be trained anew; a lot of spare parts are interchangeable. Thus AC operators are encouraged to form their fleet entirely from Airbus planes as the latter are likely to satisfy all their needs in terms of seating capacity and range.

General-duty aircraft designers do not want to fall behind. At present several foreign corporations are testing VIP versions of 16000-km-range business AC.

To make AC more cost-effective, up to 75% of friction drag might be eliminated due to the laminarization of the flow-past by means of **boundary layer suction**. The air sucked off the wing surface and surfaces of other units might be used in the power plant engines. Substantial financial and intellectual resources were invested in the attempts to attain this goal by those involved in the Lockheed C-5A Galaxy and An-124 Руслан projects. However, their solutions did not go beyond the stage of experimental AC flight tests. The application of boundary layer control to serially produced airplanes is hampered by the resulting low operational characteristics, especially during take-off from and landing on natural-surface runways. In future, due to innovative design and manufacturing solutions, it might become possible to increase the lift-to-drag ratio, save fuel and raise the efficiency of cargo airplanes.

The creation of **nuclear-powered aircraft** (or nuclear-propelled AC) is another promising way of development. Their range is practically limitless and is restricted only by the crew's psycho-physiological parameters. Both the USSR and the USA (General Electric Company) spared no expense and succeeded in designing experimental strategic strike aircraft driven by nuclear power plants. The programs were kept secret and have only recently been reported in the media. Though both countries had solved the main engineering problems, the programs were stopped because of radiation danger for the crew and the threat of environmental pollution in case of accidents.

Unmanned aircraft designers are leaders in the development of long-range AC. For example, the RQ-4A Global Hawk reconnaissance unmanned aircraft (Fig. 11.12) designed by Teledyne Ryan Aeronautical Corporation (USA) has the following parameters: range: 22,200 km, flight endurance: 36 hours; take-off weight: 11,620 kg; payload: 910 kg; wingspan: 35.4 m. The RQ-4A standard flight is formed as follows: 5,500-km flight to the surveillance zone, 24-hour patrol in the zone, 5,500-km return flight. The RQ-4A aircraft is able to patrol a 137,000-km<sup>2</sup> area on a single flight; its automated operation is based on data obtained from satellite navigation systems. Another example is the «Helios» large-size unmanned experimental AC (Fig. 11.13), designed by AeroVironment

Incorporation (USA). Currently, the AC can stay in the air, at the altitude of 15 km, for 96 hours, but its designers claim that the flying time may be increased to one year. The Helios's power plant is composed of 14 electric motors, 1.5 kilowatt each, powered by solar batteries. The aircraft is tailless; its flight speed: 45 km/hour; take-off weight: 750 kg; wingspan: 74 m.



**Fig. 11.12. RQ-4A Global Hawk Reconnaissance Unmanned Aircraft Designed by Teledyne Ryan Aeronautical Corporation (USA) )**

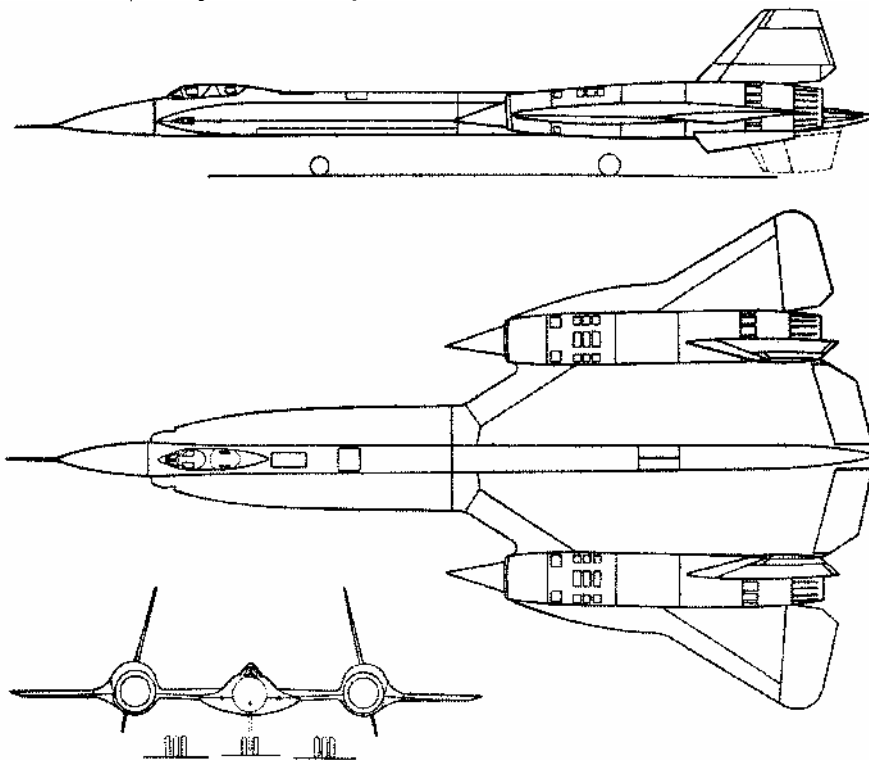


**Fig. 11.13. «Helios» Large-Scale Unmanned Experimental Aircraft with the Propulsion System Composed of Electric Motors Powered by Solar Batteries**

## 11.5. Power Plants for High-Speed Aircraft: Design Philosophy

Predictably, the fastest AC are fighter-interceptors. The Soviet and US aviation experts independently developed **superhigh-speed fighter-interceptors** to counteract the creation of the XB-70 strategic bomber in the USA and M-50 strategic bomber in the USSR respectively. The fighter-interceptors of this class were supposed to fly at the speed of  $M \approx 3.5$ .

The US government planned to establish the North American (USA-Canada-Alaska) joint air defence system with a completely radar-covered zone, a joint air observation post, high-speed fighter-interceptors and a set of ground equipment. Lockheed Corporation was contracted for building the YF-12 fighter (Fig. 11.14, Table 11.3). As the aircraft was to be equipped with highly effective long-distance high-speed manoeuvrable air-to-air missiles, it did not need high-level manoeuvrability itself. Therefore, to save on the aircraft mass, the flight maneuvering acceleration was limited to  $\sim 3.5$ , similarly to transport AC.



**Fig. 11.14. Lockheed's YF-12 Super High-Speed Fighter-Interceptor**

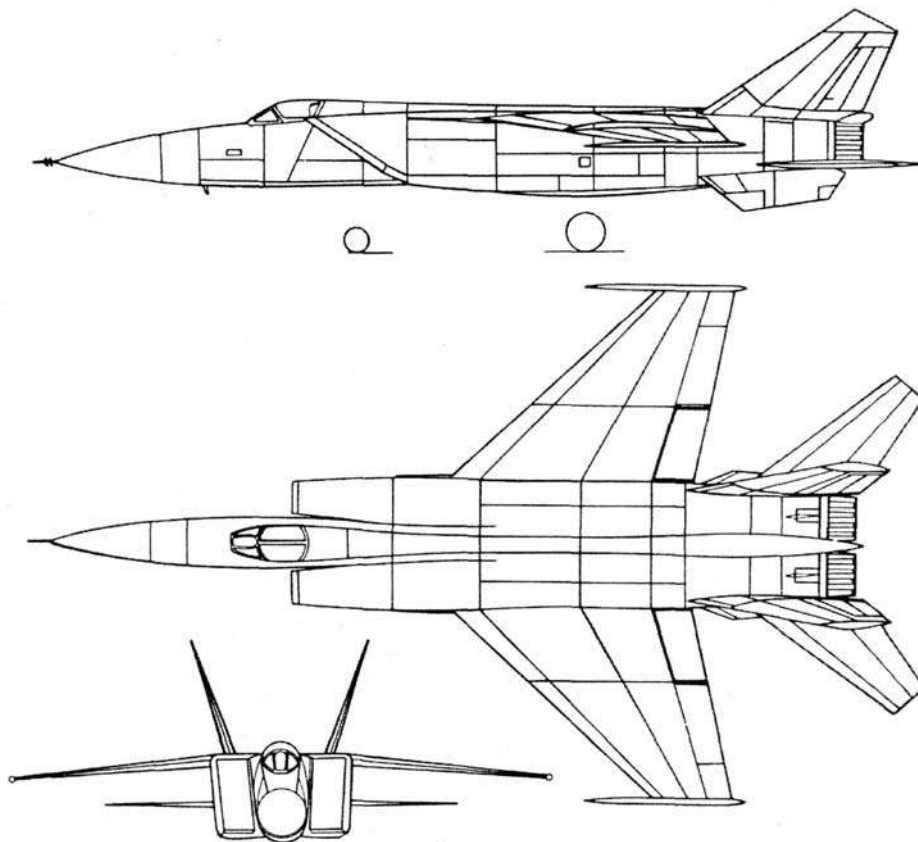
Following the development of intercontinental ballistic missiles, the XB-70 and M-50 programs were cancelled, as was the expensive YF-12 project. However, some of the groundwork was later used in the commercially produced Lockheed SR-71 Blackbird superhigh-speed strategic reconnaissance plane. Although similar in its outward appearance and altitude-speed performance to the predecessor, the SR-71 has lower stan-

dards of operational development, manufacturability and power plant characteristics. For example, the duration of the SR-71 pre-flight preparation exceeds 24 hours, which is inadmissible for a fighter-interceptor.

When the MiG Aviation Scientific Industrial Complex was given a task to create a countermeasure against reconnaissance flights over the USSR territory, they solved this complicated problem by designing the MiG-25 superhigh-speed interceptor (Fig. 11.15, ref. Table 11.3), without lowering manoeuvrable overload standards. The AC was put into serial production as interceptor; its versions were used as superhigh-speed reconnaissance aircraft. Later the MiG-31 aircraft was designed, with an airframe and power plant similar to those of MiG-25, but provided with advanced avionics that gave a sixfold increase in the AC effectiveness.

Pilotless semiexpendable anti-aircraft-missile complexes can be viewed as the next step in the development of this concept. Fighter-interceptors that operate from their on-duty position on the ground represent the first (reusable) stage of the anti-aircraft defence system consisting of a fighter-interceptor and air-to-air missile.

Passenger airplane designers made their own contribution to the development of superhigh-speed aircraft. After the creation of military supersonic AC, both small fighters and heavy strategic bombers (such as



**Fig. 11.15. MiG-25 Superhigh-Speed Fighter-Interceptor**

B-58, XB-70), the idea of supersonic passenger airplane was in the air; no wonder that aviation specialists in Great Britain, USSR, USA and France began preliminary studies almost simultaneously. The main challenges in designing a **supersonic passenger airliner**, or **supersonic transport (SST)**, concerned the development of its aerodynamic configuration and efficient power plant. In essence, both factors are important for minimizing the AC impact on the environment: noise created by its powerful power plant makes during the take-off, noxious gas emissions and the effect of compression shocks upon people and nature along the route.

The development of this class of airplanes was not seen as a purely engineering problem. There was a political dimension: the success of the Soviet program would have demonstrated the advantages of socialism over capitalism. In Europe, the creation of the SST would have meant the victory of European science and engineering in competition with Americans. British and French teams came up with an SST conceptual design which was technically feasible but required joint financial, engineering and intellectual efforts. The governments of Great Britain and France took an interest in the project and in 1962 signed an agreement on the joint development, production and operation of the «Concorde» SST.

From the very start, «Concorde» was intended for transatlantic passenger operations. After a take-off in London or New York the airplane would fly at a supersonic speed mainly over the sea. Besides, the USA — Europe transatlantic flights carried a high-density passenger flow, and among passengers on this route quite a few were assumed ready to pay more to save time and enjoy a comfortable and prestigious journey across the Atlantic. A range of 6,000 km, speed of  $M=2.2$  and 100-seat capacity were chosen as the most cost effective combination.

The Soviet SST Tu-144 was designed for operation within the USSR, above the vast Soviet territory, with then existing national standards of admissible noise both in airports and along airways.

Meanwhile, in the USA, President J.F. Kennedy approved an ambitious, politically motivated decision to build an American SST that would excel the European and Soviet counterparts in all principal parameters, including a cruising speed of  $M=2.7-3$ , 180–200-passenger capacity, 6,400-km range. Subsequent events proved that the idea was intrinsically wrong and could not be implemented at the level of science, engineering and technology of that time. Not only the airplane size, but also the cost of its development and production, as well as the prospective ticket price would be gigantic. Thus the Boeing 2707 program remained on paper, while Boeing concentrated on designing wide-body high-capacity passenger aircraft.



**Fig. 11.16. Tu-144 and «Concorde» Supersonic Passenger Airliners**

The Tu-144 and «Concorde» (Fig. 11.16, ref. Table 11.5) looked remarkably similar, not because of plagiarism but rather as a result of identical requirements and specifications in addition to the same level of technological development. Moreover, before long European and Soviet designers started cooperating to expedite their SST entering into service ahead of the USA. Having abandoned the plans of creating their own SST, the USA tried to hamper the European SST project by hitting its sorest point — the environmental issue. SST flights were prohibited over the US territory after American operators had placed more than half of the total number of orders for «Concorde». Some European countries followed suit: for example, Switzerland prohibited SST flights over the Swiss Alps in view of a probable increase in avalanche formation caused by compression shocks.

However, it was the global energy crisis, bringing about soaring fuel prices, which became fatal for SST programs. In the USSR the program was stopped on the stage of flight tests<sup>1</sup>.

«Concorde» entered service on January 21, 1976 with flights London–Bahrain (British Airways) and Paris–Dakar–Rio de Janeiro. Flights to the USA (Washington) were permitted in May, 1976. In 1976 the «Con-

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<sup>1</sup> Formally, the Tu-144 did enter service and performed 55 regular flights from Moscow to Alma-Ata (distance of 3,260 km) from November 1977 to May 1978. Over this period of time the 80-seat airplane delivered 3,284 passengers. But the airplanes belonged to the Ministry of Aviation Industry, and the Soviet aircraft operating agency «Aeroflot» did not take an interest in operating SST.

corde» airplanes performed about 1000 flights and transported more than 45,000 passengers. 16 airplanes were manufactured, but they could not cover the cost of the SST development. To make «Concorde» operation cost effective, ticket prices were raised to an exorbitant level: 60 times that of the ordinary airfare.

In 1990 the second stage of SST development (**SST-2 program**) was started when NASA launched two programs. The focus of the first program was on studying environmental issues; the second program was meant to concentrate on SST aerodynamic features and power plant design. Companies from Japan, France, Italy, Germany and Russia were involved in the programs. Soon the necessity of pre-production research in flying laboratories became apparent; for this purpose the Tu-144D aircraft was restored and converted into a flying laboratory. In 1991 the Tupolev ANTK (USSR) and Rockwell Corporation (USA) signed an agreement on collaboration. As the USSR disintegrated and Russia ceased financing the program, the USA remained the only source of funding. In November 1996 the newborn Tu-144LL (flying laboratory) made its maiden flight. That was followed by 19 more flights and a number of major experiments. In January 1999 the US experts summarized the results of SST-2 program: the current level of scientific and technological development is not sufficient for the construction of supersonic passenger aircraft with acceptable parameters of price, efficiency, weight and environmental impact.

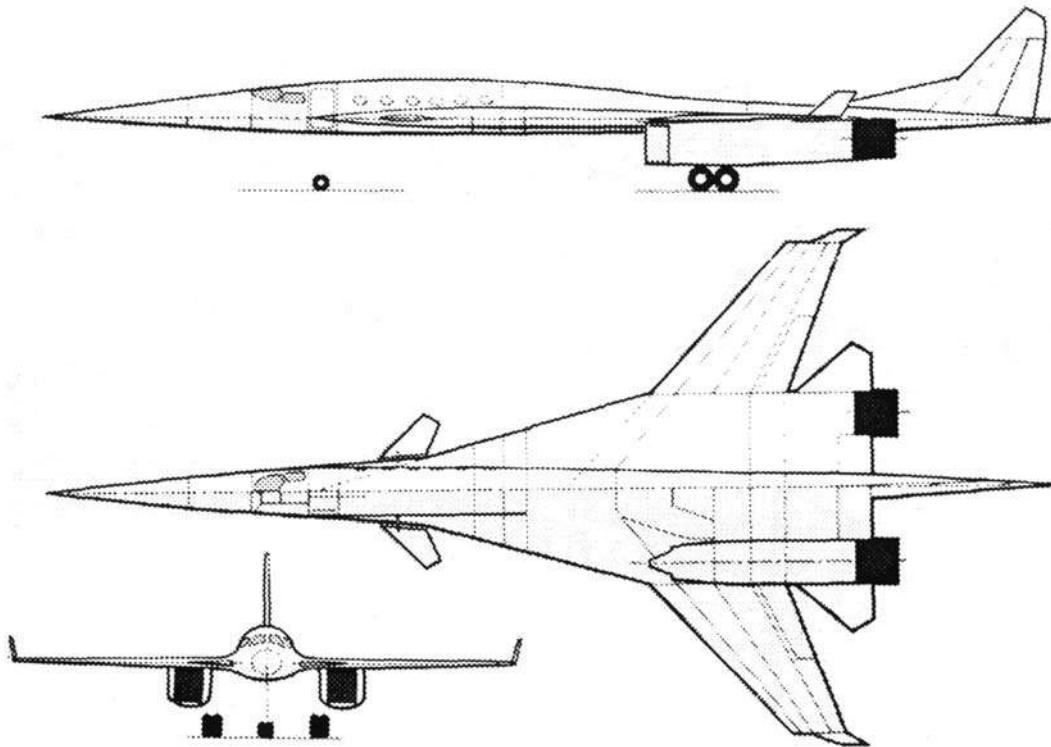
The ground overpressure during an SST flight was shown to depend on the airplane length. As a result, a new concept of **the small SST of business class** (administrative supersonic airplane) was created and named a Supersonic Business Jet (SSBJ). In the USA two companies — Aerion Inc. and SAI Inc. — have been developing two parallel projects since 2004. In Russia the concept is known under the name «supersonic administrative airplane» and is developed by the Tupolev ANTK (Tu-444 program) and AVPK Suchoi together with Gulfstream Aerospace corp. (C-16 program — Fig. 11.17, ref. Table 11.5). Experts predict a demand for 300-400 AC at a price of \$80 mln. per commercially produced plane.

Interestingly, the Boeing 747SP is able to fly non-stop from London to Singapore (flight time: 20.8 hours). The «Concorde» supersonic transport airplane covers this route (with two landings, in Beirut and Mumbai) in 18.4 hours.

