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AN INTRODUCTION TO ROCKETRY

BY

V. I. Feodos'yev and G. B. Sinyarev

FOREIGN TECHNOLOGY DIVISION

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By: V. I. Feodos'yev and G. B. Sinyarev

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Edited by JPRS

V. I. Feodos'yev i G. B. Sinyarev

VVEDENIYE
V RAKETNUYU TEKHNIKU

IZDANIYE VTOROYE
ISPRAVLENNOYE I DOPOLNENNOYE

Dopushcheno
Ministerstvom vysshego obrazovaniya SSSR
v kachestve uchebnogo posobiya
dlya vysshikh tekhnicheskikh
uchebnykh zavedeniy

Gosudarstvennoye

Nauchno-Tekhnicheskoye Izdatel'stvo

Oborongiz

Moskva 1960

Pages 1-506

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U. S. BOARD ON GEOGRAPHIC NAMES TRANSLITERATION SYSTEM

Block	Italic	Transliteration	Block	Italic	Transliteration
А а	*А а*	A, a	Р р	*Р р*	R, r
Б б	*Б б*	B, b	С с	*С с*	S, s
В в	*В в*	V, v	Т т	*Т т*	T, t
Г г	*Г г*	G, g	У у	*У у*	U, u
Д д	*Д д*	D, d	Ф ф	*Ф ф*	F, f
Е е	*Е е*	Ye, ye; E, e*	Х х	*Х х*	Kh, kh
Ж ж	*Ж ж*	Zh, zh	Ц ц	*Ц ц*	Ts, ts
З з	*З з*	Z, z	Ч ч	*Ч ч*	Ch, ch
И и	*И и*	I, i	Ш ш	*Ш ш*	Sh, sh
Й й	*Й й*	Y, y	Щ щ	*Щ щ*	Shch, shch
К к	*К к*	K, k	Ъ ъ	*Ъ ъ*	"
Л л	*Л л*	L, l	Ы ы	*Ы ы*	Y, y
М м	*М м*	M, m	Ь ь	*Ь ь*	'
Н н	*Н н*	N, n	Э э	*Э э*	E, e
О о	*О о*	O, o	Ю ю	*Ю ю*	Yu, yu
П п	*П п*	P, p	Я я	*Я я*	Ya, ya

* ye initially, after vowels, and after ъ, ь; e elsewhere. When written as ё in Russian, transliterate as yё or ё. The use of diacritical marks is preferred, but such marks may be omitted when expediency dictates.

FOLLOWING ARE THE CORRESPONDING RUSSIAN AND ENGLISH

DESIGNATIONS OF THE TRIGONOMETRIC FUNCTIONS

Russian	English
sin	sin
cos	cos
tg	tan
ctg	cot
sec	sec
cosec	csc
sh	sinh
ch	cosh
th	tanh
cth	coth
sch	sech
csch	csch
arc sin	$\sin^{-1}$
arc cos	$\cos^{-1}$
arc tg	$\tan^{-1}$
arc ctg	$\cot^{-1}$
arc sec	$\sec^{-1}$
arc cosec	$\csc^{-1}$
arc sh	$\sinh^{-1}$
arc ch	$\cosh^{-1}$
arc th	$\tanh^{-1}$
arc cth	$\coth^{-1}$
arc sch	sech^{-1}
arc csch	csch^{-1}

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rot	curl
lg	log

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FOREWORD

The recent achievements in rocketry have evoked the interest of a vast number of technical groups in the questions of rocket flight; this has caused a pressing need for technical literature devoted to this question.

The authors have taken it upon themselves to introduce the reader to a course dealing with rocketry in general, not requiring serious preliminary preparation, particularly in special courses of aerogasdynamics and thermodynamics. An understanding of the material here presented requires only a knowledge of general physics and chemistry and the beginnings of higher mathematics.

Since rocketry at all times has developed mainly with the aim of military application, questions in rocketry and the theory of rocket flight must in some way or other be associated with the applicable military problems. In the present work, such a relationship has been stressed only when necessary, mainly when describing the design and purpose of rockets. The present work does not pretend to discuss the military application of rockets or general problems of rocket artillery. A discussion of many special questions of rocketry, those of an independent nature such as radio control, homing, telemetry, and certain others, has either been omitted or given in greatly abbreviated form. However, the book gives sufficient information so that the reader can get a clear picture of rocketry as a whole.

When the book was prepared for the second edition, errors in the first edition were corrected, and additional information and changes brought about by achievements in rocketry in recent years were included.

Chapters III, IV, V, and most of Chapter VI were written by G. B. Sinyarev. The remaining chapters and sections were written by V. I. Feodos'yev, who also served as general editor for the book.

In their work the authors continually had the friendly support and help of their colleagues, to whom at this time they would like to express their heartfelt thanks.

The Authors

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INTRODUCTION

When we speak of reaction propulsion we have in mind movement due to a repulsive force, i.e., the reaction of a stream of particles ejected from an apparatus.

It is not always possible to draw a strict line between reaction and nonreaction propulsion. We must not forget that any means of movement in the broad sense of the word is based on the reaction principle, i.e., the ejection of some mass in the opposite direction. Boats and ships move due to the reaction of a mass of water thrust in the opposite direction. The propeller of an airplane creates thrust, repelling an air mass. A high-jumper simultaneously repels the earth, although at an incomparably slower speed than he himself moves upward.

As we see, in all cases, the imparting of speed to any mass involves the imparting of an opposite speed to another mass, and in this sense any mechanical movement can be considered to be based on the reaction principle. However, a certain restriction should be imposed on the concept of reaction propulsion; it is characterized by the repulsion of relatively small masses at relatively high speeds, creating direct reaction.

Boats, ships, and planes move as a result of <u>indirect reaction</u>. Here, between the engine, which is the power source, and the repulsed water or air mass, there is some intermediate mechanical element — the propeller. For a boat, the oarsman is the "motor," and the oars — the propeller. For a ship, the screw propeller is the propeller, while for a plane the airscrew is the propeller. A high-jumper is propelled using his legs, which in this case perform the function of a propeller.

In these examples the propeller is a characteristic sign of indirect reaction. When there is no propeller, the reaction created by the engine is called <u>direct</u>

reaction. For example, this is the case with a gunpowder rocket. Here the engine is the chamber in which gas forms as the powder burns. The reaction is created directly by jets of exhaust gas. There is no intermediate mechanism between the engine and the repulsed mass of gases.

When speaking of jet engines we must distinguish between turbojet engines, operating in conjunction with the surrounding medium, and rocket engines, which can generally operate regardless of the surrounding medium.

Airplanes equipped with turbojet engines are called jet aircraft; when these are controlled by autopilots and used as missiles, they are called self-propelled missiles. Turbojets are supplied with fuel from onboard the airplane; oxygen from the air is used as the oxidizer for this fuel. Accordingly, the operation of a turbojet depends on the surrounding medium, and a vehicle using such an engine can move in airless space only by inertia.

Apparatuses equipped with rocket engines are called rocket vehicles, or simply rockets. The gases which escape from the rocket engine are formed entirely due to the substances (usually fuel and an oxidizer) contained aboard the rocket. It is this very fact which makes the operation of a rocket engine independent of the surrounding atmosphere. A rocket can have powered flight even in airless (interplanetary) space.

Turbojet engines are sometimes called simply jet engines as opposed to rocket engines, although the concept "rocket" is included in the more general concept "jet."

Let us now examine the reasons why contemporary technology has turned to the reaction principle, and why jet technology has developed so rapidly at this particular time.

The reaction principle is best suited to high speeds, while aviation, at the present time, is just attaining those speeds for which jet engines are more efficient than prop engines. With fast flight speeds (or, in general, with fast movement), the resistance of the medium (air, water) sharply increases. Therefore, to maintain increased speed it is necessary that the engines be more powerful, without changing the weight of geometry of the plane or ship.

An increase in power of the ordinary airplane reciprocating engine involves complex design with, and this is most important, a proportional increase in weight. At the same time, an increase in the weight of the engine and, consequently, the weight of the plane, requires increased thrust, i.e., increased power, to maintain the same speed. Thus it appears that it is impossible to attain high speeds without decreasing the specific weight, i.e., the weight of the engine per unit of power, and the reciprocating engine does not produce the required effect in this case. Therefore, at high speeds it is more advantageous to use, first, an engine in which the airscrew is turned by a gas turbine, i.e., the so-called turboprop engine. This engine permits a sharp decrease in specific weight compared with the reciprocating engine (approximately two-fold).

The next step toward attaining high speeds was the elimination of the propeller, the main reason for an increase in the weight of the engine, i.e., the conversion to direct reaction and, consequently, to turbojet engines.

At slow speeds this elimination of the propeller is not justified, since at these speeds all properly designed propellers are sufficiently effective. Therefore, it is hardly likely that the reaction principle will be used, for example, in automobiles or motorcycles. Its future lies in high-speed aviation. To attain even higher speeds, the limits of the atmosphere must be exceeded. A long flight in the atmosphere at speeds of about 2 km/sec and higher is extremely difficult due to the intense heating of the craft. A plane with a turbojet engine cannot penetrate the highly rarefied layers of the atmosphere, where it would not encounter great drag. At very high altitudes there is not enough air for the engine to perform normally.

Thus it is useless in this case to even consider turbojet, much less reciprocating, engines which cannot operate without air. The only engine suitable for such a flight is the rocket engine.

This engine has yet another feature; it will enable man to conquer outer space. Even at present, man has begun his conquest of outer space using artificial earth satellites and space rockets. The time is not too far away when man, using such rockets, will overcome the Earth's gravity and embark on distant and alluring flights through the universe. This problem has not as yet been solved, but contemporary science and technology are fast approaching the solution.

* * *

Man has not created these powerful modern rockets overnight. Many centuries of persistent work and searching in the most diverse branches of technology and natural science have been necessary to accumulate the knowledge necessary to make the first noteworthy steps in the field of rocketry.

The principle of motion using direct reaction has been known for many years.

The earliest known rocket engines were those of gunpowder rockets used about 5000 years ago in China, first for entertainment, and then for military purposes. N. G. Chernyshev has stated (quite correctly, to our way of thinking) that the invention of gunpowder rockets and their use in every country is a direct result of the appearance of gunpowder, that "the idea of rockets, unavoidably and independently, was born simultaneously where the art of gunpowder preparation was prevalent, anticipating the development of ideas on firearms."*

Actually, the person who prepared the gunpowder and automatically studied its properties of necessity encountered the phenomenon of the combustion of the powder in a half-closed vessel. He could not help but notice the unusual propulsion of the vessel in a direction opposite that of the gas exhaust, and the tongue of flame and smoke that accompanied the combustion. It is natural to assume that the unusual nature of this phenomenon would serve as an impetus for man's creative activity, leading to the invention of the rocket. In any event, although we do not know the name of the inventor of the rocket, or that of the inventor of gunpowder, we are quite correct in assuming that the invention of rockets in all countries was accompanied by the invention of gunpowder, or its introduction into these countries.

The Chinese are responsible not only for the invention of rockets, but also for the first attempts to send a man aloft using powder rockets.

There are many literary sources which attest to the fact that the use of gunpowder and rockets was quite well known in Europe in the 10th-13th centuries. We have pertinent reliable data on the military application of rockets, or "arrows of flying fire" as they were then called. In later centuries the development of smoothbore artillery and its vast successes pushed rocketry into the background.

At the end of the 18th century in Europe, interest was again evoked in the military application of powder rockets. The impetus for this was the predatory wars which the British were waging in India; during these wars the Indians used military powder rockets. The simplicity of these rockets and the concentration of fire, and the psychological effect on the infantry — all these quanities of an already-forgotten weapon caused the British to carefully size up rockets. As a result, within several years the military rocket had become an integral part of the equipment of the British army, and then the armies of other European countries.

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*N. G. Chernyshev. "The Role of Russian Scientific-Technical Concepts in the Development of the Concept of Jet Flight," Izdatel'stvo MVTU im. Baumana, 1949.

The person responsible for the development of rocketry in the British army at that time was Lt. Col. William Congreve; he improved the powder rocket and increased its range to 2.5 km.

The advantages of rocket weapons included their rapid rate of fire and the possibility of massing fire. The lightness of the launching stands, and the fact that heavy barrels and recoil mechanisms were not needed, made rocket artillery maneuverable and transportable, which was particularly suited to conditions of mountain warfare and for overcoming water barriers.

The development of rockets in Europe continued until the 1880's, when rifled breech-loading artillery pieces appeared. The rockets of that era could not compete with these artillery pieces in rate of firing, range, or close-grouping of shots, and rockets disappeared from the equipment of all armies until the start of World War II.

In Russia (in what literature is available) ancient Russian rockets were first described by gun-master Onisim Mikhaylov in his work: "Regulations on War, Gun and Other Matters Concerning Military Science" (1607-1621). Describing in detail the Russian rockets, he indicated their use not only for entertainment, but for military weapons.

Much attention was devoted to questions of the military application of powder rockets during the time of Peter I. As early as 1680 a "Rocket Institute" was established by special decree in Moscow; Peter himself later participated in this institute. During the time of Peter I, a one-pound signal rocket was developed and used, ascending to a height of 1 km. This signal rocket, "model 1717," remained in used to the end of the 19th century.

Despite the vast operations using rockets, they were not widely used militarily, either during the time of Peter I or later, until the 1820's. This was due mainly to the outstanding achievements of Russian artillerymen in the preparation and military application of artillery pieces.

The Russian War of 1812 placed new requirements on the artillery: increased range, increased mobility of the artillery pieces, massed artillery fire. The greatest minds among Russian military technologists and artillerymen struggled with this problem. Further developing Russian artillery rocketry, they independently created successful designs for military rockets (demolition and incendiary) and light launching platforms. Here we should first mention the work of General Aleksandr

A. D. Zasyadko

Dmitriyevich Zasyadko (1779-1837) who, in the 1820's, contributed much to the development of Russian rocketry.

He created 2-, 2.9-, and 4-inch rockets with a range of 2.7 km. He also designed an improved launching platform (instead of the previously used "carriages" he designed a wooden tripod with an iron pipe which rotated horizontally and vertically on the tripod).