### **11.6. Power Plants for Environmentally Safe Aircraft: Design Philosophy**

The **concept of environmentally safe aircraft** represents another trend in aircraft development. It is not as «glamorous» as designing SST, but no less important and quite complicated in engineering terms. Large

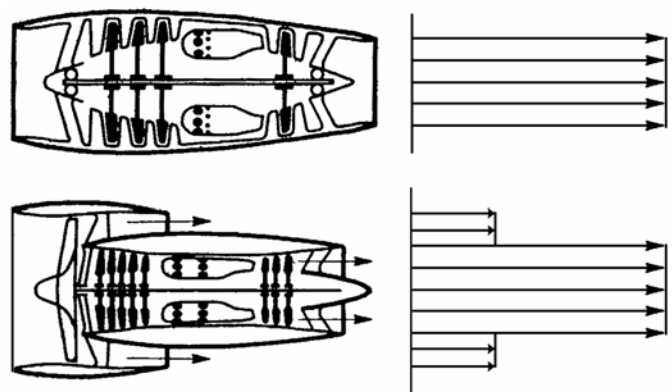




**Fig. 11.17. C-16 Administrative Supersonic Airplane Designed by AVPK Suchoi (Russia) in Collaboration with Gulfstream Aerospace corp. (USA)**

airports are located near big cities, in densely populated areas, so airplane noise and emissions create serious problems for people living in the vicinity. Governments in many countries have set strict regulations regarding the level of noise and air pollution any aircraft is permitted to produce. Domestic airplane designers have either to comply with the imposed standards or forget about international air routes.

The preference given to turbofan engines and propeller-fan power plants in the new generation of airplanes is largely due to their lower *noise level*. The main source of AC noise is jet blasts; the noise level depends on the velocity step between the gas jet and surrounding gas. Naturally, a bypass engine produces less noise than a conventional turbo-jet engine with the same thrust (Fig. 11.18).



**Fig. 11.18. Speed Patterns in Turbojet and Bypass Engines**

Hydrogen fuel may provide a principal solution to the problem of *atmospheric pollution with harmful emissions*. In the USSR, just before its disintegration, some promising results were obtained in this area. Extensive research efforts were aimed at the gradual introduction of hydrogen as fuel for new AC generations. For example, the most powerful Soviet carrier rocket, «Энергия», was a liquid oxygen/liquid hydrogen (LOX/LH) rocket-powered vehicle. Hydrogen fuel is not only pollution-free but also has a high power generating capacity. Its combustion heat is more than 114 megajoules per kilogram compared to ~43 megajoules per kilogram that the TS-1, a commonly used aviation fuel, provides. Crucially for SST, its cold resource exceeds that of other aviation fuels. The hydrogen boiling point of 20<sup>0</sup>K and its evaporation heat of 455 kilojoules per kilogram enable an effective cooling of AC heat-loaded structural components. However, a large-scale application of hydrogen fuel is conditioned by the development of nuclear-hydrogen power engineering and the availability of solutions to major research and design problems related to structural materials, thermal insulation and AC operating safety. For example, at the temperature of ~1000<sup>0</sup>C hydrogen penetrates deep into any material and dissolves in solid titanium, nickel and other metals. In 1988 flight tests involving the experimental Tu-155 AC, which was able to use liquid hydrogen and other cryogenic liquids as fuel, were started; unfortunately, the program was cancelled after the disintegration of the USSR.

### **11.7. Power Plants for Multimode and Multipurpose Aircraft: Design Philosophy**

When super-high-speed aircraft are designed, their power plants are optimized for supersonic super-high-speed conditions. During the take-off, landing and flight at a subsonic speed, a higher level of fuel consumption and the deterioration of other flight parameters are allowable. It is much more complicated to develop a power plant for a multimode AC that has to retain effective performance under varying flight conditions (different values of flight speed, air density, etc.)

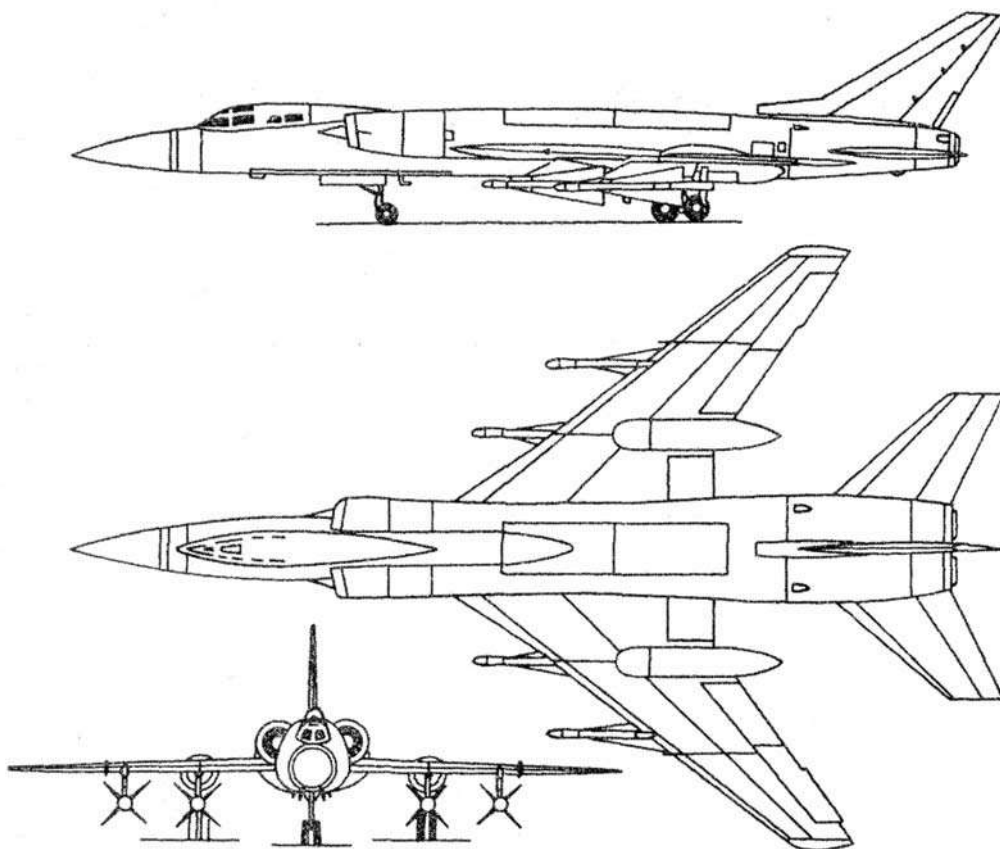
A **patrol interceptor** intended for anti-aircraft defence against high-speed bombers is a typical example of multimode AC. A conventional interceptor, equipped with powerful engines and operating from the ground, takes several minutes to climb the altitude of 11,000 m, but the XB-70 bomber is able to fly at 60 km per minute, that is 300 km in 5 minutes! Although patrol interceptors on duty in the air, at the altitude of 11,000 m, do not need super-short response time, their operation costs are much higher. The first Soviet all-weather Yak-25 fighter and its variant, the Yak-28 supersonic aircraft, represent the class of modern patrol interceptors. Their power-plant designers have successfully combined two flight

modes: long-term patrol in an economical subsonic cruise mode and a supersonic spurt for interception.

Designers of **escort fighters** and heavy **long-range fighters** face similar problems. Indeed, designing this class of AC is a matter of high complexity because of their operating conditions: these aircraft fly distances of 1,500–2,000 kilometres to engage in combat missions against enemy local («aboriginal») fighters which

- have shorter range and, as a result, are lighter and more manoeuvrable;
- stay within the operative range of their ground-based radar stations;
- are more numerous and can be supported by additional fighters on call missions.

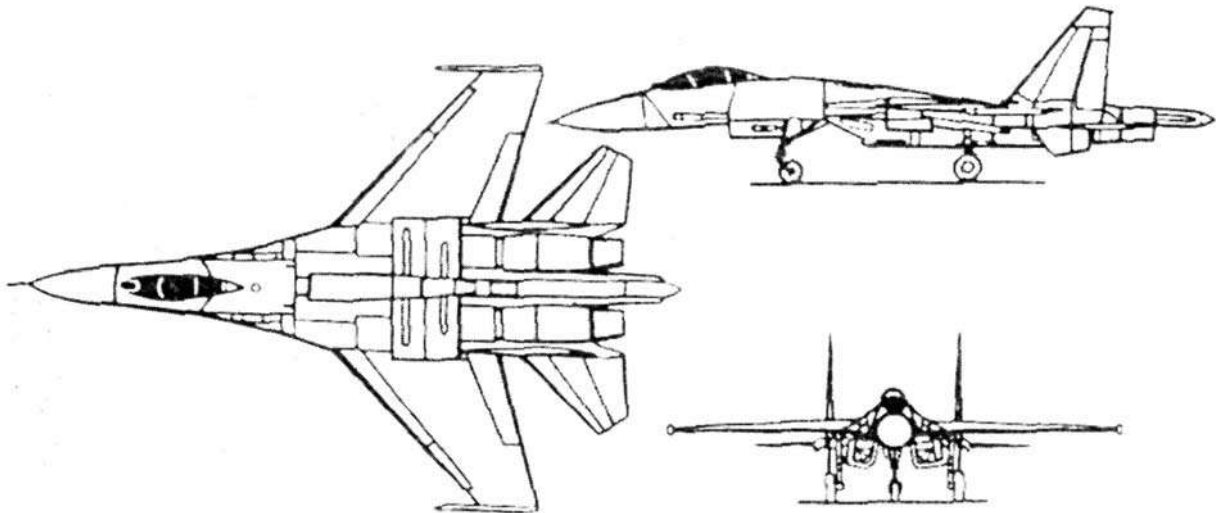
In the USSR this concept was implemented in the Tu-128 AC (Fig. 11.19, ref. Table 11.3) designed by the Tupolev ANTK. This AC also met some additional requirements set by the then existing Soviet realities: in fact, it was a **long-range fighter**. Its main goal was to provide interception lines outside the USSR borders in order to prevent the US aircraft attacks from the north. Deterioration in the Soviet-Chinese relations made defence problems urgent for the eastern frontiers, because the Soviet Far



**Fig. 11.19. Tu-128 Supersonic Long-Range Fighter**

East aerodromes were sparse while radar-tracking and anti-aircraft-missile defence did not cover the entire area. The Tu-128 was a breakthrough in aircraft design as its power plant managed to satisfy a number of conflicting requirements.

The Su-27 aircraft (Fig. 11.20, ref. Table 11.3) is another example of a remarkable breakthrough. Designed as an escort fighter for the Su-24, Tu-22, Tu-22M aircraft in the European theatre of war, it proved to be so effective that an extended family of AC was designed on its basis, allowing to consider it a rare successful example of **multipurpose aircraft**.



**Fig. 11.20. Su-27 Superiority Fighter**

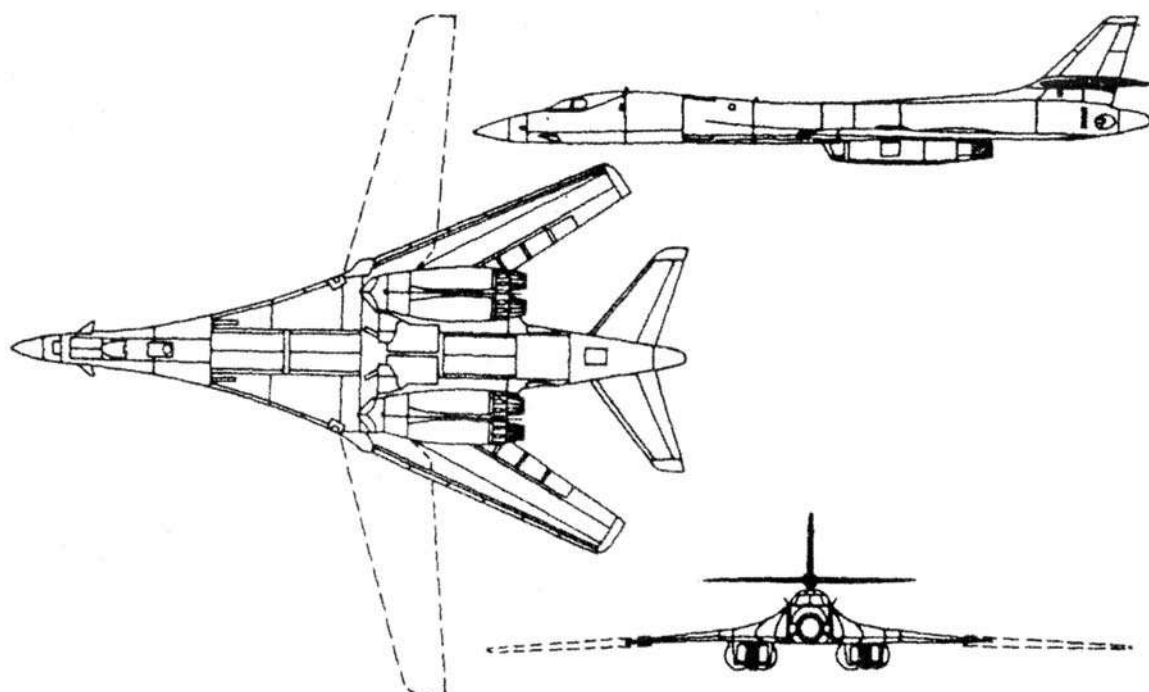
However, attempts to implement the concept of multifunctionality in aircraft design encounter serious difficulties. The concept is popular in the USA, where even the classification of military AC presupposes their multifunctionality: there is a single "F" (fighter) AC class in the USA, while in the USSR, and later in Russia and Ukraine, this AC class has been subdivided into three self-contained aircraft subclasses: fighters proper, fighter-bombers and front bombers. Understandably, any multipurpose, multifunctional item (unit, device, aircraft) is inferior to a specialized one, whose efficiency is optimized for a specific purpose. In AC development, insisting on multifunctionality leads to unnecessary risks and is justified only under certain specific conditions, namely:

- a) technical and operating requirements to the AC which are to be functionally joined are similar, while the weight of avionics that expands aircraft functionality is small as compared to the AC weight;
- b) there is a considerable technological superiority over the enemy, which is likely to remain during the AC life cycle;
- c) lack of funds makes the development of specialized airplanes impossible;
- d) there is a need to reduce the number of aircraft, simplify their operating and servicing requirements.

If a country's aviation technologies are at a reasonable level of development and the economy is healthy, it is hardly sensible to opt for multifunctionality.

**The low-altitude strategic bomber** is another example of multi-mode AC. After the failures [30] of the Convair B-58 Hustler and North American XB-70 supersonic high-altitude bombers and multipurpose General Electric F-111 aircraft (ref. Table 11.1) programs, Rockwell International finally implemented the new concept of strategic attack aircraft and produced the B-1B variable-sweep wing aircraft (Fig. 11.21, ref. Table 11.1). Now these AC are the most numerous<sup>1</sup> US strategic bombers.

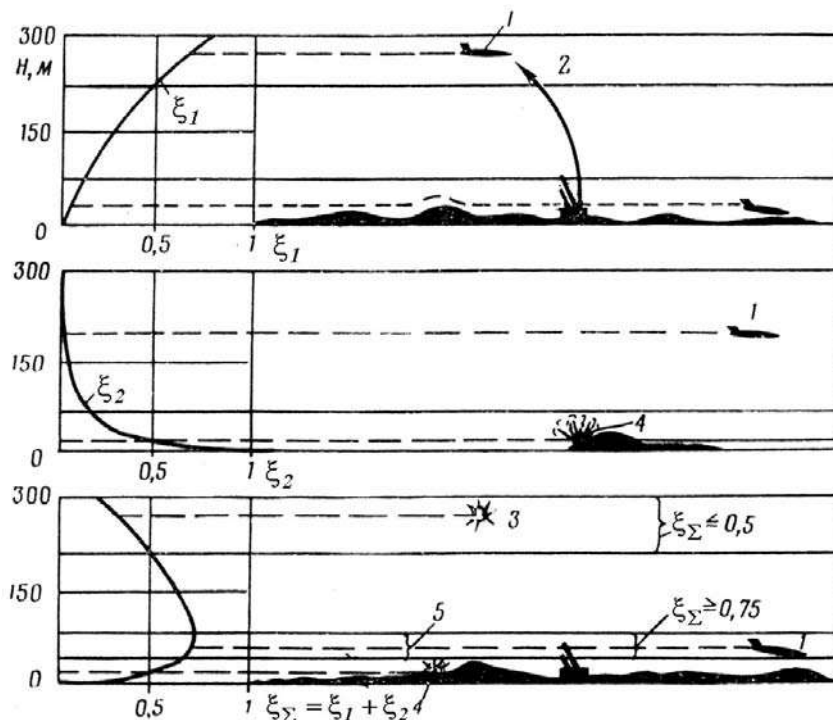
The analysis of anti-aircraft defence effectiveness shows that the probability of AC being destroyed by enemy anti-aircraft artillery or missiles rises dramatically with the growth of flight altitude (Fig. 11.22), and for an AC flying at an altitude of more than 300...400 m approaches 100%. Consequently, a new strategic aircraft generation was oriented towards low-altitude flying. However, when AC are flying at altitudes lower than 50 m at transonic or supersonic speeds, they are in danger of ground collision. Actually, for an AC to perform its combat mission with a reasonable probability of success, it should fly at a high speed within a narrow «safe passage corridor» of altitudes (80...150 m). As flying in the dense layers of atmosphere entails dealing with a high aerodynamic drag and,



**Fig. 11.21. Rockwell International B-1B Strategic Bomber**

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<sup>1</sup> At the beginning of 2003 the US Air Force had 83 B-1B, 74 B-52H aircraft and 21 B-2A airplanes.



**Fig. 11.22. Low-Altitude «Safe Passage Corridor» (flight altitudes allowing to perform a combat mission with a reasonable probability of success):**

- $\xi$  — probability;  $\xi_1$  — probability of AC being destroyed by the enemy anti-aircraft defence;  $\xi_2$  — probability of ground collision,  $\xi_\Sigma = \xi_1 + \xi_2$  — the total probability of aircraft loss; **K** — «safe passage corridor»; **H** — flight altitude

as a result, high fuel consumption, the AC and its power plant should be able to follow an economical mode of long-range, high-subsonic-speed flight at normal altitudes. The standard multimode flight profile (called **low-altitude spurt**) is as follows:

- long-range flight at an altitude of 9–11 km to reach enemy anti-aircraft defence zone or, alternatively, long-term patrolling near this zone);
- high-speed low-altitude spurt to the target (terrain-following flight mode);
- leaving the anti-aircraft defence zone (same mode);
- home-coming (economical mode).