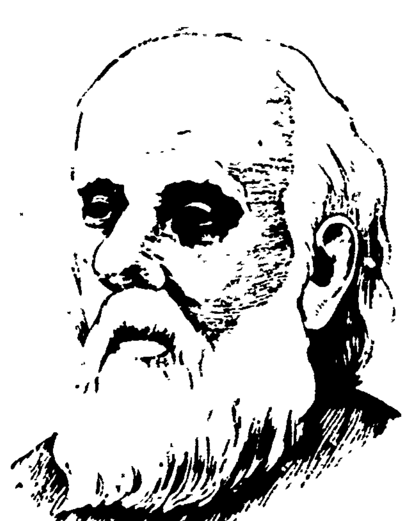
We should point out that the best British rockets of that time had a range of about 2.5 km and were launched from a heavy massive cannon. France began to use light platforms, similar to the Russian ones, only in 1853.

The successful tests of Zasyadko's rocket met with the approval of military specialists, and were incorporated into the Russian army equipment.

K. I. Konstantinov

The eminent scientist and artilleryman, General Konstantin Ivanovich Konstantinov (1819-1871) became active in the 1840's; he contributed much to the creation of Russian rocket artillery.

Konstantinov improved the technology of the production of powder rockets and created rockets having good shot grouping and range (for that era). Attributing great significance to experiments and measurement accuracy when developing new designs, Konstantinov designed a rocket electroballistic pendulum which could be used to measure the value of the reaction force at various moments during the burning of the powder mixture. This was the basis for rocket ballistics.



K. E. Tsiolkovsky

Konstantinov did much to improve rocket design, and introduced mechanization into the production process.

Despite the great successes of military rockets, Russian and non-Russian rockets lost their significance in the second half of the 19th century because of the achievements of barrel artillery, and disappeared from army equipment in the 1880's. However, the idea of rocket flight still persisted and appeared in the subsequent writings of various inventors.

The idea of rocket flight has its deepest scientific roots in the classic works of the talented Russian scientist Konstantin Eduardovich Tsiolkovsky (1857-1935).

The reaction principle began to interest Tsiolkovsky in 1883, and in 1903 the journal "Science Review" ("Nauchnoye Obozreniye") published his scientific work: "The Investigation of Space Using Rocket Devices," where he first pointed to the rocket as a means for interplanetary flight, mentioned methods for mastering outer space, and gave the basic laws of rocket movement.

This same article first contained the idea of liquid-propellant rockets, not realized until 30 years later. Tsiolkovsky worked out a possible structural design for a liquid-propellant rocket and anticipated, in general outline, the construction of present-day rockets. He wrote of a rocket using liquid oxygen and hydrogen as the propellant, the use of propellant components for cooling the rocket engine, the need for using pumps to force-feed fuel to the engine, the possibility of controlling rockets with rudders placed in the exhaust stream, and, finally, he foresaw the automatic guidance of rockets using instruments. In later works he indicated a new propellant, using atomic energy.

Tsiolkovsky examined the conditions of rocket flight in interplanetary space, the conditions for the escape of a rocket from the Earth. He indicated the expediency of constructing intermediate stations for making flights to other planets, advancing the idea of an earth satellite.

In 1929 Tsiolkovsky had the idea of constructing multi-stage rockets or, as he called them, "rocket trains."

In Tsarist Russian the prominent works of Tsiolkovsky went unnoticed, and were not immediately recognized by his contemporaries, since they were published in very limited editions.

Tsiolkovsky's ideas began to be recognized after the well-known writer Ya. I. Perel'man published his book "Interplanetary Communication" in 1915, where he presented, in popular form, the basic ideas of Tsiolkovsky. This book was very successful and was widely read, both at home and abroad.

In the 1920's Tsiolkovsky's ideas received universal recognition. Famous foreign rocket specialists wrote of this. For example, in 1929 the famous German scientist Hermann Oberth wrote in a letter to Tsiolkovsky: "You have lighted the way, and we shall work until the greatest dream of mankind is fulfilled... Naturally, I would be the last to dispute your superiority and your services in the field of rocketry."

The true superiority of Tsiolkovsky was stressed in a number of letters and addresses to the scientist on his 70th birthday. The German "Gesellschaft für Luftraumfahrt" (Society for Space Travel) wrote the following: "The Society, from the day of its inception, has always considered you, esteemed Doctor Tsiolkovsky, as one of its spiritual leaders, and has never missed the opportunity of affirming your great services and your indisputable Russian priority in the scientific study of our great idea."

With the advent of the Soviets, the works of Tsiolkovsky received their proper attention. The Soviet government gave all possible help to the talented scientist. In the 40 years before the Revolution, Tsiolkovsky wrote 130 works; in the 17 years after the Revolution he wrote 450 treatises.

A whole generation of students and enthusiastic successors was nurtured on the works of Tsiolkovsky; they carried out the ideas of their teacher. Of the direct successors of Tsiolkovsky, the talented engineer and designer Fridrikh Arturovich Tsander occupies first place. Tsander (1887-1933) became interested in rocketry even before the Revolution, but he could not begin working in earnest until after the Revolution. In 1920 Tsander began to propagate ideas of interplanetary communication and proposed several variants of possible spaceship design.

In his works, Tsander devoted much attention to the design of winged rockets. To Tsander belongs the idea of using the metal parts of aircraft as fuel. In 192[illegible] he conducted experiments on the preparation and combustion of metal alloys.

From 1928 to 1931 Tsander worked on his basic scientific treatise: "The Problem of Flight Using Reaction Apparatuses." This work was published in 1932. Its value lies not only in the ideas expressed by Tsander, but also in the engineering approach to the problems of creating rocket engines. In addition, Tsander gave specific, though simplified, technical computations for the basic processes occurring in rocket engines.

In 1929-1930 Tsander created the first model of a rocket engine, the OR-1 (Fig. 0.1). The basic elements of this model have been incorporated in today's liquid-fuel rocket engine. The model operated on gasoline and gaseous air, pressure-fed to the combustion chamber. The chamber is cooled by air which then enters the combustion chamber itself. The design provided for replaceable nozzles.

Tsander considered his OR-1 as an experimental apparatus, designed to study the basic processes in liquid-fuel rocket engines. Many firings were made with the model.

On the basis of these experiments and theoretical exploitations, a general program on creating rocket engines was proposed; the first step which Tsander made in this direction was the development of the OR-2 (Fig. 0.2), with a thrust of 50 kg, operating with gasoline and liquid oxygen. This engine was tested in 1933, after Tsander had died.