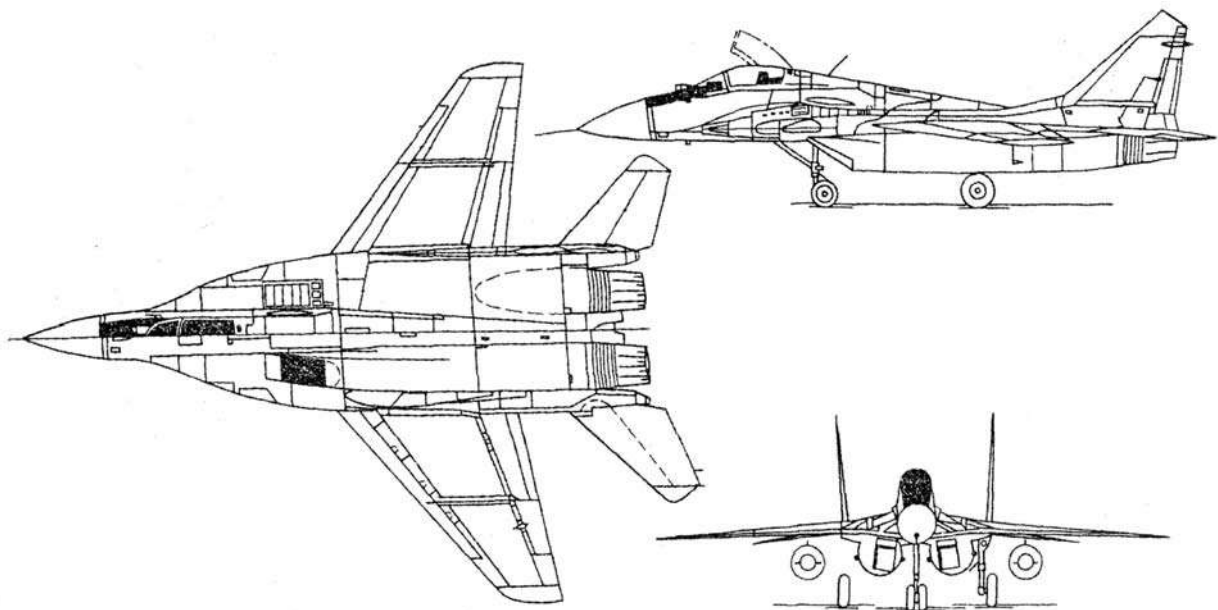
Employing a variable-sweep wing was the only possible way to ensure aerodynamic characteristics necessary for this flight profile. Outer wing cantilevers, hinged to the wing centre section, could be turned by a special hydraulic drive from 5 to 67.5°, the sweep angle at super-low altitudes being 55°. Naturally, air intakes and other power plant units are supposed to operate efficiently under all flight conditions.

## 11.8. Power Plants for Highly-Manoeuvrable Aircraft: Design Philosophy

Designers of power plants for highly-maneuvrable aircraft have to address problems similar to those accompanying the multimode AC. However, in their case to ensure the reliable operation of all power plant components is even more difficult, since the set of varying flight parameters also includes the angles of attack and slide, g-loads and, occasionally, even the direction of flight.

The concept of highly manoeuvrable fighters, also called **dog fighters**, is an example of how these complicated problems were successfully solved. It was developed after analyzing the experience of Vietnam War, when supersonic AC pilots lowered their supersonic speed to trans-sonic levels, to make manoeuvring possible, and fought in the World War II manner, that is, performing all kinds of combat turns and figures. American pilots and military experts had a special name for that: dog's fight. As fighter-bombers available at that time were badly suited for the purpose, military strategists worked out a new concept of a fighter — «dog-fighter», which technically meant **a fighter optimized for high manoeuvrability in air fighting**.

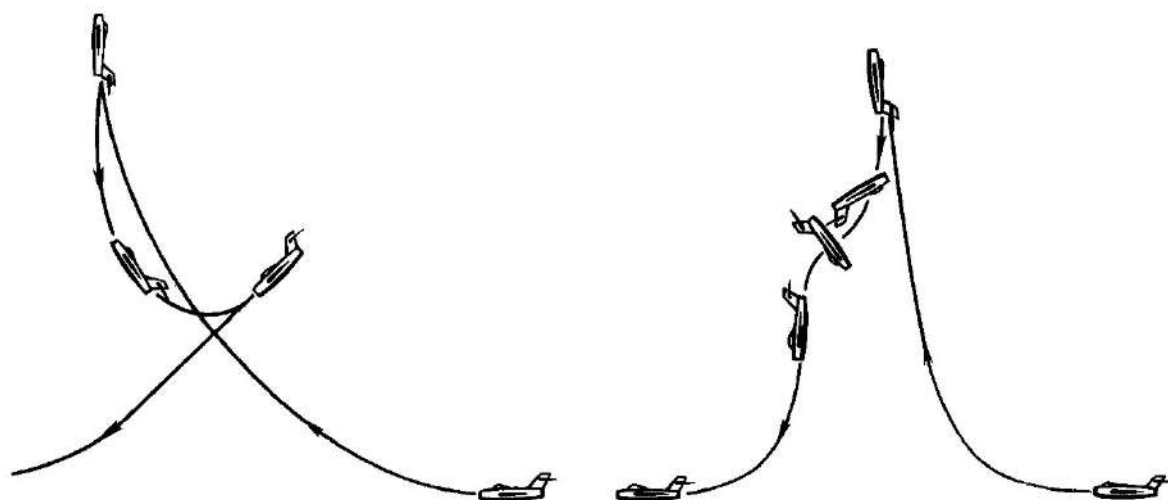
The concept found its implementation in the Russian MiG-29 fighter (Fig. 11.23, ref. Table 11.3), designed by the ANPK MiG. Among the innovative features of this aircraft are the following: integrated aerodynamic layout, active system of artificial stability, electronic scanning radar station that allows to trace 10 targets simultaneously, ability to aim up to 4



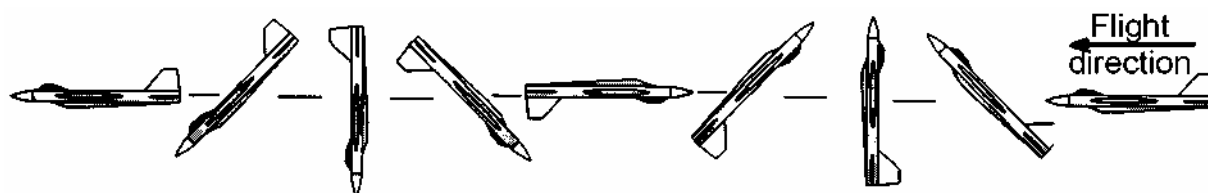
**Fig. 11.23. MiG-29 Fighter Optimized for High Manoeuvrability in Air Fighting**

guided missiles and to fly in the terrain-following mode. The outstanding quality and optimum operational coordination of its airframe, power plant, avionics and armament performance account for the AC superiority over existing analogs. The MiG-29 fighter is exported to more than 20 countries.

In the USA, the dog-fighter concept was further transformed into the concept of a **superiority fighter**, that is, «super-fighter», the most state-of-the-art fighter at the current level of air science, technology and production techniques (the Russian equivalent is a **fighter for winning air superiority**). The concept was realized in the McDonnell-Douglas F-15 Eagle operated by the US Air Force and the Grumman F-14 Tomcat operated by the US Navy. Their Russian counterparts are the family of the Su-27 — Su-37 fighters (ref. Fig. 11.20, ref. Table 11.3). The development of these fighters required unprecedented scientific, technological and financial resources which other countries had not been able to pool. Superfighters surpass the latest models of other aircraft in maneuverability and thrust-to-weight ratio; are armed with long-range missiles; have better on-board equipment, avionics and other advanced features. The complexity of designing a power plant for this AC may be illustrated by one example: in some cases the super-fighter flies tail forwards (Fig. 11.24, 11.25), while retaining full controllability and a steady operation of the engines.



**Fig. 11.24. Two Versions of «Bell» mAneuver  
(first performed by Su-27 superiority fighter)**



**Fig. 11.25. «Chakra» Maneuver  
(first performed by Su-35MKI superiority fighter)**



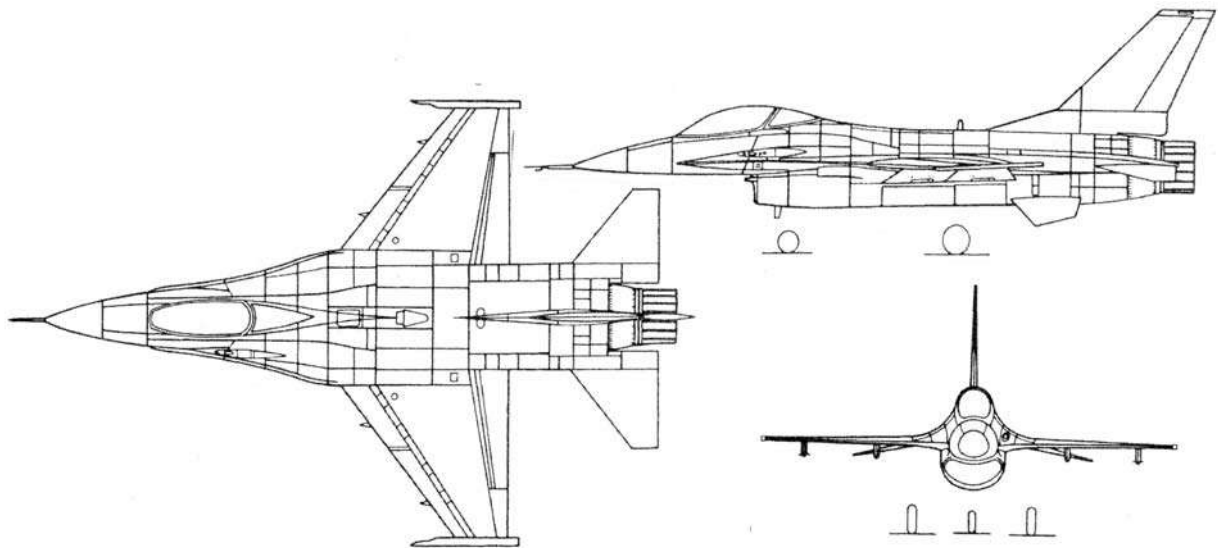
## 11.9. Power Plants for Super-cheap Fighters: Design Philosophy

In spite of the colossal expenditures on super-fighters, their creation did not solve all of the US military problems. The McDonnell-Douglas F-15 Eagle proved to be an excellent but over-priced fighter, its real cost being \$30 million against the planned cost of \$2-4 million; the F-14 Tomcat is armed with a long-range missile that costs \$ 600,000. The US military and political authorities acted as usual: super-fighters were put into commercial production; their showy photos could be seen in all kinds of magazines; but at the same time Congress approved the funding of another fighter program. Both financial and military considerations lay in the foundation of program nicknamed by the media «**the concept of a super-cheap fighter**», although it was officially called «**the concept of a light-weight fighter**» (LWF) or «**the concept of a day-time fighter**». The fighter was conceived as an AC that would combine good operating and tactical characteristics with reasonable costs. Light-weight fighters were to be used together with superiority fighters (two-three F-15s + five-eight LWFs) to ensure the reliable defence of the US air space and the US air superiority in the regions of its strategic interests.

In the competition of five leading UC aircraft companies General Dynamics Inc. won with its YF-16 project. To minimize design costs, GD specialists had borrowed a lot of engineering solutions from other AC (General Electric FB-111, Lockheed SR-71, Northrop F-5A, McDonnell-Douglas F-15, Grumman A-6 Intruder, A-7 Corsar II) and later did without building the AC mock-up model and prototype for static ground tests. The General Dynamics F-16 Falcon aircraft (Fig. 11.26, ref. Table 11.3) was initially priced at about \$5.16 – \$6.09 million. However, in 1977, allowing for inflation, the cost rose to \$9.3 million per unit: airframe and its systems: \$2.77 million; engine: \$1.2 million; radar equipment: \$0.37 million.

The F-16 is a unique aircraft that successfully balances two seemingly incompatible characteristics: a comparatively low cost and relatively high flight performance, especially at speeds 0.6–1.2 M and altitudes up to 7,000 m. These impressive results were obtained by the application of light-weight composite materials, miniaturised avionics, automatically controlled wing high-lift devices, high thrust-to-weight ratio, etc. 1388 aircraft were produced for the US Air Force, 160 for Iran and 75 for Israel. Having outperformed «Mirage F1E» and «Viggen», the F-16 aircraft was awarded the so-called «contract of the century» for 348 aircraft to replace the F-104 in several European NATO member-countries.

One should have no doubts concerning the relative nature of the terms *low-cost* and *easy-to-use aircraft*. What is low-cost and easy-to-use



**Fig. 11.26. General Dynamics F-16 Falcon Light-Weight Fighter**

for the USA and Western Europe may prove to be expensive and over-complicated for poorer and less technologically advanced countries. For example, Pakistan canceled its contract for 48 F-16 fighters after obtaining the first few units and purchased the MiG-21 aircraft of Chinese production instead.

Obviously, the development of low-cost and easy-to-use fighters did not detract from the concept of superiority fighter. After the F-14, F-15 and Su-27 had been built, there appeared a demand for the fighter with not just high, but super-high, manoeuvrability. The designers of the previous generations of fighters had had to reduce the AC stability to the minimum. The **concept of super-manoevrability** required that the principle of unstable centre-of-gravity positions should be implemented. Highly sophisticated active control systems were developed to address the issue of instability by creating artificial stability. Power plants with a controlled thrust vector (Fig. 11.27), twin-engine layouts and engines located separately along the AC longitudinal axis increased the manoeuvrability (especially at the supercritical angles of attack), reliability and invulnerability of the AC. Power plants based on these innovative principles enabled the aircraft pilot to assume independent control and handle the stabilization system.

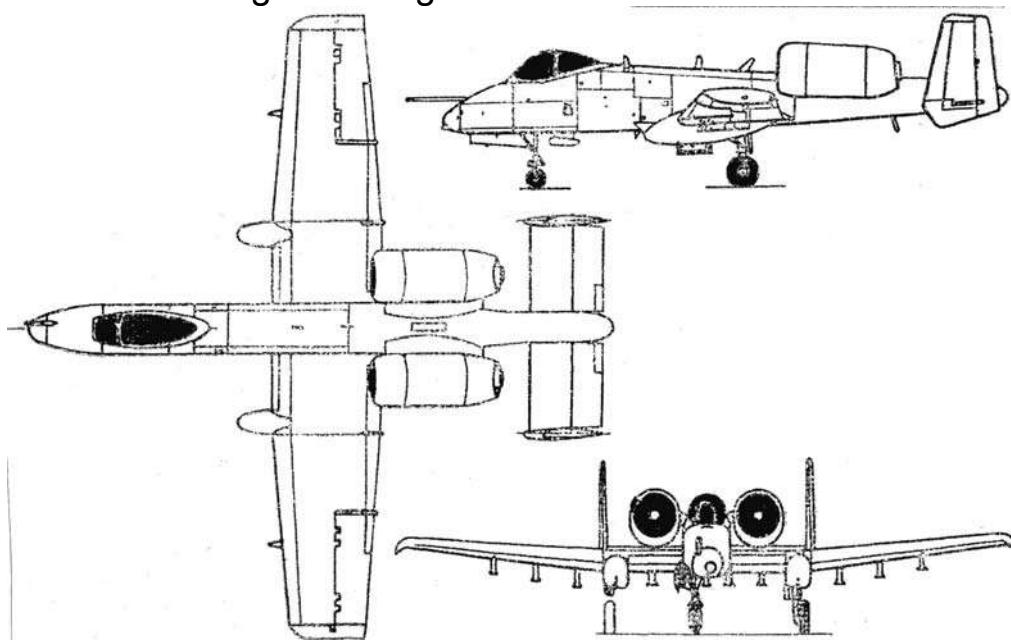
### **11.10. Power Plants for Aircraft of High Reliability, Invulnerability and Survivability: Design Philosophy**

A crucial requirement to be met by any modern aircraft, both military and civil, is high standards of reliability, survivability and safety of flights. Power plants are under special attention, because any damage sustained by the power plant components in hostile actions or their operation-induced impairments may cause not only the frustration of the AC flying mission but also a fire or explosion on board.



**Fig. 11.27. Su-37 Fighter Power Plant with Deflectable Thrust Vector**

The A-10 Thunderbolt II attack aircraft (Fig. 11.28, ref. Table 11.2) designers found an effective way to raise the standard of the AC combat survivability. Curiously enough, the fundamental design principles of the A-10 were strikingly similar to those of its Soviet predecessor, the IL-2, but the design was considerably upgraded to engage the state-of-the-art technologies of that time, e.g., strong armoring (the A-10 pilots are seated in a 680-kg titanium «bath» with a 38-mm thick floor); landing gear with semi-retractable wheels (for emergency wheels-up landings); a comparatively simple, cheap and maintainable structure; powerful arms (30-mm caliber, 2,100–4,200-shots-per-minute gun and 1,174 shells of ammunition); externally placed payload of 7,260 kg on 11 pylons. The A-10 power plant had the following advantages:

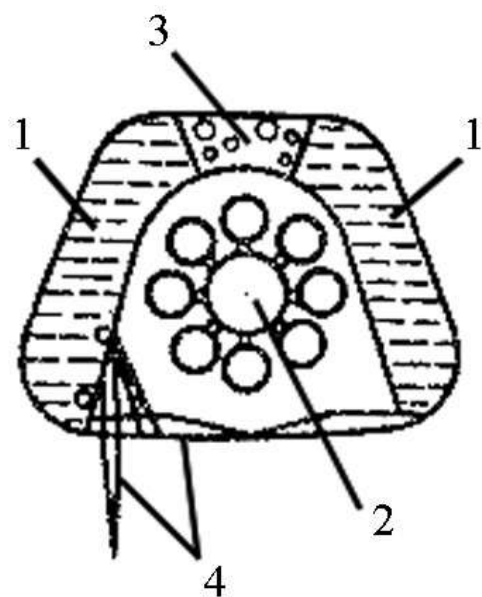


**Fig. 11.28. Fairchild Republic A-10 Subsonic Attack Aircraft**

1. The AC layout ensures high standards of its power plant invulnerability: engine nacelles are shielded by the wing, fuselage and tail unit from almost all probable angles of sight.
2. The arrangement of engines on pylons prevents fire spreading when one engine is damaged. The AC is able to fly, fulfill its mission and land with one operable engine, having lost one horizontal stabilizer console (with the fin).
3. A large fuel feedtank is arranged inside the main tank (and thus is screened by the main tank fuel); its capacity provides for 20 minutes of flying, which is usually enough for the AC to disengage and return to the base.
4. The AC maintainability and operational efficiency are ensured by the interchangeability of the left-hand and right-hand engines, air intakes and nozzles. The between-flight servicing takes 30 min. (including the loading of 6 bombs and full gun ammunition); the time of engine replacement is only 30 min. 713 aircraft were manufactured in 1975–1984.

The B-1A aircraft is first and foremost characterized by a high level of invulnerability. As a rule, front-line bombers and attack AC operate under fighters' escort, but no fighter can escort a strategic bomber over the length of its flight. So, numerous engineering solutions were worked out to protect the AC power plant from damage. In particular, due to the excellent fire safety characteristics of the AC fuel system, fuel tanks were used for screening armament compartments (Fig. 11.29) and other vital units. The B-1A invulnerability is almost unbelievable: the AC can continue operating after it took *two hits into the same system*. Three hits into the same system lead to the battle mission failure, but the AC is able to continue the controlled flight and return to the base.

**Fig. 11.29. Screening Armament  
Compartment with Fuel Tank  
(B-1A fuselage cross-section)**  
1 — fuel tank; 2— cruise missile  
barrel inside the armament  
compartment; 3 — control circuit  
and other system linkages;  
4 — armament compartment  
doors



### 11.11. Power Plants for Low-Visible Aircraft (Stealth Technology): Design Philosophy

In the mid-1980s, anti-aircraft defence systems became so effective that the probability of a combat mission accomplishment by airplanes and helicopters acting in the enemy anti-aircraft defence zone became inadmissibly small. For example, according to some media sources, the C-300 anti-aircraft defence system can shoot down objects flying at a speed of 0–3000 m/s ( $M \approx 10$ ) and altitude of 25–30 km by missiles able to maneuver with the g-load  $n_y = 60$ . Under these conditions, neither the most perfect fuselage, power plant and automatic control system design, nor the most advanced tactical modes, nor the heroism of the crew can save the AC. On the other hand, there are a lot of military missions that cannot be executed without penetrating the enemy anti-aircraft defence zone. Thus, the progress of anti-aircraft defense systems brought about a new stage in AC development: the concept of **low-visible aircraft**, or **stealth technology**.

At the initial stage of the program<sup>1</sup>, a task was set to design aircraft characterized by the minimum possible visibility of all kinds: low-observable, first of all, by radars; then by infrared, acoustical, video-optical, etc. detectors. Understandably, there is a limit to invisibility: an AC cannot be less visible than such natural objects as large birds (pelicans, storks), flocks of 5–6 medium-size birds (crows, sea-gulls), clouds, etc.

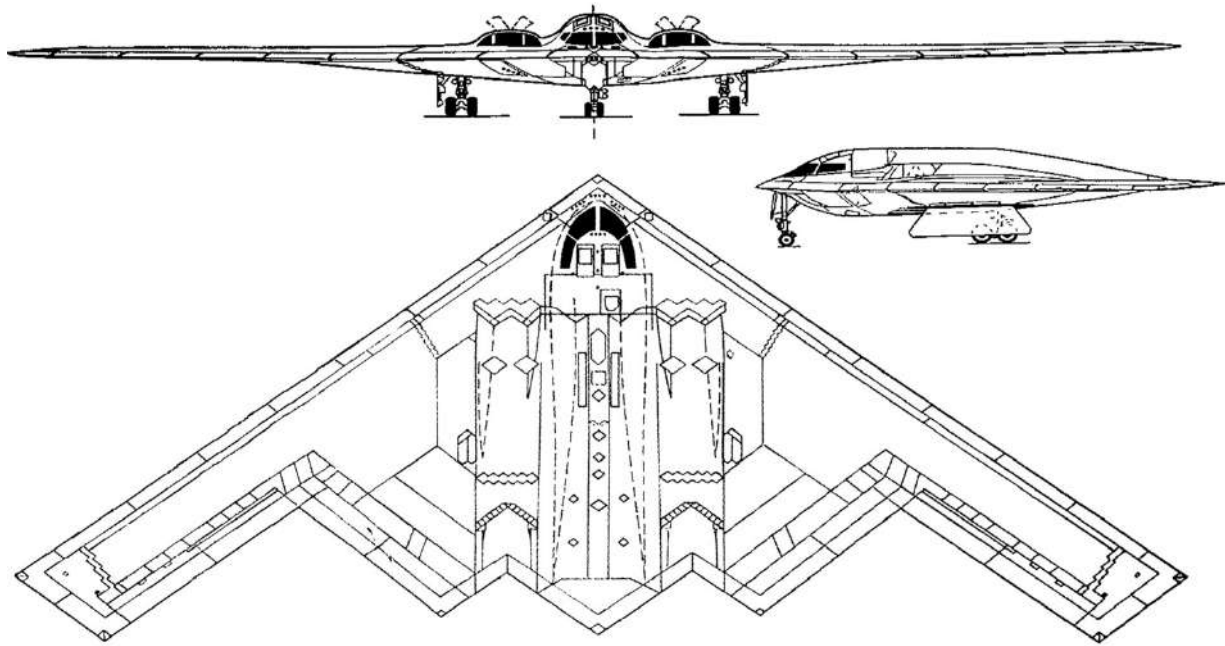
At the cost of huge financial resources and intellectual efforts, two «classical» low-visible aircraft were designed in the USA: the Northrop B-2 Spirit strategic strike aircraft (Fig. 11.30, ref. Table 11.1) and Lockheed F-117 tactical strike aircraft (Fig. 11.31).

The state of then existing scientific and technological development placed restrictions on the stealth program: only subsonic AC with the low standards of manoeuvrability, stability and controllability could have been made low-visible. The following figures may illustrate the scope of the stealth program: the Lockheed F-117 program cost \$ 6.56 billion, of which \$2 billion was spent on fundamental research; the cost of one commercially produced AC was \$ 111 million. 24,000 hours of wind tunnel testing made the B-2 the most tested subsonic AC in history, while the cost of one B-2 exceeded \$ 1 billion.

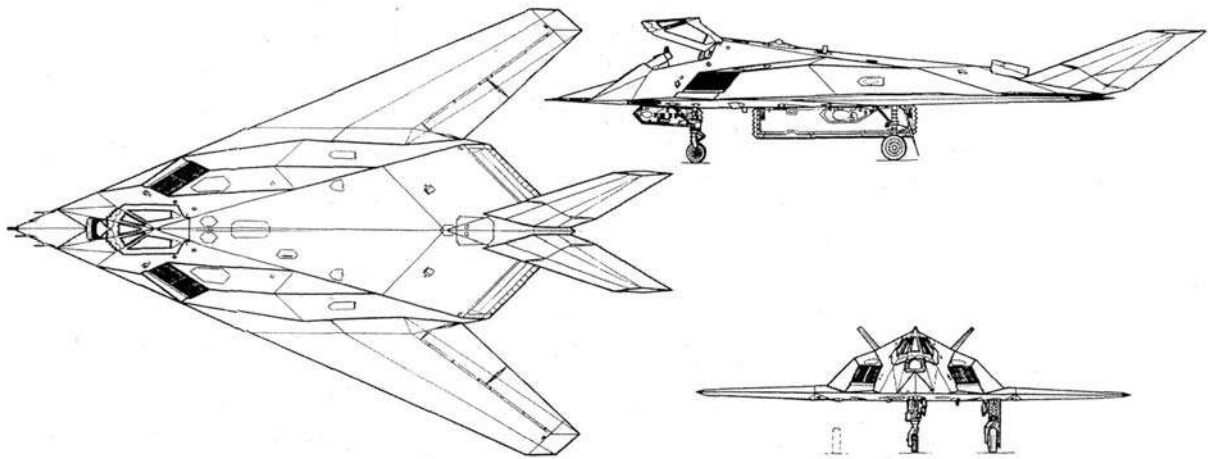
Designing power plants for low-visible AC is of special interest because an AC's radar, infra-red and acoustic visibility, as well as its inversion-tracking, mainly depend on its power plant characteristics. Let us

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<sup>1</sup>The next stage will be considered in chapter 11.3.



**Fig. 11.30. Northrop B-2 Spirit Strategic Strike Aircraft**



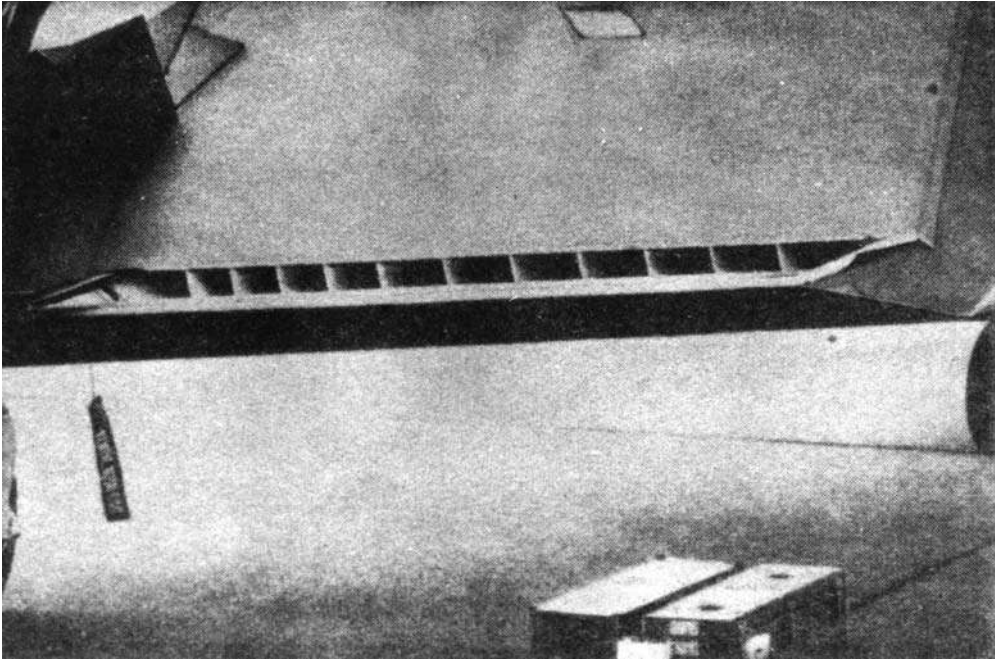
**Fig. 11.31. Lockheed F-117 Night Hawk Tactical Strike Aircraft**

look at the following examples<sup>1</sup>:

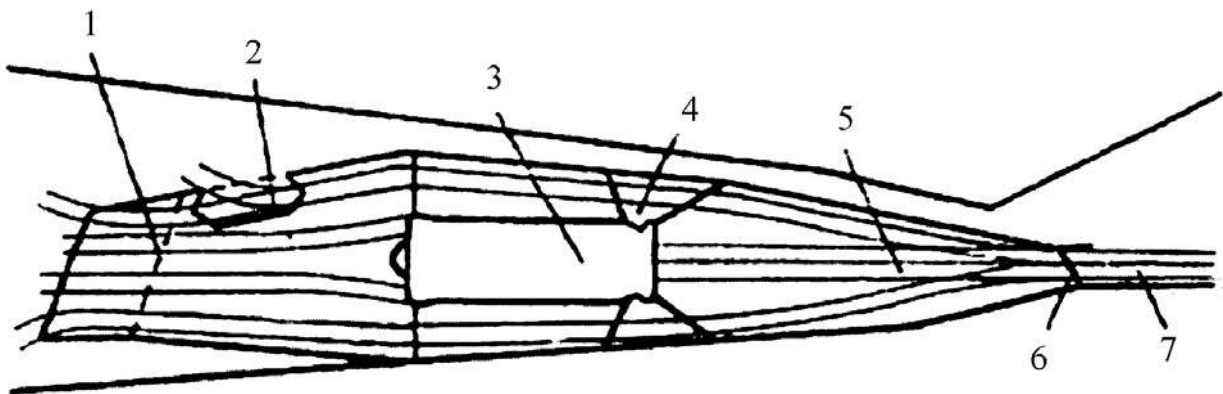
- The F-117 air intakes are permanently screened with thin-walled, composite material grids (cell size: 3.15×2.15 cm). To prevent the grids from icing and to reflect radio-waves, the air intake is coated with a layer of conductive paint; when an electric current passes through the layer, it heats the grid and prevents ice formation. In the Lockheed F-22 Raptor multi-purpose fighter, titanium grids cover air intake openings; the grid is formed by hundreds of precisely positioned openings cut by a hydroabrasive jet.

<sup>1</sup> Refer to [31 pp. 255-306] for detailed information.

- A radar-absorbent material covers not only the grids but also the intake-engine air duct walls.
- Stealth technology experts suggested making air intakes flat, with the maximum width/height ratio, and pressed firmly to the upper wing surface. The recommendation is at variance with the standard arrangement of power plant components. Air pressure is known to be reduced on the upper wing surface, while conventional air intakes are designed so as to ensure the maximum engine inlet air pressure. Usually air intakes (even side ones) are arranged at a distance from the fuselage or another adjoining unit to drain off boundary layers and to prevent vortexes from getting into intakes. However, these are considered to be issues of minor importance as against the conceptual task of the AC minimum radar visibility. In the B-2 design the problem is solved as follows: the air pipe of the so-called tunnel, or S-shaped, intake goes from the upper surface of the fuselage-wing unit into the fuselage at a distance that considerably exceeds the engine radius, thus eliminating the direct reflection of radar beams by compressor blades.
- The output units of stealth AC are designed as narrow flat-shaped nozzles; jet streams from nozzles disperse quickly, mix with the surrounding air and ensure the AC's minimum infra-red visibility. For example, each F-117 engine has a corresponding 1.65×0.12 m nozzle-slot (Fig. 11.32). The off-takes, which remove gas jets from the engines, are designed as pipes with a variable cross-section: its shape varies from the initial circular one to a rectangle with an indicated area. Each nozzle is divided into 12 sections by 20-cm-long vertical walls. The overhanging lower edges of the nozzle are made of a titanium alloy and are covered with heat-resistant ceramic slabs (similar to those used in the Space Shuttle aerospace vehicle). The walls are used both to provide the structural stiffness and to reduce a scan sector from the aft hemisphere, from which the heated elements of the engine are directly visible.
- To reduce the F-117 infrared visibility, the amount of air sucked through its air intakes is much larger than it is required for engine operation. A major portion of cold air is separated behind the intake (Fig. 11.33), flows through by-pass channels inside the fuselage (past the engine), mixes up with the jet stream of incandescent exhaust gas in an ejector, and gets to the slotted nozzle.
- The emergence of small perpendicular surfaces on the butt of deflected controls considerably increases the AC radar visibility, so designers work to minimize the number of control surfaces. For example, the B-2 (ref. Fig. 11.30) has neither horizontal nor vertical tail units. The controllable nozzle deflectors are arranged behind, with a large arm to the AC centre



**Fig. 11.32. Lockheed F-117 Engine Nozzle-Slot**



**Fig. 11.33. Air-Flow Passage in Lockheed F-117:**

**1 — air intake; 2 — additional air scooping blisks; 3 — engine; 4 — ejector; 5 — slotted nozzle passages; 6 — slotted nozzle; 7 — slotted jet stream**

of gravity. Each deflector arranged on the side of the nozzle consists of two sections with hinge axes forming an angle with the direction of exhaust gas outflow. If four sections of deflectors, in couples, deviate in the same direction, they create a pitch moment; if they move in the opposite directions, a roll moment is created. Each section is able to create a yaw moment due to the tilt of hinge axes. In case both sections of each pair of deflectors deviate in the same direction, their moments cancel out; in case both sections of each pair of deflectors deviate in the opposite directions, the pitch moment is cancelled out and the yaw moment appears.

The success of the stealth program was proven in one of the recent armed conflicts, the Desert Storm operation (Gulf War). Table 11.6 com-



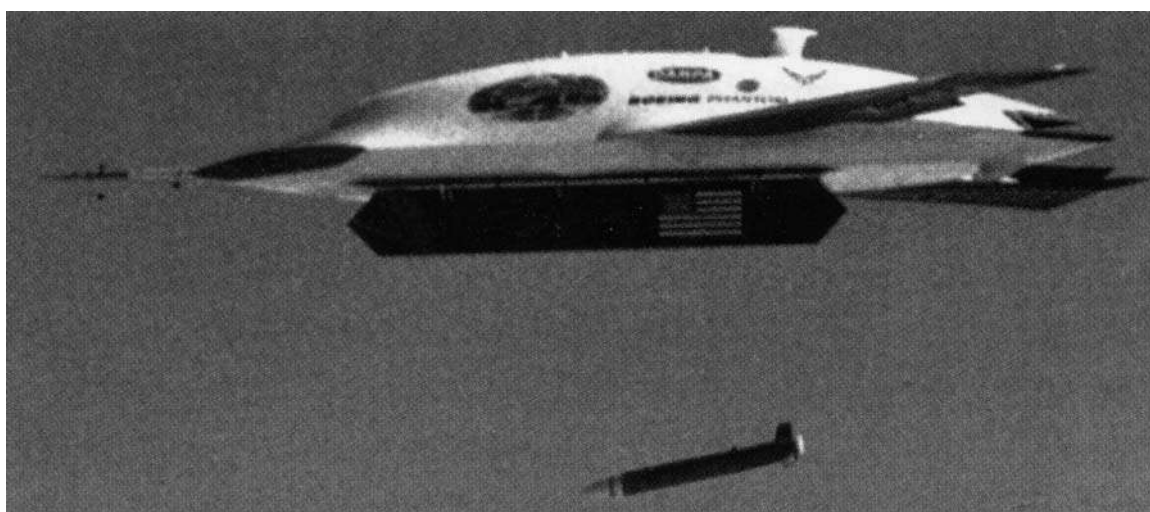
compares the effectiveness of the F-117 Night Hawk aircraft equipped with high-precision weapon and the previous generations of AC. According to some media sources, during the First Gulf War, 45 F-117 airplanes flew 1,272 sorties totaling 6,905 hours, which constituted about 1% of the total number of multinational force sorties and less than 4% of combat sorties. They destroyed about 40% of all strategic ground targets in the zones of advanced anti-aircraft defence. Different sources quote the US AC losses from zero to three items (the first figure appears to refer to direct loss due to AC detection and shooting down; and the second one shows incidents in which the F-117s were casually spotted by Iraqi fighters of the Soviet production).