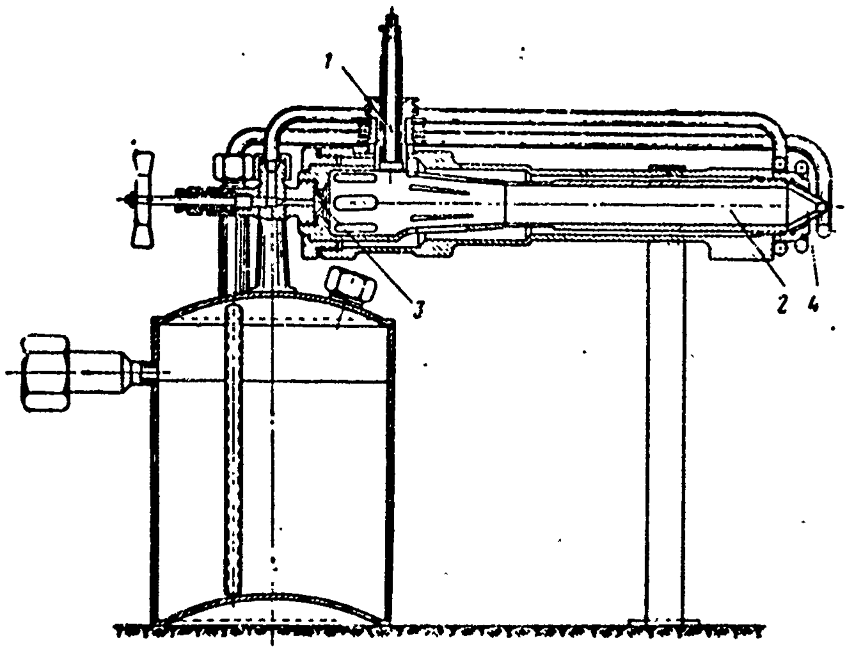


Fig. 0.1. Diagram of the OR-1 engine. 1) spark plug; 2) combustion chamber; 3) fuel injector; 4) nozzle.

Any list of Soviet leaders in the field of rocketry would not be complete without mention of the works of Yu. V. Kondratyuk, the talented self-taught mechanic who in the period 1917-1925 advanced many original ideas in rocketry. He was the first to propose the use of ozone instead of oxygen as the oxidizer in a rocket engine. Not knowing of the works of Tsiolkovsky, he repeated a number of his conclusions and also, independently of Tsander, proposed using the combustion of metals to increase the efficiency of rocket engines.

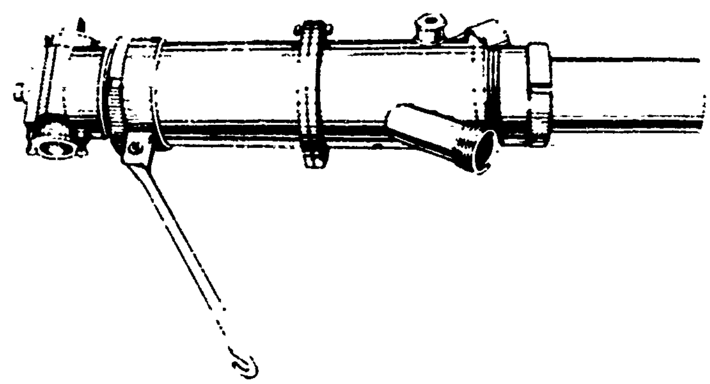


Fig. 0.2 The OR-2 engine.

The questions raised by Tsiolkovsky and Tsander attracted the attention of engineering circles and in 1928-1929 the Group for the Study of Reaction Propulsion (GSRP) was formed by the Voluntary Society for the Promotion of Aviation. Simultaneously with the organization of the GSRP, in May 1929 operations began at the Design Bureau for Liquid Fuel Engines at the Gas-Dynamics Laboratory (GDL).

At the GDL, operations began in May 1929 on the study and design of liquid fuel rocket engines. Particular attention was paid to the selection of the power sources which could be used in rocket engines. As a result of this work, various fuel components, including metals, were proposed.

In 1930 the ORM-1 (Fig. 0.3) was tested; this was the first liquid-fuel rocket engine using nitrogen tetroxide (N_2O_4) and toluene, with a thrust of 20 kg. The engine was still not cooled by circulation, but statically. Further work led, in 1933, to the construction of the ORM-52 (Fig. 0.4), a large engine for that time, with a thrust of 300 kg and working on nitric acid and kerosene. The engine had a system of chemical ignition and circulational cooling of the combustion chamber by the fuel components. By 1936 we had the still more improved ORM-65 engine, also working on nitric acid and kerosene.

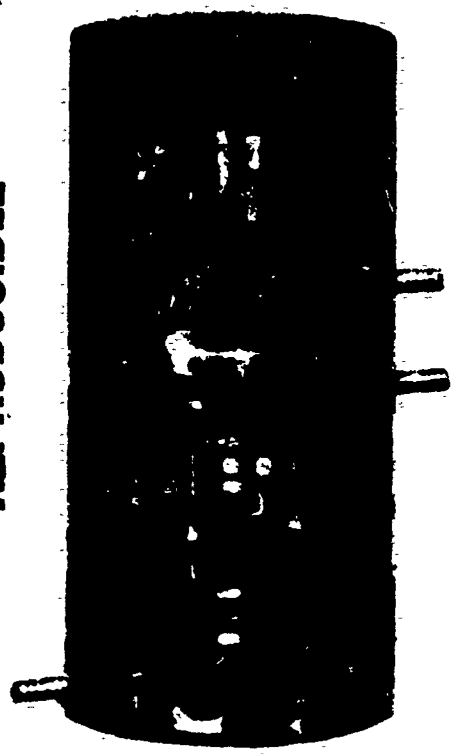


Fig. 0.3. The ORM-1 engine.

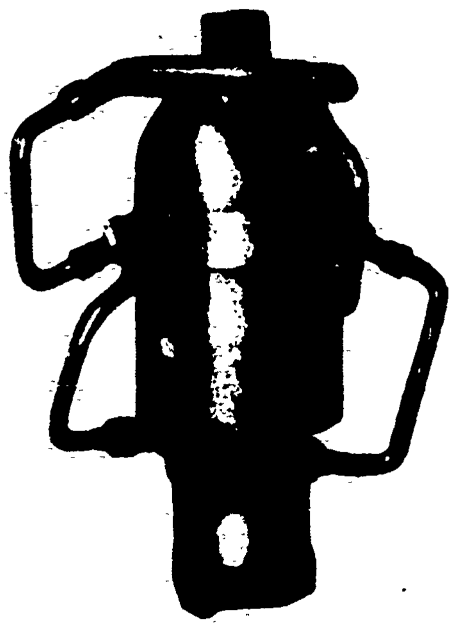


Fig. 0.4. The ORM-52 engine.

We should also mention the 12-K engine, created in the 1930's, having a thrust of 306 kg and working on alcohol and oxygen, with an uncooled ceramic nozzle; the RDA-1-180, working on nitric acid and kerosene; and others.

At the Moscow branch of the GSRP and later in other organizations using this as their basis, much work was done in rocketry, covering questions associated not only with engines, but also with rocket aircraft.

Of the numerous designs we should mention the liquid-fuel rockets, first created by the Soviets. The first of these rockets (Figs. 0.5 and 0.6) was successfully fired in August, 1933.

The rocket weighed 18 kg, 4.45 kg of which was fuel. The engine worked on liquid oxygen and jellied gasoline (napalm) placed directly in the combustion chamber. Six such rockets were launched from August, 1933, through May, 1934.