Table 11.6

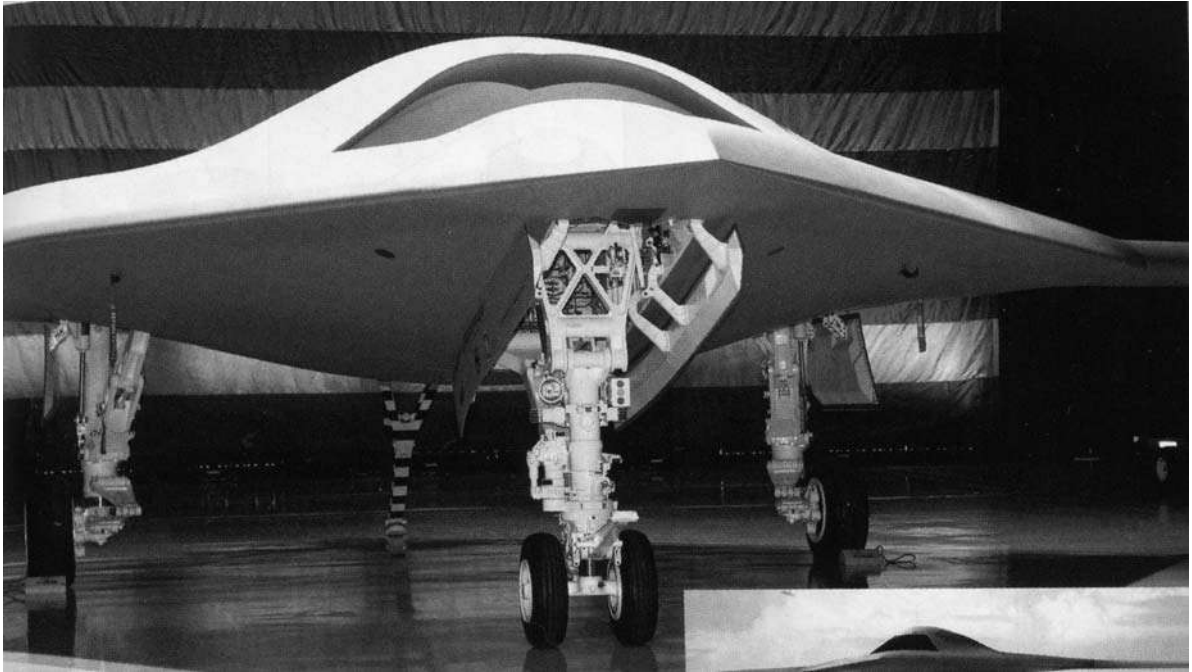
**Effectiveness of aircraft of various generations**

War	World War II	Vietnam War	(First) Gulf War
Aircraft type	B-17	F-105	F-117A
Number of combat sorties	4500	95	1
Number of bomb drops	9000	190	1
Precision of aim, m	3000	300	0,3

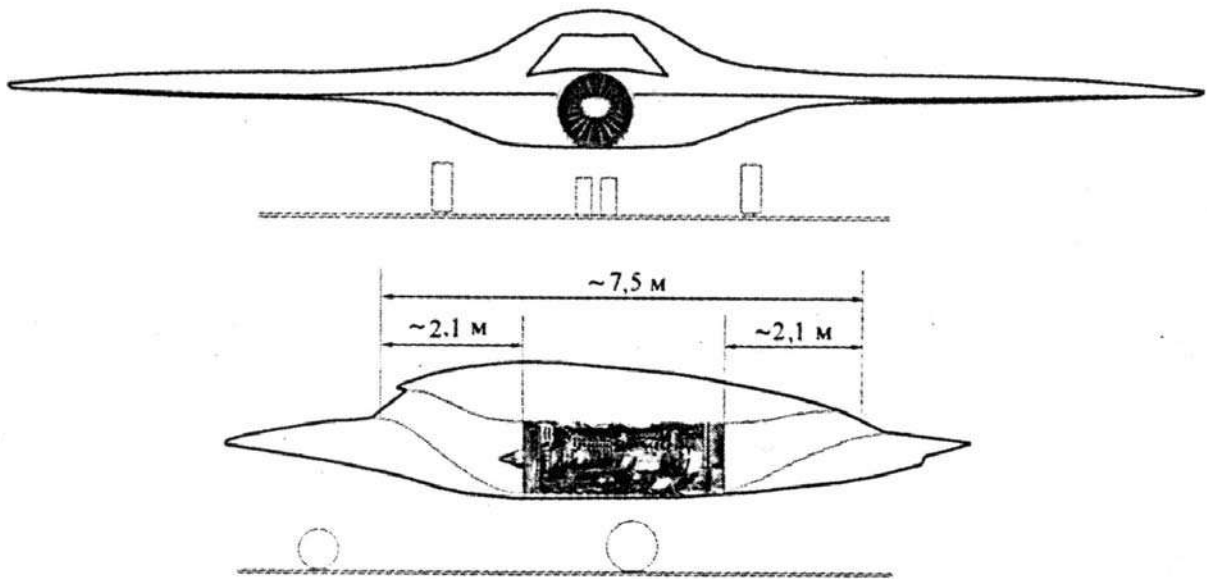
The next stage of attack aircraft development has been focused on designing low-visible unmanned air vehicles based on recent advances in microelectronics, nanotechnology, automatic control systems and engine design. First experimental efforts (e.g. the X-45A, X-47A programs, Fig. 11.34) demonstrated high efficiency of this new armament. Now US specialist begin the flight tests of X-47B strike vehicle (Fig. 11.35, 11.36, ref. Table 11.1) with take-off mass 20250...24000 kg, payload 1800 kg and flight time 12...14 h. (up to 100 h with refueling). It is equipped with by-pass engine F100-PW-2220U. X-47B test flight program will continue up to 2013.



**Fig. 11.34. Boeing X-45A Low-Visible Experimental Unmanned Air Vehicle**



**Fig. 11.35. Boeing X-47B Low-Visible Unmanned Strike Air Vehicle**

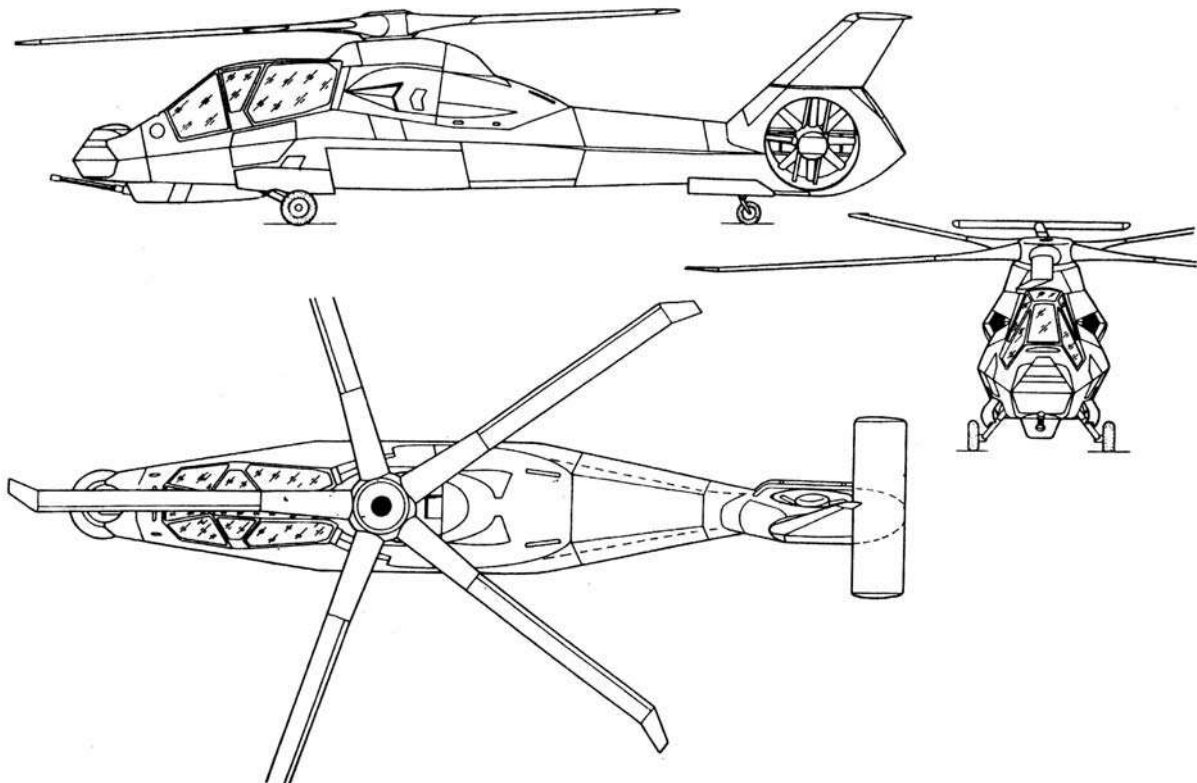


**Fig. 11.36. Boeing X-47B Engine Arrangement**

Stealth technology was also used in the design of the RAH-66 Comanche low-visible attack helicopter (Fig. 11.37, ref. Table 11.2). The helicopter was to satisfy an unprecedented set of requirements: low radar visibility, the capability of fighting against enemy helicopters and fighters, and – initially – its speed was supposed to exceed that of any serial helicopter by 180 km/h (the latter requirement was subsequently renounced as impracticable and replaced by the flight speed of 350 – 380 km/h). The association of Boeing Helicopter and Sikorsky (USA) won the contest to design a low-visible helicopter that was designated RAH-66 Comanche.

To answer the purpose of low visibility the RAH-66 helicopter power plant has the following design features:

- radar-absorbing coating of all of the power plant outside surfaces;
- main-rotor irregularly shaped hub covered with a smooth fairing;
- hingeless rotor blades made of radiolucent composite materials;
- tail screw in the ring channel («fantail») arrangement, which reduces radar and acoustic visibility and increases operating safety;
- exhaust gas blow-out through a slotted nozzle on all length of the tail beam; rapid mixing with the ambient air assisted by the main rotor airflow.



**Fig. 11.37. RAH-66 Comanche Reconnaissance-Attack Low-Visible Helicopter**

As a result, the RAH-66 has as much as 600 times (!!!) the radar invisibility of the AH-64 Apache of the same size and 200 times of the Bell OH-58 light helicopter. The RAH-66 infra-red visibility is 60 % less than AH-64's. In fact, the US designers pin all their hopes on the low visibility of the helicopter: its thin kevlar armoring cannot protect the crew even from submachine-gun bullets.

To sum up, we need to emphasize that the success of stealth aircraft designers is not only judged by how much the aircraft visibility was reduced, but by their proficiency in finding the right balance between various conflicting requirements and diverse conditions without sacrificing the AC essential performance characteristics.

## 11.12. Power Plants for Light and Ultra-Light Aircraft: Design Philosophy

Our attention to the design of large super-heavy aircraft power plants at the beginning of this section does not imply that light and super-light general aviation AC equipped with standard propeller power plants and piston engines are to be neglected.

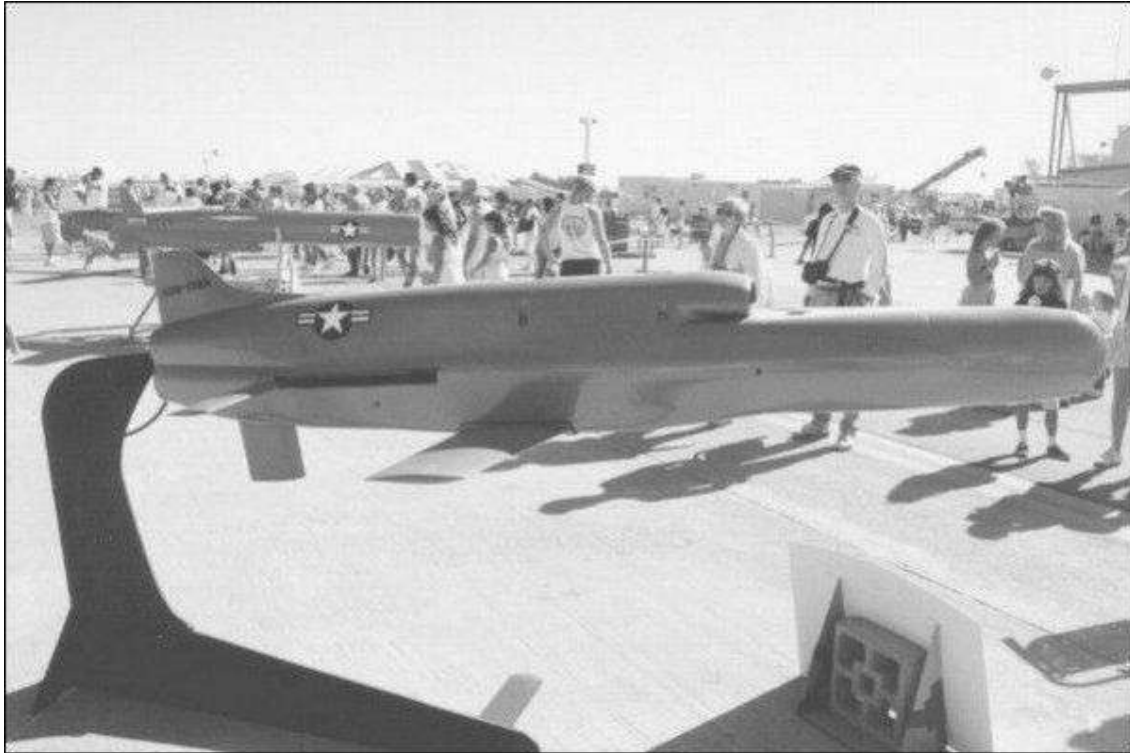
Research achievements in the areas of microelectronics, nanotechnology, automatic control systems and AC engine design have made it possible to develop the concept of the **strategic strike AC carrier of cruise missiles**. This carrier of long-range cruise missiles with high guidance accuracy operates as follows:

- it flies over neutral waters or friendly countries close to the enemy territory but outside the enemy anti-aircraft defence zone;
- in case of need it launches cruise missiles that penetrate the system of enemy anti-aircraft defence and destroy the target.

Small-size cruise missiles fly at a high speed and low altitude, are relatively invulnerable and inexpensive. For example, the AGM-86 (USA) cruise missile is low-sized (its length is 6.31 m, wing-span: 3.65 m, fuselage diameter: 0.63 m) and low-visible. Its range is 2500...3000 km, flight speed: 805–885 km/h, flight altitude: 60–150 m, and precision of aim: 10–30 m (!!!). The missile is able to carry a 0.2-megaton W-80 nuclear war-head weighing 123 kg, a chemical or electromagnetic war-head. The missile take-off weight is 1,200...1,500 kg, and it is powered by a light (60-kg) and miniature F-107-WR-100 by-pass turbojet with a thrust of 270 kg. The B-52H strike aircraft can carry 20 cruise missiles, and the Rockwell International B-1B aircraft — 22 missiles.

The by-pass turbojet F-121 (Fig. 11.38,b), designed by Williams International, is even smaller. Its thrust is 310 N only. It is intended for anti-radar medium-range missile AGM-136 (Fig. 11.38,a) (1950 N take-off weight, 2.54 m length, 1.56 m span of retractable wing). It is launched at the distance up to 90 km, rich the necessary altitude, waits for radar engaging and kills it.

The development of micro-sized jet engines and power system elements would allow to implement a long-standing and promising idea in helicopter engineering: to design a jet-driven rotor. If any engine installed in the helicopter fuselage rotates the main rotor, the fuselage is loaded with a support reaction. To balance this load, the helicopter should be equipped with a tail propeller installed on a long tail beam, which increases the structure mass and lowers the engine capacity. Another solution is to create a helicopter with two main rotors rotating in the opposite directions (coaxial, longitudinal and transversal configuration); but that



a

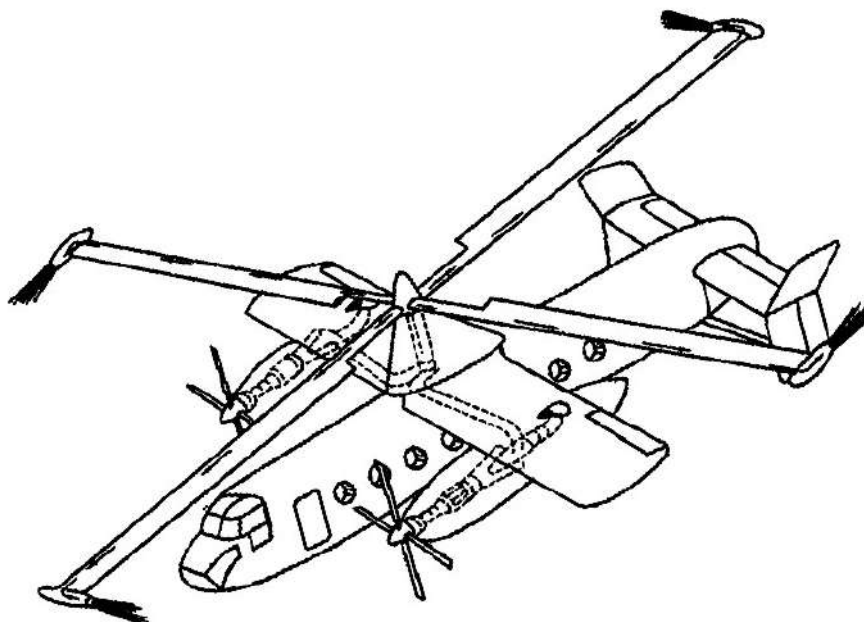


b

**Fig. 11.38. AGM-136 Anti-Radar Missile (a) and its By-Pass Turbojet F-121 (b)**

would also increase the weight of the power plant. If the helicopter employs small turbojet engines installed on propeller blade tips or with nozzles to discharge the jet of compressed air (fed via a channel located inside the blade), the engine weight effectiveness will rise considerably. However, nobody has yet been able to develop a small turbojet or bypass engine and its systems for operation under conditions of high overload, vibration and curved flow-around, although some experimental prototypes of such a helicopter (Fig. 11.39) have already been tested.