Yet another liquid-fuel rocket was launched in November, 1933; this weighed 29.5 kg, of which [illegible] kg was fuel (Fig. 0.7). The engine of this rocket, conceived by Tsander and developed by his students and successors, worked on alcohol and liquid oxygen.

A rocket glider (Fig. [illegible]) was produced in 1939; in February, 1940, pilot V. P. Fedorov made a number of successful flights. The glider had an RDA-1-150 liquid-fuel engine.

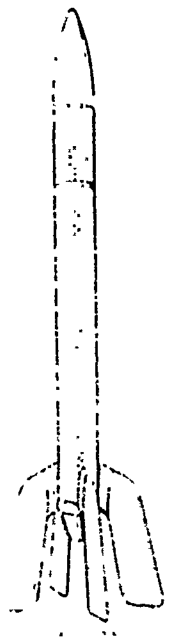


Fig. 0.5. The first Soviet liquid-fuel rocket.

Fig. [illegible]. The rocket in flight.

At this time liquid-fuel unguided rockets with a range of 25 km, designed for military purposes, were produced and successfully tested. During World War II Soviet engineers created new designs of rocket engines and new models of rocket armament, e.g., the widely-known Katyusha [multi-rail rocket launcher].

In addition to the above-mentioned works of Soviet scientists, in the 1920's the first investigations were made abroad on the theoretical problems of space flights, and then liquid- and solid-fuel rockets. Of the Western scientists who [illegible] their efforts to these problems we should mention Esnault-Pelterie (France), whose works were first published in 1913, and Robert Goddard (USA), who began working about 1915, subsequently creating several types of meteorological rockets. Great contributions to the theory of rocket flight were [illegible] by Oberth (Germany) and Sänger (Austria).

Fig. 0.7. The GSRP-Kh rocket.

The best developed rockets of the time were the German V-2 liquid-fuel guided rockets created during World War II.

Fig. [illegible]. Rocket [illegible]

In recent times the Soviet Union has developed a great many types of meteorological and geophysical rockets used to conduct a wide-spread program of upper-atmosphere investigations. The rockets and devices for investigating the upper layers of the atmosphere and outer space are continually being improved, and the research programs are continually expanding.

Soviet designers and scientists are confidently developing and embodying the ideas expressed by Tsiolkovsky. Their great experience served as the basis for the creation of the most advanced and powerful rocket technology in the world.

The banner of Soviet science shone brightly around the world on October 4, 1957, when the first artificial earth satellite was launched. The subsequent launchings of even larger satellites, equipped with the most modern of scientific apparatus, were new triumphs of Soviet rocketry. January, 1959, saw the creation of Lunik I ("Mechta"), the first artificial planet of the solar system. On September 14, 1959, the pennant of the Soviet Union was landed on the moon, and in October, 1959, using the automatic interplanetary station of the third cosmic rocket, the first photographs of the far side of the Moon were obtained.

C H A P T E R I

BASIC RELATIONSHIPS IN THE THEORY OF REACTION PROPULSION

1. Rocket Thrust

The Meshcherskiy Equation

By basic relationships of reaction propulsion we mean the laws of mechanics and the related consequences which determine the propulsion of a rocket. One of the basic relationships is the Meshcherskiy equation, the equation of motion for a point with variable mass.

Let a body with variable mass M have forward motion with velocity v. The mass of the body varies with time due to the continual addition of particles with relative velocity w. When setting up the equation of motion for this mass, we will proceed from Newton's law as the material point: a change in the momentum equals the momentum of the forces acting on the system.

In time Δt, mass ΔM is attached to mass M with relative velocity w. If we assume that this relative velocity is opposite in direction to that of the forward motion of the body, the absolute velocity of mass ΔM will be v - w (Fig. 1.1).

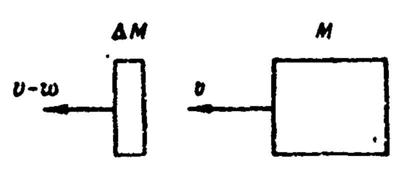


Fig. 1.1. Derivation of the Meshcherskiy equation.

The momentum of the system before the combination of masses was

$$Mv + \Delta M(v - w),$$

and after combination it will be

$$(M + \Delta M)(v + \Delta v).$$

According to Newton's law,

$$(M + \Delta M)(v + \Delta v) - Mv - \Delta M(v - w) = \Delta t \Sigma P_i,$$

where ΣP_i is the sum of the external forces acting on the system in time interval Δt.

This gives

$$M\Delta v + \Delta M w + \Delta M \Delta v = \Delta t \Sigma P_i .$$

Dividing both sides by Δt and approaching the limit, we get the Meshcherskiy equation

$$M(dv/dt) = -(dM/dt)w + \sum P_i . \qquad (1.1)$$

For a constant mass ($dM/dt = 0$) we get the usual expression for Newton's second law

$$M(dv/dt) = \sum P_i . \qquad (1.2)$$

Comparing these two expressions we see that the first can be written in the form of the second if the term

$$-(dM/dt)w,$$

a force, is examined as a force applied to a body with mass M.

When the direction of the relative velocity w is opposite to that of the forward motion of a body v ($w > 0$), the force indicated above will be a braking force (decreasing the velocity v) if $dM/dt > 0$, i.e., if the mass of the body increases. If, however, the mass decreases ($dM/dt < 0$), force $-(dM/dt)w$ will be a propelling force.

For a rocket,

$$dM/dt < 0.$$

The magnitude $-dM/dt$ is called the mass flow rate per second and is designated by m:

$$-dM/dt = m. \qquad (1.3)$$

The relative repulsion velocity w is called the exhaust velocity.

Thus, expression (1.1) assumes the form

$$M\dot{v} = mw + \Sigma P_i , \qquad (1.4)$$

where

$$\dot{v} = dv/dt.$$

Thrust

The equation of motion for a rocket is usually written in the form of Newton's law (1.2):

$$M\dot{v} = \Sigma P_i . \qquad (1.5)$$

Here mass M is considered a time function, defined by the law of the repulsion of mass. Forces P_i include all external forces acting on the rocket: gravity, pressure distributed along the surface, etc. Comparing (1.5) and (1.4) it becomes evident that the reaction force mw obtained above should appear in some form or other under the summation sign.

Of the above-mentioned forces, that created by the engine, viz., the propelling force, is particularly important. This force is called the engine thrust, or simply thrust.

Thrust is the axial resultant of the pressure forces distributed along the entire surface of the rocket. Here, first of all, we include the pressure p_g of the gases exhausted from the rocket engine and acting on its inner surface. In addition, thrust includes the axial force of atmospheric pressure p acting on the outer surface of the rocket (Fig. 1.2).

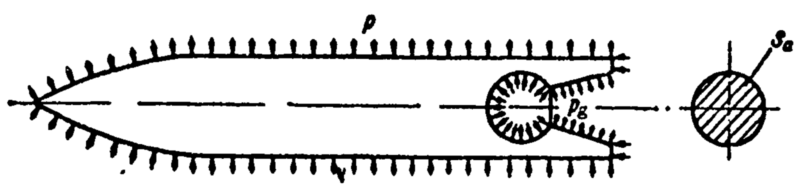


Fig. 1.2. Distribution of pressure along the surface of the rocket (derivation of the thrust equation).