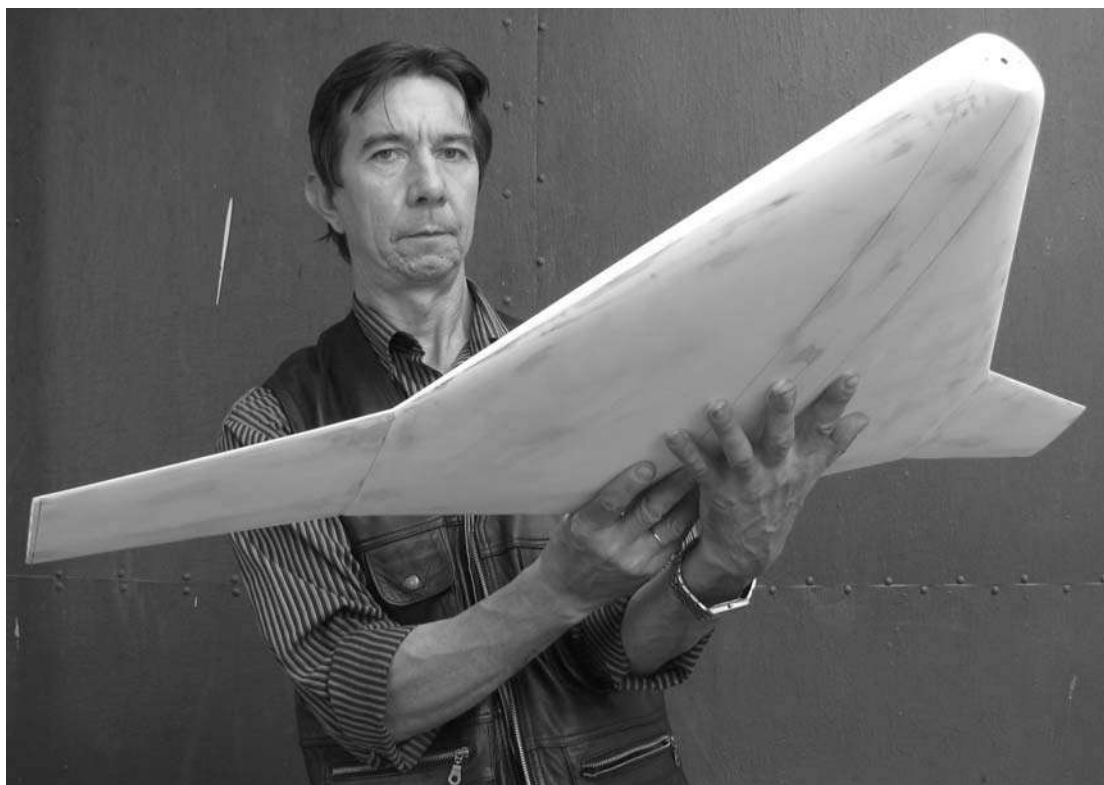
**Fig. 11.39.**  
**Configuration**  
**of Helicopter**  
**with Jet**  
**Compressor**  
**Drive of Main**  
**Rotor**



Among the latest developments, super-light unmanned aircraft, or micro air vehicles (MAV), occupy a prominent place. They are driven by multifarious power plants. Designed at the NII PFM «KhAI», the Bopobей-M light multi-purpose unmanned aircraft for day-night observation in real time is shown in Fig. 11.40. The Bopobей-M aircraft is started «by hand». Its range is 90 km, take-off weight: 2.5 kg, payload weight: 0.3 kg, length: 0.6 m, wingspan: 1.05 m. The main components of its power plant are an electric engine with a reducer and storage batteries. Its propeller is specifically designed to increase the power plant efficiency. Images from the onboard TV-camera are sent in real time to the observer's monitor, videotape recorder and other ground tools. It is possible to install two telecameras operating in optical and infra-red frequency bands or creating a stereoscopic presentation of the image.

The Fancopter light unmanned helicopter designed by the EMT Corporation (Germany) is shown in Fig. 11.41. The Fancopter is designed as an unmanned flying vehicle (UFV) with the main rotor in a ring channel, its diameter being ~500 mm, take-off weight: 750 g, flight time: ~15 minutes. The basic components of its power plant are a highly effective accumulator and an electric engine.

The above mentioned aircraft do not set a limit to AC microminiaturization. It has recently become a tradition to hold international sporting events in flying experimental unmanned micro air vehicles (MAV) whose mass is about a hundred grams and size is no more than 6 inches (~15 cm) (Fig. 11.42 - 11.44). The Black Widow ultra-light unmanned flying vehicle designed by the AeroVironment Corporation (USA) (ref. Fig. 11.42) has the following parameters: flying time: 30 min; maximum



**Fig. 11.40. The Воробей-M Light Multi-Purpose Unmanned Aircraft  
Designed at NII PFM «KhAI»**

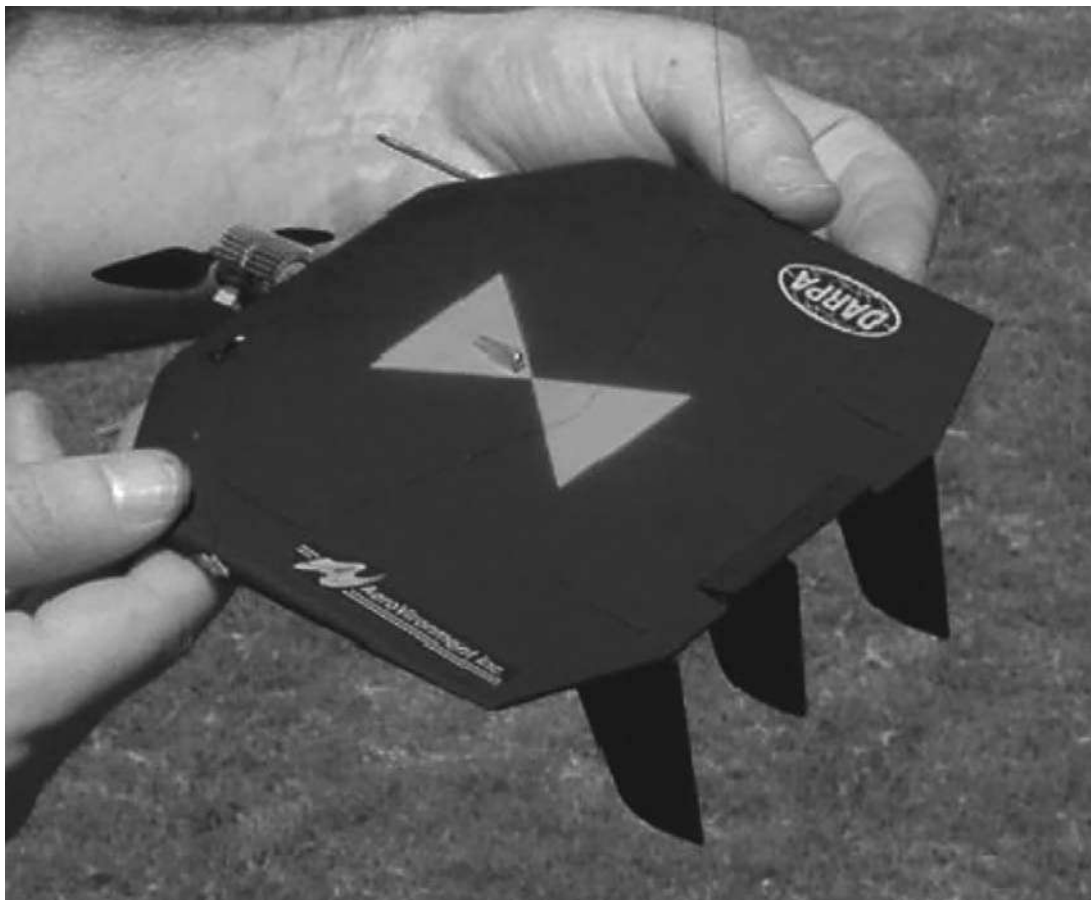


**Fig. 11.41. «Fancopter» Light Unmanned Helicopter**



flight speed: 54 km/hour, take-off weight: 100 g, length: 150 mm; wingspan: 150 mm.

There is a growing interest in designing flying vehicles equipped with moving or flapping wings (ornithopters and orthopters, or insectofopters). The ornithopter power plant imitates a bird's flight, which is a complicated process. A bird's wing operates in the auto-oscillation mode, resembling flutter in its nature; and a bird's plumage controls the boundary layer. The flight of a bird may also involve other features, probably unknown to the present-day science and impossible to copy in currently available engineering structures. An insect's flight is simpler: it moves its wings like oars and just pushes off from the air. The majority of existing ultra-light UFV with flapping wings are orthopters (ref. Fig. 11.43-11.44). The designers of orthopter power plants face a number of problems, e.g. they have to select the engine type, develop the devices which will transform an elementary (usually rotary) motion to a kinematically complicated wing motion. Highly promising in the orthopter design is the application of «chemical muscles». Provided the whole set of these complicated problems is solved, it will become possible to design flying vehicles capable of not only vertical take-off and landing, but of on-wall or on-ceiling take-off and landing as well.

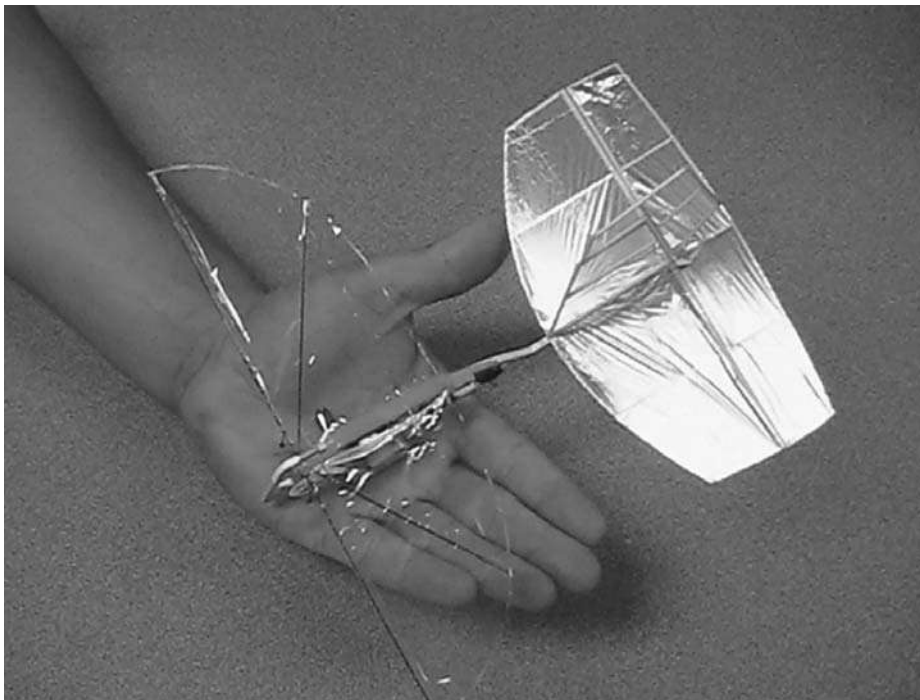


**Fig. 11.42. «Black Widow» Ultra-Light Unmanned Micro-UFV**





**Fig. 11.43. Ultra-Light Orthopter of Microbat Series**



**FIG. 11.44. Ultra-Light Orthopter of Electricfly Series**

What about manned aircraft, the minimum take-off weight has so named personal VTOL flying crafts. Some examples of this type AC are presented on Fig. 11.45. They may be equipped with jet (rocket) engine with hydrogen peroxide fuel («rocket pack» or «rocket belt» — Fig. 11.45,a, b), by-passed turbojet engine (the Williams Aerial Systems Platform (WASP), also known as the "flying pulpit — Fig. 11.45,c) and counter-rotating ducted fan inside a platform (the Hiller Aircraft Corporation VZ-1 Pawnee — US military designation, earlier Army designation: HO-1 — Fig. 11.45,d). VZ-1 was controlled by the pilot shifting his body weight around to tilt the platform.



a



b



c



d

**Fig. 11.45. Personal VTOL Flying Aircraft:**  
a, b — «jet pack» or «rocket belt» Williams Research Corporation;  
c — WASP «flying pulpit»; d — Hiller VZ-1 Pawnee direct lift rotor aircraft

### 11.13. Power Plants for VTOL Aircraft: Design Philosophy

Vertical take-off and landing (**VTOL**) features are necessary not only for ultra-light unmanned AC, but also for conventional airplanes, both military and civil. The first step in this direction was made by developing **the concept of rocket-launched** aircraft, fighters in particular. Their creation provided efficient and effective solutions to the following previously insoluble problems:

- support of battle operations in situations when the overwhelming majority of airfields and alternate landing areas are inoperative;
- support of battle operations in areas with underdeveloped infrastructure, especially where there are no airdromes;
- support of battle operations of naval forces.

In the Soviet Union, the concept of the rocket-launched fighter started being developed in view of a sudden dramatic deterioration of the Soviet-Chinese relations. There had never been any need to upgrade the facilities on the lengthy Soviet-Chinese national border because the two countries had always had friendly relations. Firstly, to reinforce the national border quickly and efficiently, a decision was made to use powerful solid-propellant booster rockets for the take-off of conventional fighters. Then they found a way to reduce the take-off distance; which makes it possible for fighters to enter combat operations after taking off from short undamaged sections of ruined runways. The accumulated experience and further efforts provided an almost ideal answer to the problem: fighters took off from short launcher rails mounted on a heavy lorry (Fig. 11.46), which could also be used for AC transportation.



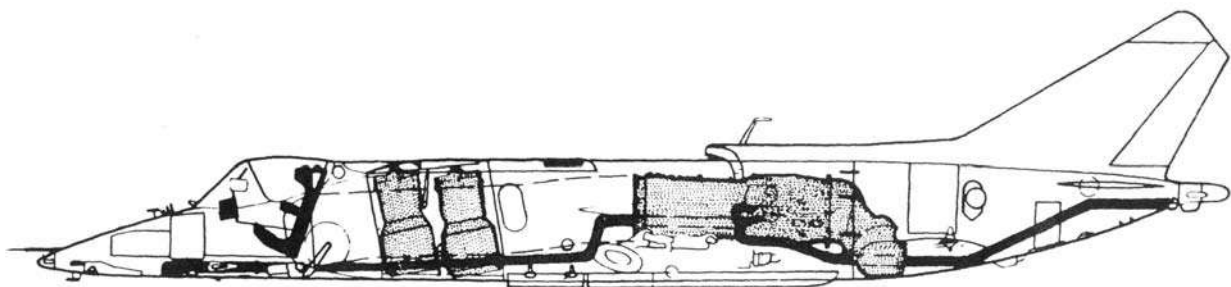
**Fig. 11.46. Fighter With Powerful Solid-Propellant Booster Rockets on Short Launcher Rails Mounted on Truck Trailer**

The next radical step was developing **the concept of vertical take-off and landing aircraft**: fighters, attack aircraft, front-line bombers, etc. As far as fighters are concerned, they are engaged in operations against ground and above-water targets, patrol aircraft, enemy deck fighters and combat helicopters. Speaking about fighters, a high level of efficiency may be provided by one of the following:

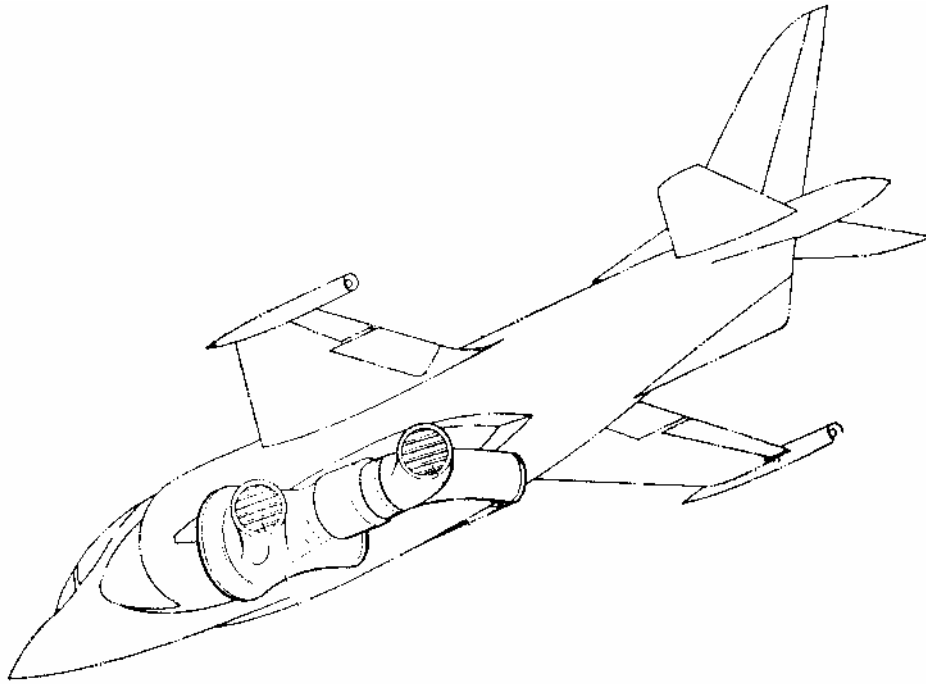
- high manoeuvrability and supersonic speed of the fighter armed with short-range air-to-air missiles and gunning;
- use of long-range high-effective air-to-air missiles, the fighter's subsonic or trans-sonic speed, as in the case of VTOL AC on fighter missions.

VTOL attack aircraft and front-line bombers are as much in demand as VTOL fighters. As mentioned above, they are used to support both land and naval forces. The navy needs protection against the action of «small-size movable objects», such as torpedo boats, missile boats, etc. Landing operations and other missions in coastal strips far from the navy bases also have to be supported from the air. Mega-operations far from own territory usually involve aircraft-carriers, which are large, expensive, unique and very few in number. So it would certainly be desirable to have a small fighter-bomber able to take-off vertically from a small free zone on the deck of a cruiser or anti-submarine ship.

Only two AC design schools were able to solve this problem: the Soviet Yakovlev OKB (Yak-38 — Fig. 11.47, ref. Table 11.3) and British Hawker Siddeley (Harrier — Fig. 11.48, ref. Table 11.3) aircraft families were created as a result. In the USA, in spite of substantial investments and the construction of numerous experimental VTOL prototypes, there have never been production models able to operate under army service conditions. Eventually, the McDonnell-Douglas Corporation (USA) accrued a licence for the British airplane, upgraded it, and started its commercial production under the name AV-8D Harrier II. The principles of vertical thrust formation in the Yak-38 and Harrier aircraft are different, but in both cases are based on the new conceptual solutions of the power plant design (for more information, refer to special literature [13, 26 — 29]).