It must be stressed that here we are speaking exclusively of the barometric pressure of the surrounding medium, not of the true pressure on the surface of the rocket; the value and the distribution law of this pressure depend on, in addition to altitude, the flight speed and the aerodynamic shape of the rocket. All additional forces originating due to the movement of the rocket in the atmosphere are usually examined as aerodynamic forces.

Regardless of the shape of the rocket or the engine, if we consider the engine as isolated, the axial component of external static pressure of the surrounding medium is

$$-pS_a,$$

where p is the external static pressure;

S_a is the nozzle exit area (see Fig. 1.2).

The minus sign indicates that this force is always directed against the thrust and is a braking force.

We must now find the resultant of the pressure forces for the inner surface of the engine. To do this, let us examine separately the volume of gas bounded by the inner surface of the engine and the nozzle exit (Fig. 1.3a).

The forces of gas pressure p_g which act on the inner surface of the engine and

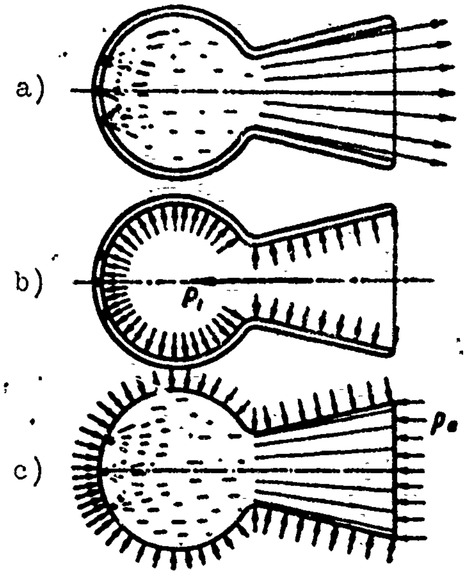


Fig. 1.3. Re: the derivation of the thrust formula.

produce the desired term P_1 (Fig. 1.3b) will act just the same way on the released volume of gas and will yield for it the same resultant, only directed in another direction.

Considering the force arriving at the nozzle exit (Fig. 1.3c), let us find, for the liberated gas volume, the resultant of the surface forces in the form

$$P_1 - p_a S_a,$$

where S_a is, as before, the area of the nozzle exit;

p_a is the pressure in the gas stream at the nozzle exit (this pressure is not necessarily equal to the pressure of the surrounding medium; it can be either greater or less than it).

The impulse $P_1 - p_a S_a$ in time Δt equals the change in momentum of the gas:

$$(P_1 - p_a S_a)\Delta t = -\Delta M w,$$

where ΔM is the mass of gas leaving in time Δt with velocity $\underline{w}$.

Passing to the limit and considering (1.3), let us find the desired value:

$$P_1 = mw + p_a S_a.$$

Adding now the static pressure component found above, we get the following expression for thrust:

$$P = mw + S_a(p_a - p).$$

Since the external static pressure $\underline{p}$ of the medium surrounding the engine is a function of altitude $\underline{h}$, it is best to designate it p_h; then the thrust expression assumes the final form:

$$P = mw + S_a(p_a - p_h). \qquad (1.6)$$

We should note that this expression could have been obtained directly from the Meshcherskiy equation (1.4). Let us assume, e.g., that the rocket (or only its engine) is rigidly attached horizontally to a stand (Fig. 1.4). Then, since the rocket is stationary, $v = 0$. Of the forces acting on the outer surface of the rocket and the nozzle exit, we have the difference

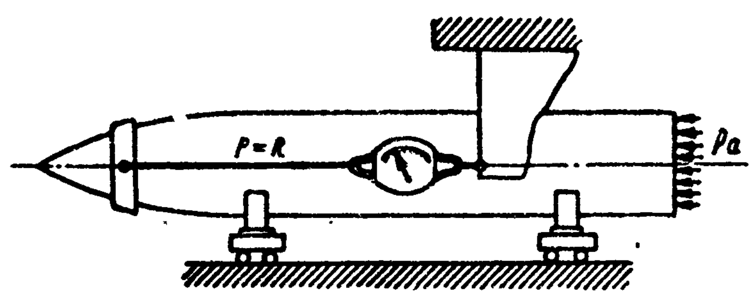


Fig. 1.4. The forces acting on a rocket attached to a test stand.

in pressure forces $p_aS_a - p_hS_a$ and the binding force R applied to the rocket from the stand and equal, in this case, to the thrust P. This force can be measured with a dynamometer.

When the engine is operating at height h, Eq. (1.4) assumes the form

$$0 = mw + p_aS_a - p_hS_a - R,$$

from which we get the expression found above:

$$R = P = mw + S_a(p_a - p_h).$$

With such a derivation, however, we do not explain the system of pressure forces leading to the appearance of thrust.

It has already been mentioned that the thrust component $-p_hS_a$, obtained because of the static pressure component of the surrounding medium, is a function neither of the shape of the rocket nor of the shape of the engine. The value of the gas pressure forces on the inner surface of the engine is also not a function of the shape of the rocket body and, as we shall see (Chapter VI), it is also neither a function of the flight speed nor of the conditions of the surrounding medium.

Thus, the thrust of a rocket engine can be examined as characteristic only for the specific thrust of a given rocket. Most of the thrust belongs to the reaction force component mw.

It follows from an examination of (1.6), by the way, that the rather widespread concept that a rocket engine is repelled from the surrounding air by an exhaust jet is unfounded. Expression (1.6) shows that with a decrease in atmospheric pressure p_h, the thrust not only does not decrease, but even increases. The maximum thrust value P is reached when $p_h = 0$, i.e., in airless space. The rocket vehicle is repelled not from the air but, if we may express it thus, from gas particles exhausted from the combustion chamber. It is because of this repulsion of gas that it acquires its velocity. The greater the exhaust velocity, the more impulse the rocket receives with this same ejected mass.

Expression (1.6) for thrust can be given in another form. Let us use p_0 to designate atmospheric pressure at ground level. Then (1.6) can be rewritten in the form

$$P = mw + S_a(p_a - p_0) + S_a(p_0 - p_h)$$

or

$$P = P_0 + S_a(p_0 - p_h). \tag{1.7}$$

Let us solve this expression for P_0, the thrust at ground level:

$$P_0 = mw + S_a(p_a - p_0).$$

Expression (1.7) gives the engine thrust vs. height, and holds only for a flow rate m which remains constant along the trajectory.

When a rocket flies outside the atmosphere, the external pressure $p_h = 0$ and the value of the thrust in this case P_v (the so-called thrust in a vacuum) is, according to (1.6),

$$P_v = mw + S_a p_a. \tag{1.8}$$

Sometimes the expression for thrust in a vacuum is written in the form

$$P_v = mw_e, \tag{1.9}$$

where w_e is the so-called effective exhaust velocity.