**Fig. 11.47. Arrangement of Vertical Take-off and Landing Yak-38 Fighter-Bomber**



**Fig. 11.48. Hawker Siddeley GR.Mk1 Harrier  
Attack and Reconnaissance Aircraft of Vertical Take-off and Landing**

The foundation of success in the design of the Harrier VTOL airplane is its unique by-pass engine with four deflectable nozzles (ref. Fig. 11.48). A jet of burning hot combustion products from the inside (hot) duct flows out through two aft nozzles; a relatively cold jet from the outside duct flows out through two forward nozzles. Deflectable nozzles are able to change the thrust direction from an initial forward one to vertical (or slightly backward). The total engine thrust slightly exceeds the AC weight. During the take-off the nozzles are directed downward. After the AC has climbed to a minimum altitude, the nozzles are deflected slightly back to create a thrust vector component directed forwards and to accelerate the AC. As the AC is accelerating, the wing comes into operation. The lift created by the wing counterpoises some part of the weight, so the nozzles can be deflected further backward in order to increase the thrust component for the AC acceleration. The aircraft enters the cruise flight mode when the wing creates all the lift needed for flying. The engine nozzles become forward directed, and the further flight proceeds as in the case of flying a conventional airplane with its traditional controls (ailerons, elevators and rudders). The landing is carried out in the reverse order. It may be worth mentioning that the usual aerodynamic control surfaces are not effective during take-off and landing because of the absence or insufficient speed of the free-stream flow. To control and stabilize the flight under these conditions, the AC is equipped with an additional system of jet controls: compressed air is bled from the engine compressor and directed via

pipes to small nozzles arranged in the wing tips, nose and tail sections of the fuselage. Necessary control actions are effected by changing the value or direction of their thrust.

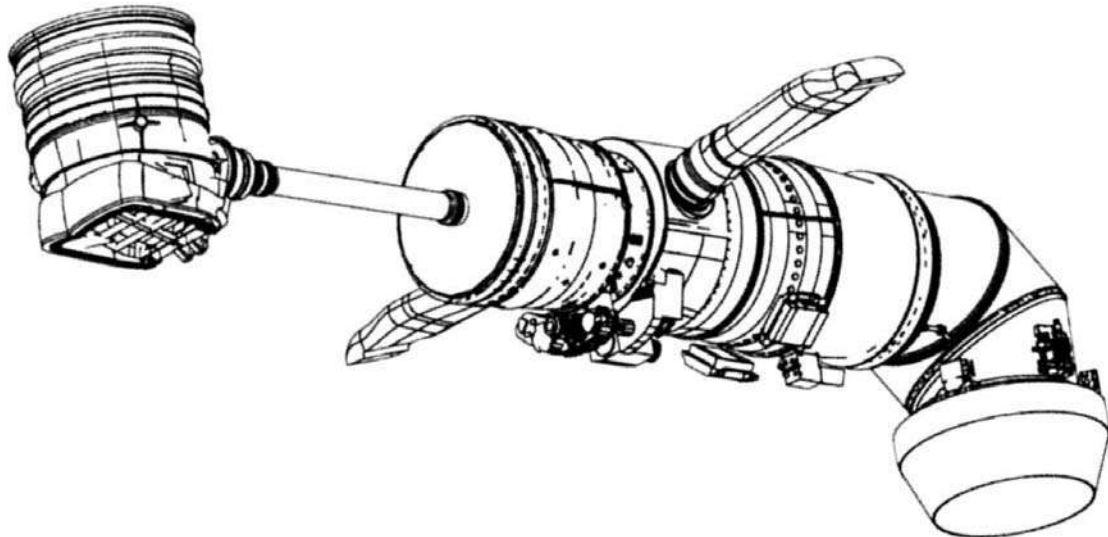
Soviet aircraft designers have taken another course. The Yak-38 aircraft (ref. Fig. 11.44) has a lift-cruise engine with two deflectable nozzles; in addition there are special lifting engines directed vertically and arranged in the fuselage compartment behind the cabin. The operating life and continuous service time of these engines are short, but they have a high specific thrust and so develop vertical thrust sufficient for a vertical take-off. After the AC acceleration these engines are turned off, the compartment doors being closed. During the take-off, landing and hovering, the Yak-38 power plant jet controls provide the AC stabilization and control. The British Harrier and the world's first VTOL supersonic AC, Russian Yak-141 (Fig. 11.49, ref. Table 11.3), use similar control principles. However, by perfecting aerodynamic characteristics and upgrading the power plant, the Yak-141 designers managed to increase its speed up to 1,850 km/h, while the Harrier flies at the speed of ~730 km/h.

In the USA, VTOL supersonic aircraft are being actively developed on a competitive basis, and two supersonic VTOL fighters are being tested: the Boeing X-32 and Lockheed X-35 equipped with a high-powered, vertically mounted, contra-rotating, shaft-driven lift fan located forward of the main engine in the centre of the aircraft. Bypass air from the cruise engine turbofan exhausts through a pair of nozzles in the wings, while the lift fan balances the vectoring cruise nozzle at the tail (Fig. 11.50).

VTOL-capability is highly desirable in transport aircraft. In the USA, Bell Helicopter partnered with Boeing to develop and manufacture the



**Fig. 11.49. Yak-141 Russian Supersonic VTOL Fighter-Bomber**



**Fig. 11.50. Power Plant of Lockheed X-35 Supersonic VTOL Fighter**

Bell-Boeing MV-22B Osprey VTOL military cargo aircraft (Fig. 11.51, ref. Table 11.4). Nacelles of this AC turboprop engines are arranged on the wing tips and can be turned  $\sim 90^\circ$  to provide vertical thrust for vertical take-off (nacelles vertical, rotors horizontal). Once airborne, the nacelles rotate forward  $90^\circ$  for horizontal flight, converting the AC to a more fuel-efficient, higher-speed turboprop airplane: the horizontal speed increases; the wing begins to create lift; and the AC is able to reach the cruising speed of 510 km/h. Maximum take-off weight: 27,440 kg; maximum payload: only 9,070 kg.

There is a passenger version of the AC for 22 passengers. It is necessary to note that the layout of mentioned AC eliminates a possibility of short take-off and landing that is intrinsically for all present-day VTOL aircraft.

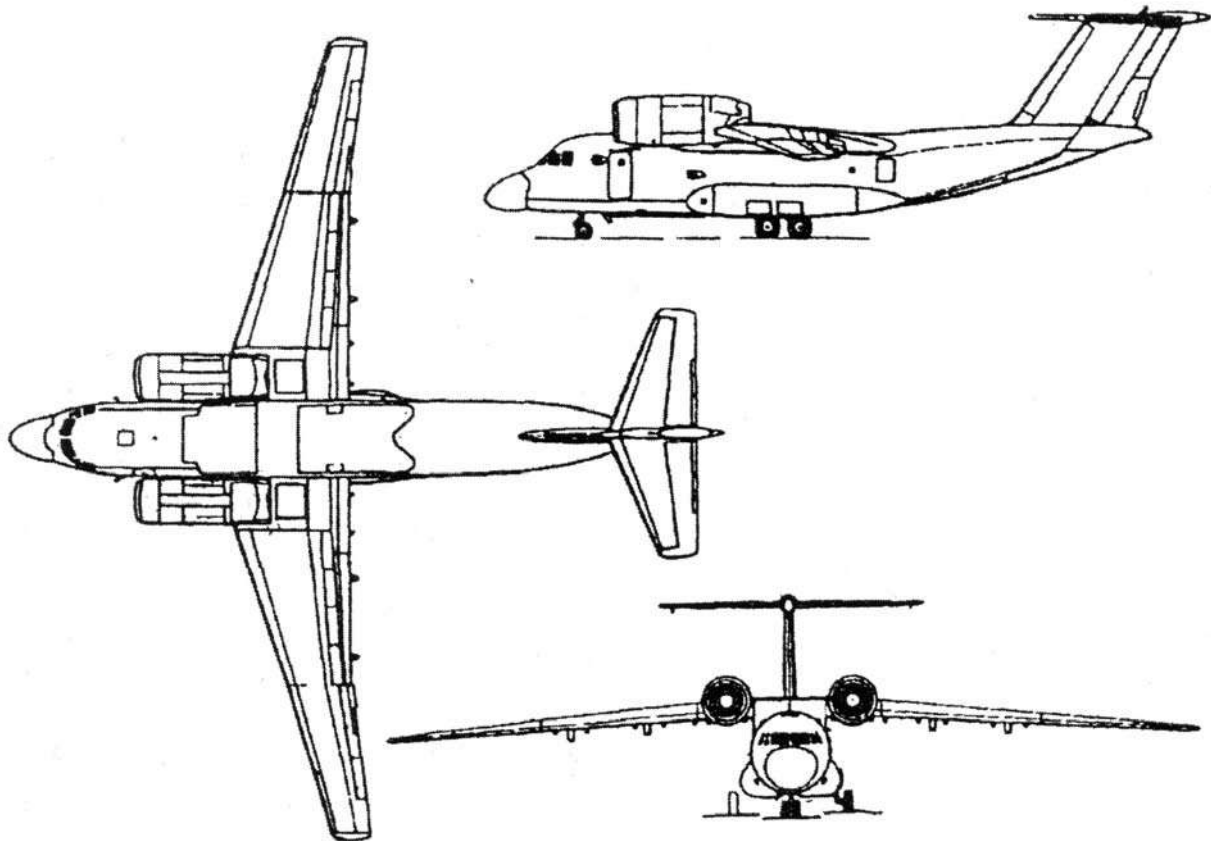


**Fig. 11.51. Bell-Boeing MV-22B Osprey Military VTOL Cargo Aircraft**

### 11.14. Power Plants for STOL Aircraft: Design Philosophy

The fact that passengers are more interested in shortening the total time of their journey, rather than raising flying speed, makes the concept of a passenger **short take-off and landing** (STOL) airplane more attractive than that of the SST. Indeed, the time it takes to get to the airport of departure and from the airport of arrival is often comparable with the flight time, especially for local and regional airlines. Nobody doubts the principal feasibility to design STOL passenger airplanes (helicopters are not competitive due to their speed limitations). However, it is a task for the aviation of tomorrow to develop the airplane that will take-off from and land on a city square or the roof of a tall building and will be safe for both its passengers and city dwellers.

The long-term successful operation of the An-72 and An-74 cargo STOL airplanes (Fig. 11.52, ref. Table 11.4), followed by the creation of the An-70 (Fig. 11.53, ref. Table 11.4), which is able to take off and land on an aircraft carrier deck, proves that Ukrainian AC designers have serious achievements in the development of the STOL passenger airplane concept. In the An-74 basic version the engines are located over the wing



**Fig. 11.52. An-74 STOL Transport Aircraft designed by Ukrainian Aviation Scientific-Technical Complex named for O.K. Antonov**



in accordance with Coanda effect. Its highly-mechanized wing has significant camber when the flaps and slats are extended. The jet stream from powerful turbofan engines, located forward, «sticks» to the wing surface and is directed downward by the wing high-lift devices. Thrust reversal is applied to shorten the landing distance, the jet stream being deflected forward and upward to prevent foreign objects from getting into air intakes. These engineering solutions, in addition to high-arranged engines, ensure the AC operation from natural-surfaced, even pebbly, runways. The implementation of a number of promising engineering solutions in designing the An-74 and its power plant has laid the foundations for the whole AC family: patrol, airborne warning and control system (AWACS), cargo-passenger, convertible cargo-passenger aircraft; administrative, business and VIP-passenger versions, etc.



**Fig. 11.53. An-70 STOL Transport Aircraft designed by Ukrainian Aviation Scientific-Technical Complex named for O.K. Antonov**

It has not been our goal to analyse all promising trends in designing current generation AC, but only those related to important innovations in power plant design. For more information, refer to «The Concepts of Modern Aerospace Engineering» course.

## **Appendix.**

### **BRIEF AND FULL NAMES**

### **OF AIRCRAFT-BUILDING AND ENGINE-BUILDING FIRMS**

**Aero Vironment** — AeroVironment Inc., AeroVironment Incorporated (рус. Аэро Виронмент). It is a technology company. The company is probably most well-known for developing a series of lightweight human-powered and then solar-powered vehicles. (Headquarters: Monrovia, California and Simi Valley, California, USA) [34].

**Aérospatiale** — Aérospatiale SNI, France (рус. Аэроспасьяль) [31]. It was a French aerospace manufacturer that built both civilian and military aircraft, rockets and satellites. It was originally known as Société Nationale d'Industrie Aérospatiale (SNIAS). The company (as SNIAS) was created in 1970 by the merger of the state-owned companies Sud Aviation, Nord Aviation and Société d'études et de réalisation d'engins balistiques (SÉREB). In 1992, DaimlerBenz Aerospace AG (DASA) and Aérospatiale combined their helicopter divisions to form the Eurocopter Group. In 1999, Aérospatiale merged with Matra Haute Technologie to form Aérospatiale-Matra. In 2000, Aérospatiale-Matra merged with Construcciones Aeronáuticas SA (CASA) of Spain and DaimlerChrysler Aerospace AG (DASA) of Germany to form the European Aeronautic Defence and Space Company (EADS). In 2001, Aérospatiale-Matra's missile group was merged with Matra BAe Dynamics and the missile division of Alenia Marconi Systems to form MBDA [34].

**Airbus** — Airbus Industrie, Airbus SAS (рус. Эрбас, Эйрбас). It is an aircraft manufacturing subsidiary of EADS, a European aerospace company. The company produces around half of the world's jet airliners. Airbus began as a consortium of aerospace manufacturers. Consolidation of European defence and aerospace companies around the turn of the century allowed the establishment of a simplified joint stock company in 2001, owned by EADS (80%) and BAE Systems (20%). After a protracted sales process BAE sold its shareholding to EADS in 2006. Airbus employs around 57000 people at sixteen sites in four European Union countries: Germany, France, the United Kingdom, and Spain. Final assembly production is at Toulouse (France), Hamburg (Germany), Seville (Spain) and, since 2009, Tianjin (China). Airbus has subsidiaries in the United States, Japan and China [34]. (Headquarters: 1 Round Point Maurice Bellonte, F-31707 Blagnac Cedex, France [33]).

**Allison** — Allison Gas Turbine Division (рус. Аллисон). It was a US aircraft engine manufacturer. In the mid-1970s the Allison Division of General Motors Corporation in Detroit designed ceramic components into the truck engine. Allison continued to work with General Motors on development of ceramic-turbine powered engines until the early 1990s. In 1995 was acquired by Rolls-Royce plc to become a subsidiary, Rolls-Royce Corporation. With the acquisition of Allison, Rolls-Royce expanded its product line to the point where it can now offer engines in virtually all market segments, from helicopters to the largest wide-body aircraft [34]. (Headquarters: P.O. Box 420, Indianapolis, Indiana 46206-0420, USA [33]).

**АНПК МиГ (АНПК МиГ)** — MiG Aviation Scientific Industrial Complex (Russian: Авиационный научно-промышленный комплекс МиГ). Headquartered: Россия, Москва, Ленинградское шоссе 6 [33].

**АНТК Antonov (АНТК АНТОНОВ)** — Aviation Scientific-Technical Complex named for O.K. Antonov (Russian: Авиационный научно технический комплекс имени О.К. Антонова). Headquartered: Украина, Киев, ул. Академика Туполева, 1 [33].

**ATLAS Elektronik** — Atlas Elektronik GmbH (GmbH in Germany — Gesellschaft mit beschränkter Haftung). It is a naval/marine electronics and systems business. The company was a subsidiary of BAE Systems until 2005 when it was to ThyssenKrupp and EADS. Prior to 2003 Atlas Elektronik was part of STN Atlas Elektronik GmbH, a joint venture between Rheinmetall and BAE Systems. In 2003 the company was split; Rheinmetall acquired the land systems business (Rheinmetall Defence Electronics) and BAE took control of the naval business, keeping the name Atlas Elektronik. BAE's involvement dates back to 1997 when British Aerospace (BAE Systems prior to Marconi merger) acquired 49% from Rheinmetall. (Headquarters: Bremen, Germany) [34].

**АВПК Сухои (Сухои, АВПК Сухой)** — State Unitary Enterprise Aviation Military-Industrial Complex Suchoi, which affiliate is Suchoi experimental Design Bureau JSC (Russian: Государственное унитарное предприятие авиационный военно-промышленный комплекс Сухой, в состав которого входит Опытно-конструкторское бюро Сухого АООТ). Headquartered: Россия, Москва, ул. Поликарпова 23А [33].

**BAC** — British Aircraft Corporation. It was a British aircraft manufacturer formed from the government-pressured merger of English Electric Aviation Ltd., Vickers-Armstrong (Aircraft), the Bristol Aeroplane Company and Hunting Aircraft in 1960. In 1977, BAC, the Hawker Siddeley Group and Scottish Aviation were nationalised and merged under the provisions of Aircraft and Shipbuilding Industries Act. This new group was established as a statutory corporation, British Aerospace (BAe) [31, 34].

**BAe** — BAe Systems plc, British Aerospace Public Ltd. It was a UK aircraft, munitions and defence-systems manufacturer. In 1999 it purchased Marconi Electronic Systems, the defence electronics and naval shipbuilding subsidiary of the General Electric Company plc, to form BAE Systems. The company was formed in the United Kingdom as a statutory corporation in 1977 as a result of the Aircraft and Shipbuilding Industries Act. This called for the nationalisation and merger of the British Aircraft Corporation, Hawker Siddeley Aviation, Hawker Siddeley Dynamics and Scottish Aviation. In 1979 BAe officially joined Airbus, the UK having previously withdrawn support for the consortium in 1969 [34]. (Headquarters: Warwick House, PO Box 87, Farnborough Aerospace Centre, Farnborough, Hampshire GU14 6YU) [33].

**Bell Boeing** —merger of Bell Helicopter Textron, USA and the Boeing Rotorcraft Systems, USA for V-22 Osprey developing and manufacturing. The Bell-Boeing V-22 Osprey is a multi-mission, military, tiltrotor aircraft with both a vertical takeoff and landing (VTOL) and short takeoff and landing (STOL) capability. It is designed to perform missions like a conventional helicopter with the long-range, high-speed cruise performance of a turboprop aircraft. The V-22 was developed and is built jointly by Bell Helicopter, which manufactures and integrates the wing, nacelles, rotors, drive system, tail surfaces, and aft ramp, as well as integrates the Rolls-Royce engines, and Boeing Helicopters, which manufactures and integrates the fuselage, cockpit, avionics, and flight controls. The joint development team is known as Bell Boeing. [34]. (Bell-Boeing V-22 Joint Program Office, PO Box 70, Patuxent River, Maryland 20670-0070) [33].