A comparison of (1.9) and (1.8) shows that the effective exhaust velocity

$$w_e = w + (S_a p_a/m). \tag{1.10}$$

The basic feature of the effective exhaust velocity is that it is not a function of flow rate m since, as will be seen in Chapter VI, the pressure p_a at the nozzle exit is proportional to m, while the actual exhaust velocity w is not a function of flow rate (in a relatively small range of flow-rate changes). The difference between the effective and actual velocity is not great; usually it does not exceed 10-15% of w.

Specific Thrust and Specific Fuel Consumption

One of the best indications of the efficiency of a rocket engine is the specific thrust.

By specific thrust we mean the thrust of the engine related to the total weight flow rate per second of the exhausted mass:

$$P_{sp} = P/mg_0,$$

where the weight flow rate mg_0 is taken for conditions reduced to surface level (g_0 is the gravitational acceleration at the Earth's surface).

According to (1.6),

$$P_{sp} = \frac{w}{g_0} + \frac{S_a}{mg_0}(p_a - p_h). \tag{1.11}$$

For a given engine, the specific thrust is a function of the external atmospheric pressure, i.e., flight altitude. Within the limits of a change in pressure from one atmosphere to a total vacuum the specific thrust usually changes by 10-20%.

The specific thrust is determined by the exhaust velocity of the gas, which in turn is a function of the calorific value of the fuel and the structural features of the engine. The greater the calorific value of the fuel, the higher the specific thrust.

The conditions of fuel combustion and gas exhaust change as a function of the design of the engine; this directly affects the specific thrust. For example, the specific thrust is affected by the pressure in the combustion chamber and the degree of nozzle expansion. In addition, in large rocket engines there is a certain amount of mass expended on the internal requirements of the engine, e.g., the expenditure of steam-gas generation components on turbine operation. Because of such an additional mass expenditure the total specific thrust of the entire engine decreases.

Since the specific thrust of a given engine is a function of flight altitude, we must indicate the height for which the specific thrust is given. The least specific thrust for a given engine will occur when the engine is operating at surface level. Such a thrust is designated $P_{sp.0}$. The maximum specific thrust occurs in a vacuum ($p_h = 0$). This is the so-called vacuum specific thrust

$$P_{sp.v} = \frac{w}{g_0} + \frac{S_a p_a}{m g_0} = \frac{w_e}{g_0} \qquad (1.12)$$

or, numerically,

$$P_{sp.v} \approx 0.1 w_e \text{ kg/kg/sec},$$

where w_e is the effective exhaust velocity in m/sec.

In addition to specific thrust, rocket engines can also be described by the specific fuel consumption, i.e., the weight flow rate per second, per unit of specific thrust.

The specific fuel consumption G_{sp} is the inverse of specific thrust, and is usually measured in kg/sec per ton of thrust. Therefore,

$$G_{sp} = \frac{1000}{P_{sp}} \text{ kg/sec/ton.}$$

Here P_{sp} is in kg/kg/sec.

The specific thrust of modern rocket engines is 200-270 kg/kg/sec (for more detail see Table 4.5). The specific fuel consumption is 5-3.6 kg/sec/ton, respectively.

2. The Tsiolkovsky Formula for Ideal Rocket Velocity

The Ideal Velocity of a Single-Stage Rocket

Let us determine the velocity which a rocket can have in the ideal case, when it moves not only outside the atmosphere, but outside the gravitational field as well.

In this case, from expressions (1.5) and (1.9) we get

$$M\dot{v} = mw_e.$$

According to (1.3),

$$m = -(dM/dt),$$

in this case,

$$M(dv/dt) = -w_e(dM/dt),$$

whence

$$dv = -w_e(dM/M).$$

Considering that the effective velocity w_e can be taken as a constant, we get, after integration,

$$v = -w_e(\ln M - \ln C),$$

where C is an arbitrary constant.

When $v = 0$ the rocket mass is equal to the initial mass M_0 (the total structural mass and fuel supply). Therefore,

$$v = -w_e \ln(M/M_0). \qquad (1.13)$$

This relationship was first obtained by Tsiolkovsky. Here M is the current value of the rocket mass.

The ratio M/M_0 is usually designated by μ ($\mu \leq 1$). The inverse number is called the Tsiolkovsky number and is designated by $\underline{z}$ ($z \geq 1$).

By the time of engine cutoff, mass M has assumed its minimum value M_t, equal to the structural mass of the rocket with slight amounts of fuel in the tanks and lines. Here the maximum (terminal) rocket velocity

$$v_t = -w_e \ln \mu_t. \qquad (1.14)$$

Often this formula is written thusly:

$$v_t = w_e \ln \frac{1}{\mu_t} = w_e \ln z_t = w_e \ln \frac{M_t + M_f}{M_t},$$

where M_t is the rocket mass at the moment of engine cutoff;

M_f is the mass of the burned fuel.

As follows from the basic relationship for terminal velocity (1.14), greater velocity v_t can be attained either by increasing the effective exhaust velocity w_e or by decreasing the relative terminal mass μ_t.

Let us estimate the relative influence of w_e and μ_t on the ideal terminal velocity. If we differentiate (1.14), first taking the natural logarithms of both sides and replacing the differentials by finite increments, we get

$$\frac{\Delta v_t}{v_t} = \frac{\Delta w_e}{w_e} + \frac{1}{\ln \mu_t}\frac{\Delta \mu_t}{\mu_t}\ .$$

From this we can conclude that the effective exhaust velocity w_e has a relatively greater effect on the ideal terminal velocity v_t, provided

$$|1/\ln \mu_t| < 1,$$

i.e., when

$$\mu_t < 1/e \approx 0.37,$$

where e is the base of natural logarithms.

Above it was stated that the effective exhaust velocity w_e is determined by the calorific value of the fuel and the quality of the engine.

The value μ_t determines the quality of the rocket design. The smaller the μ_t the better is the design and the greater the velocity the rocket can attain with a given exhaust velocity w_e.

We can get some idea of the true values of μ_t if we remember, e.g., that the launch weight of the V-2 rocket was about 13 tons with a structural weight (without fuel) of approximately 4 tons. Accordingly, for this rocket $\mu_t \approx 0.3$. If we consider that the design of this rocket was far from perfect and could have been improved, this value should be treated only as some low, easily attainable indication of the design quality. By the same token, for single-stage rockets it is difficult to imagine the possibility of sharply decreasing μ_t (if only to a value of the order of 0.1), especially if we consider that the weight of an unfueled rocket should include the payload (instruments or crew).

Thus we can consider that the value of μ_t for actual designs of single-stage rockets can lie within limits 0.3-0.1.

Let us again turn to the V-2, which operated on aqueous ethanol and liquid oxygen; for the effective exhaust velocity from the engine nozzle we get a value of approximately 2000 m/sec. Thus, for this rocket, according to (1.4), the ideal terminal flight velocity

$$v_t = -2000 \ln 0.3 \approx 2400 \text{ m/sec.}$$

Actually, because of gravity and the atmosphere, the maximum velocity dropped to 1500 m/sec (a loss in speed, roughly speaking, of approximately 1000 m/sec).