**Boeing Hel.** — Boeing Helicopters or Boeing Helicopter Company. It is previous name of Boeing Rotorcraft Systems (before that Boeing-Vertol). It is US aircraft manufacturer, part of Boeing Integrated Defense Systems. The factories are in Ridley Township, Pennsylvania, a suburb of Philadelphia, and Mesa, Arizona [34].

**Boeing** — the Boeing company (рус. Боинг). It is a US major aerospace and

defense corporation. Boeing has expanded over the years, merging with McDonnell Douglas in 1997. Now it is the largest global aircraft manufacturer by revenue, orders and deliveries, and the third largest aerospace and defense contractor in the world. Boeing is the largest exporter by value in the United States [34]. (Headquarters: Seattle, Washington\$ 100 North Riverside Plaza, Chicago, Illinois 60606) [33].

**Convair** — Consolidated Vultee Aircraft Corporation (рус. — Конвэр). It was an American aircraft manufacturing company which later expanded into rockets and spacecraft. Convair existed as a company for the design, development, and manufacturing of high-technology aerospace products, and/or sub-units of them, or else was a subsidiary of a larger corporation General Dynamics. Convair existed as a company from 1943 until 1994 [31, 34].

**DARPA** — the Defense Advanced Research Projects Agency (DARPA). It is an agency of the United States Department of Defense responsible for the development of new technology for use by the military. DARPA has been responsible for funding the development of many technologies which have had a major effect on the world, including computer networking. Its original name was simply Advanced Research Projects Agency (ARPA), but it was renamed DARPA (for Defense) in 1972, then renamed ARPA again in 1993, and then renamed DARPA again in 1996. DARPA was established during 1958 (as ARPA) in response to the Soviet launching of Sputnik during 1957, with the mission of keeping US military technology more sophisticated than that of the nation's potential enemies. DARPA has around \$3.2 billion budget. It focuses on short-term (two to four-year) projects run by small, purpose-built teams [34].

**Fairchild Republic** — Fairchild Republic company (рус. Фэрчайлд, Фэйрчайлд). It was an aircraft and aerospace manufacturing company. The company was founded in 1929 as Fairchild Aviation Corporation. In 1964, the company purchased Hiller Aircraft, changing its name to Fairchild Hiller. In 1965, the company acquired the Republic Aviation Company. Following the death of its founder, Fairchild changed its name to Fairchild Industries in 1971. After the company's takeover of Dornier's civil assets in 1996, the company was renamed Fairchild Dornier. In 1999 Fairchild Aerospace Corporation was acquired by German insurer Allianz A.G. and the United States investment group Clayton, Dubilier & Rice Inc. for \$1.2 billion. In 2002-2003, the company was taken over by M7 Aerospace. It based at various times in Farmingdale, New York; Hagerstown, Maryland; and San Antonio, Texas [34].

**General Dynamics, GD** — General Dynamics Corporation (рус. Дженерал Дайнемикс). It is a US defense conglomerate formed by mergers and divestitures, and as of 2008 it is the fifth largest defense contractor in the world. The company has changed markedly in the post-Cold War era of defense consolidation. The company has four main business segments: Marine Systems, Combat Systems, Information Systems and Technology, and Aerospace. The company's former Fort Worth Division manufactured the F-16 Fighting Falcon, the most-produced Western jet fighter, but that subsidiary was sold to Lockheed in 1993. GD reentered the airframe business in 1999 with their purchase of Gulfstream Aerospace [31, 34].

**General Electric, GE** — General Electric Company (рус. Дженерал Электрик). It is a multinational American technology and services conglomerate incorporated in the State of New York. In 2009, Forbes ranked GE as the world's largest company. The company has 323,000 employees around the world. (Headquarters: Fairfield, Connecticut, USA) [31, 34].

**Grumman** — the Grumman Aircraft Engineering Corporation, later Grumman Aerospace Corporation (рус. Грумман) . It was a leading 20th century US producer of

military and civilian aircraft. Its independent existence ended in 1994 when it was acquired by Northrop Corporation to form Northrop Grumman [31, 34].

**Gulfstream** — Gulfstream Aerospace Corporation, USA (рус. Гольфстрим). It is a producer of several models of jet aircraft. Gulfstream has been a unit of General Dynamics since 1999. (Headquarters: 500 Gulfstream Road, Savannah, Georgia 31408) [33].

**Hawker Siddeley** — Hawker Siddeley Group (рус. Хокер Сидли, Хоукер Сиддли). It was a group of British manufacturing companies engaged in aircraft production. Hawker Siddeley combined the legacies of several British aircraft manufacturers, emerging through a series of mergers and acquisitions as one of only two such major British companies in the 1960s. In 1977, Hawker Siddeley became a founding component of the nationalised British Aerospace (BAe). The non-aviation and foreign interests were retained by a holding company known as 'Hawker Siddeley Group Plc after 1980 [31, 34].

**Honeywell, HON** — Honeywell International Inc. (рус. Хоневел). It is a major US conglomerate company that produces a variety of consumer products, engineering services, and aerospace systems for a wide variety of customers, from private consumers to major corporations and governments. (Headquarters: Morristown, New Jersey, USA)

**Hughes Helicopter** — Hughes Helicopters Incorporation (рус. Хьюз Геликоптерс). It was one of the major US manufacturers of military and civil helicopters from the 1950s to the 1980s. The company began in 1947, as a unit of Hughes Aircraft, then was part of the Hughes Tool Company after 1955. It became the Hughes Helicopter Division, Summa Corporation in 1972, and was reformed as Hughes Helicopters, Inc. in 1981. However, throughout its history, the company was informally known as "Hughes Helicopters". It was sold to McDonnell Douglas in 1984 and was named McDonnell Douglas Helicopter Company in 1985 [34].

**Ilyushin AK (AK Ильюшина)** — Aviation Complex named for S.V. ILYUSHIN (Russian: Авиационный комплекс имени С.В. Ильюшина). Headquartered: Россия, Москва, Ленинградский пр. 45г [33].

**Klimov corp. (ГУНПП Климова)** — the Klimov Company, the Klimov Corporation (Russian: Ленинградское научно-производственное объединение имени В.Я. Климова, later Государственное унитарное научно-производственное предприятие (ГУНПП) «Завод имени В.Я. Климова» [31]). Headquartered: Россия, г. Санкт-Петербург, ул. Кантемировская, 11 [32 ]

**LHTEC** — Light Helicopter Turbine Engine Company. It is a joint venture between Rolls-Royce and Honeywell. The company was originally a partnership between the Allison Engine Company and AlliedSignal Aerospace. In 1995 Rolls-Royce acquired Allison, and AlliedSignal merged with Honeywell in 1999, and adopted its name. The partnership was formed to develop the T800 turboshaft engine for the United States Army's RAH-66 Comanche attack-reconnaissance helicopter. Despite the cancellation of this project, the company has been able to sell the T800, and its civil CTS800 model, for other helicopters [34].

**Lockheed Martin, LMT** — Lockheed Martin Corporation (рус. Локхид Мартин). It is US aerospace, defence, security, and advanced technology company with worldwide interests. It was formed by the merger of Lockheed with Martin Marietta in 1995. Lockheed Martin is the world's largest defense contractor by revenue as \$42,731 of 2008. In 2005, 95% of Lockheed Martin's revenues came from the United States Department of Defense, other US federal government agencies, and foreign military

customers [34]. (Headquartered: 6801 Rockledge Drive, Bethesda, Maryland 20817) [33].

**Lockheed** — Lockheed Corporation, originally Loughhead Aircraft Manufacturing Company (рус. Локхид). It was an American aerospace company founded in 1912 which merged with Martin Marietta in 1995 to form Lockheed Martin Corporation [31, 34].

**McDonnell-Douglas** — McDonnell Douglas Corporation (рус. МакДоннел-Дуглас). It was a major American aerospace manufacturer and defense contractor, producing a number of famous commercial and military aircraft. It formed from a merger of McDonnell Aircraft and Douglas Aircraft in 1967. McDonnell Douglas later merged with its rival Boeing in 1997 in a \$13 billion stock-swap to create the Boeing Company (ref.).

**Mil MVZ (МВЗ Миля)** — Moscow Helicopter Plant named for M.L. Mil JSC (Russian: Московский вертолетный завод имени М.Л. Миля, ОАО). Headquartered: Россия, Москва, ул. Сокольнический вал, 2 [33].

**МКВ Raduga (МКБ «Радуга»)** — МКВ Raduga (Russian: МКБ Радуга, meaning Raduga Design Bureau, where «raduga» literally means «rainbow»). It was formerly a division of the Mikoyan-Gurevich design bureau, before being spun off as a separate ОКБ (design bureau) in 1957. Headquartered: Россия, Московская обл., г. Дубна [34].

**НИИ PFM «KhAI» (НИИ ПФМ «ХАИ»)** — Interbranch Research Institute of Physical Simulation Problems of Aircraft Critical Flight Modes of National Aerospace University named for N.E. Zhukovsky «Kharkov Aviation institute» (Russian: Межотраслевой научно-исследовательский институт проблем физического моделирования режимов полета самолетов Национального аэрокосмического университета имени Н.Е. Жуковского «Харьковский авиационный институт» (Headquartered: Украина, Харьков, ул. Чкалова 17).

**North American** — North American Aviation (рус. Норт Амэрикан, Норт Амэрикан Эвиэйшн). It was a major US aerospace manufacturer, responsible for a number of historic aircraft, including XB-70 strategic bomber, the X-15 rocket plane, Apollo Command and Service Module, the second stage of the Saturn V rocket, the Space Shuttle orbiter and the B-1 Lancer. Through a series of mergers and sales, North American Aviation is now part of Boeing. In December 1996, the defense and space divisions of Rockwell International (including the North American Aviation divisions Autonetics and Rocketdyne) were sold to Boeing, which integrated the product lines into their Integrated Defense Systems division. Rocketdyne was eventually sold by Boeing to UTC Pratt & Whitney, in 2005 [34].

**Northrop Grumman** — Northrop Grumman Corporation (рус. Нортроп-Грумман). It is an American aerospace and defence technology company formed in 1994 purchase of Grumman by Northrop. The company was the fourth largest defence contractor in the world in 2008, and the largest builder of naval vessels. (Headquartered (Headquartered: 1840 Century Perk East, Los Angeles, California 90067-2199, but in 2010 it announced plans to move to the Washington D.C. region by 2011) [33, 34].

**Northrop** — the Northrop Corporation (рус. Нортроп). It was a leading United States aircraft manufacturer from its formation in 1939 until 1994. The company is known for its development of the flying wing aircraft, although only a few of these have entered service. In 1994, partly due to the loss of the Advanced Tactical Fighter contract to Lockheed Martin and the removal of their proposal from consideration for the

Joint Strike Fighter competition, the company bought Grumman to form Northrop Grumman [33, 34].

**PMKB (ПМКБ)** — Soloviev Design Bureau, Aviadvigatel & Perm Engine Company (Russian: ОКВ-19, later Пермское моторостроительное конструкторское бюро led by Pavel Aleksandrovich Soloviev. Following the collapse of the Soviet Union, the bureau became part of Компании Авиадвигатель). Headquartered: Россия, г. Пермь [31, 34].

**Pratt & Whitney (P&W, Пратт энд Уитни)** — United Technologies Pratt and Whitney Company (рус. Пратт энд Уитни). It is an American aircraft engine manufacturer of products widely used in both civil and military aircraft. As one of the «big three» aero-engine manufacturers, it competes with General Electric and Rolls-Royce, although it has also formed joint ventures with both of these companies. In addition to aircraft engines, Pratt & Whitney manufactures gas turbines for industrial and power generation, marine turbines, and rocket engines. The company support more than 9000 customers in 180 countries around the world. (Headquartered: Flhtc 400 Main St., East Hartford, CT 06108, USA) [32, 34].

**Progress, ZMKB Progress («Прогресс», ЗМКБ «Прогресс»)** — Ivchenko-Progress ZMKB, Zaporizhia Machine-Building Design Bureau Progress State Enterprise named after Academician O.H. Ivchenko (Russian: ОКБ-478, later Государственное предприятие «Запорожское машиностроительное конструкторское бюро «Прогресс» имени академика А.Г. Ивченко»), (Ukrainian: Запорізьке машинобудівне конструкторське бюро «Прогрес» ім. О.Г. Івченка). It is a Ukrainian design house for aircraft engines whose products are widely used in both civil and military aircraft, most notably by Antonov, Ilyushin, Tupolev, Mil and Yakovlev, PZL-Mielec. Both the largest plane in the world, the An-225 and the largest helicopter, the Mi-26 are powered by Lotarev engines. Headquartered: Украина, г. Запорожье, ул. Иванова, 2 [32, 33].

**Rockwell** — Rockwell International Corporation (рус. Рокуэлл, Роквелл). It was a major American manufacturing conglomerate in the latter half of the 20th century. It was the ultimate incarnation of a series of companies under the sphere of influence of Willard Rockwell. At the end of the 1980s, the company sold its valve and meter division (formerly Rockwell Mfg.) to British Tire and Rubber. It also sold its printing press division to an internal management team. Following the «peace dividend» and the fall of the Soviet bloc, the company sold its defense and aerospace business (including what was once North American Aviation and Rocketdyne) to Boeing Integrated Defense Systems in 1996. In 2001, what remained of Rockwell International was split into two companies: Rockwell Automation and Rockwell Collins, both publicly traded companies [31, 34].

**Rolls-Royce, RR** — Rolls-Royce plc. (рус. Роллс-Ройс). It is a British aircraft engine maker, and the largest in the world. The company has related businesses in the defence aerospace, marine and energy markets. Rolls-Royce was nationalised in 1971, by which time aircraft engines had long been the most significant part of the business (the automobile company was separated in 1973 as Rolls-Royce Motors). Rolls-Royce The present Rolls-Royce plc was re-privatised in 1987. (Headquartered: Derby, England, United Kingdom) [34].

**Rybinsk, Rybinsk motors (ОАО РМ)** — Rybinsk Motors, public corporation (Russian: Открытое акционерное общество «Рыбинские моторы»). Rybinsk Motors was created by the merger of Rybinsk Motor-Building Design Bureau RKBM and Mo-

tor-Building Plant in 1970. Headquartered: Россия, г. Рыбинск Ярославской обл., пр-т Ленина, 163 [32, 33]

**Saturn, Lyulka-Saturn, (САТУРН, Люлька-САТУРН)** — NPO Saturn (Russian: Научно-производственное объединение «САТУРН» [31], later Открытое акционерное общество (ОАО) «А. Люлька-САТУРН»). Headquartered: Россия, Москва, ул. Касаткина, 13 [32, 33]

**Sikorsky** — Sikorsky Aircraft Corporation (рус. Сикорский). It is an American aircraft (mainly helicopter) manufacturer. In 1929 the company became a part of United Aircraft and Transport Corporation (now United Technologies Corporation) [34]. (Headquartered: 6900 Main Street, Stratford, Connecticut 06615-9129) [33].

**SNECMA (рус. СНЕКМА)** — an acronym for Société Nationale d'Étude et de Construction de Moteurs d'Aviation (in English, «National Company for the Design and Construction of Aviation Engines»). It is a major French manufacturer of engines for commercial and military aircraft, and for space vehicles. In 2005, the Snecma group, which included Snecma (called Snecma Moteurs at this time), merged with SAGEM to form SAFRAN. Snecma is now a subsidiary of the SAFRAN Group and previous Snecma group subsidiaries have been reorganised within the wider group [34].

**Soyuz, Soyuz AMNTK («Союз», АМНТК «Союз»)** — Aviamotors Scientific-Technical Complex «Soyuz», public corporation (Russian: Открытое акционерное общество Авиамоторный научно-технический комплекс «Союз»). Headquartered: Россия, г. Москва, Лужнецкая наб., 2/4 [32, 33].

**Teledyne Ryan** — Teledyne Ryan Aeronautical (USA). The Ryan Aeronautical Company was founded by T. Claude Ryan in 1934. It produced many airplanes over the years, most of them trainers. T. Claude Ryan sold Ryan Aeronautical Corp. to the Teledyne Corporation in 1969. It produced a series of pilotless drones and airframes for the Bell Huey helicopter. Northrop Grumman purchased Teledyne Ryan in 1999 [34].

**Tupolev ANTK (АНТК Туполев)** — Aviation Scientific-Technical Complex named for A.N. Tupolev, JSC (Russian: Авиационный научно-технический комплекс имени А.Н. Туполева, открытое акционерное общество). Headquartered: Россия, Москва, наб. Академика Туполева, 17 [33].

**Williams Research** — Williams Research Corporation, later Williams International Corporation. It has been building small turbofan engines since the 1950s for use in cruise missiles, target, reconnaissance drones and small jet-powered aircraft. Using the missile engines, Williams developed a series of personal VTOL flying craft, including a jet-powered belt in 1969, the Williams Aerial Systems Platform (WASP), also known as the «flying pulpit» in the 1970s, and the X-Jet, which was evaluated by the United States Army in the 1980s. (Headquartered: Walled Lake, Michigan, United States [34].

**Yakovlev ОКБ (ОКБ Яковлева)** — Experimental Design Bureau named for A.S. Yakovlev JSC (Russian: Опытное-конструкторское бюро имени А.С. Яковлева). Headquartered: Россия, Москва, Ленинградский пр., 68 [33].



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Цуканов Руслан Юрійович  
Риженко Олександр Іванович  
Пехтерев Валентин Дмитрович

AIRPLANE POWER PLANTS SYSTEMS DESIGNING

ПРОЕКТУВАННЯ СИСТЕМ СИЛОВИХ УСТАНОВОК ЛІТАКІВ

Переклад: Р. Ю. Цуканов, О. І. Риженко

Редактори:

Т. А. Ястремська, М. В. Кириленко, Н. С. Дорошенко

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Національний аерокосмічний університет ім. М. Є. Жуковського

«Харківський авіаційний інститут»

61070, Харків-70, вул. Чкалова, 17

<http://www.khai.edu>

Видавничий центр «ХАІ»

61070, Харків-70, вул. Чкалова, 17

[izdat@khau.edu](mailto:izdat@khau.edu)