We have seen that the most effective method for increasing the terminal velocity of rockets is to increase the effective exhaust velocity, i.e., specific thrust. If we increase the effective exhaust velocity to 4000 m/sec, which is the limit for the power capability of chemical rocket fuels, and at the same time increase the index of structural quality μ_t to 0.12,

$$v_t = -4000 \ln 0.12 \approx 8500 \text{ m/sec.}$$

If we consider the unavoidable losses in velocity caused by gravity and the Earth's atmosphere, the obtained terminal velocity will be less than circular (see Chapter II).

Multi-stage rockets, first proposed by Tsiolkovsky, make it possible to attain cosmic velocities.

The Ideal Velocity of a Multi-Stage Rocket

The main problem concerning rockets as flying vehicles is to impart a specific velocity to a specific load (rocket crew, instruments, or warhead).

The fuel supply is determined on the basis of the given load and specific velocity. The greater the load and the desired velocity, the more fuel the rocket must contain and, consequently, the greater the structural weight of the rocket.

This is due, first, to the fact that the weight of the fuel tanks increases as their volume and, in addition, as the rocket becomes larger the more difficult it is to assure its structural strength; this makes it necessary to complicate the design and additionally increase the weight.

Here we have the main disadvantage of ordinary single-stage rockets. The velocity in these rockets is imparted not only to the payload but to the entire rocket, which necessarily causes "waste" expenditures of power. At the same time, these expenditures are so great that, as we have seen, single-stage rockets operating on ordinary chemical fuels generally cannot attain cosmic velocities.

Multi-stage rockets, for the most part, do not have this drawback. By multi-stage rocket we mean, generally speaking, one in which during its flight, when the entire fuel supply has not been consumed, the spent structural elements and those

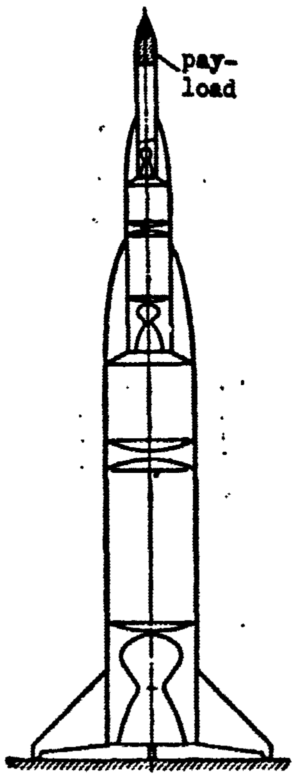


Fig. 1.5. Diagram of a multi-stage rocket.

not needed for further flight are jettisoned. Figure 1.5 shows the simplest type of multi-stage rocket.

At the outset, during the rocket ascent, the first, most powerful engine operates – the first-stage engine – capable of lifting and accelerating the entire system to a certain velocity. After the main fuel supply has been consumed, the first-stage engine, together with the empty fuel tanks, i.e., with the entire first stage of the rocket, can be jettisoned. Then the rocket flight continues using the second-stage engine; this has less thrust but can impart additional velocity to the lightened rocket.

After the second stage burns out the third-stage engine can be turned on and the second stage jettisoned from the rocket. This process of stage separation can theoretically be continued even further. In practice, however, the number of stages depends on the purpose of the rocket and the degree of structural difficulties which arise as more stages are added.

As opposed to a single-stage rocket, at the same time as the payload receives a specific terminal velocity, this same velocity is imparted to the mass not of the entire rocket, but only to the last stage. The masses of the previous stages receive less velocity.

Let us designate by μ_1 the ratio of the mass of the first stage of the rocket without fuel to the launching mass of the entire rocket; by μ_2 the ratio of the mass of the second stage without fuel to the mass which the rocket had immediately before the spent first stage was jettisoned. Analogously, for subsequent stages we will use the designations $\mu_3, \mu_4, \ldots, \mu_n$.

After the fuel of the first stage has been expended, the ideal velocity of the rocket will be

$$v_1 = -w_{e1} \ln \mu_1.$$

After the fuel of the second stage has been expended, the velocity

$$v_2 = -w_{e2} \ln \mu_2$$

is added to the previous quantity. Each subsequent stage adds an additional velocity. Finally, we get

$$v_t = -w_{e1} \ln \mu_1 - w_{e2} \ln \mu_2 - \dots - w_{en} \ln \mu_n.$$

If the effective exhaust velocity of each of the stages is the same ($w_{e1} = w_{e2} = \dots = w_{en} = w_e$), then

$$v_t = -w_e \ln (\mu_1\mu_2\dots\mu_n).$$

Each of the values of μ is less than unity. With several stages, the product $\mu_1\mu_2\dots\mu_n$ can be rather small and the value $-\ln (\mu_1\mu_2\dots\mu_n)$ can reach high values; this corresponds to a high terminal velocity.

Cosmic flight velocities can be attained by using multi-stage rockets.

The Tsiolkovsky formulas for ideal flight conditions give only the upper limit of the rocket velocity. The actual terminal velocity will always be less, due to unavoidable losses. The following factors contribute to a decrease in terminal velocity:

a) the overcoming of gravity when the rocket is launched;

b) the decrease in specific thrust when the engine operates in the atmosphere (compared to the vacuum specific thrust which determines the effective exhaust velocity);

c) the overcoming of aerodynamic drag; and

d) the need for maintaining a definite trajectory — at certain points along the trajectory the thrust orientation may not coincide with the velocity orientation.

Several of these questions will be studied in more detail further on in the book.

When the engine thrust is large compared with the weight of the rocket and the aerodynamic drag (e.g., for tactical powder rockets), the Tsiolkovsky formula is sufficiently accurate. Even for long-range rockets for which the velocity losses to gravity may be quite considerable, the Tsiolkovsky formula makes it possible to roughly estimate the velocity resources of a rocket.

The Influence of Gravity

Let us now explain how the Earth's gravity affects the movement of a rocket for the simplest case of vertical ascent.

In this case, expressions (1.5) and (1.9) for flight in airless space give

$$M(dv/dt) = w_e m - Mg,$$

where Mg is the weight of the rocket.

If the distance from the rocket to the center of the Earth changes insignificantly during the operation of the engine, the value g can be considered constant. Then this latter expression, after transformations, assumes the form

$$dv = -w_e(dM/M) - g\,dt$$

and is easily integrated:

$$v = -w_e(\ln M - \ln C) - gt.$$

As in the case of ideal flight, when $t = 0$ we have $v = 0$ and $M = M_0$. Consequently, $C = M_0$ and

$$v = -w_e \ln \mu - gt.$$

This expression for vertical ascent could have been written immediately. To do this we need only consider that due to gravity the ideal velocity v decreases by an amount equal to that obtained by a free-falling body in time t (in our case, t is the operation time of the engine).

Let us estimate the velocity lost to gravity for a vertical ascent of the V-2, whose engine operated for 64 seconds. We get $gt \approx 640$ m/sec using the above-computed ideal velocity of 2400 m/sec.

In the general case, when a rocket flies along an inclined trajectory, the velocity lost to gravity is somewhat less than for a vertical ascent, where the engine has the same operating time.

Thus, in the ideal case the terminal velocity of a rocket is independent of the fuel combustion regime (time) and is determined only by the ratio of the terminal and initial mass of the rocket, while if gravity is present, the velocity which the rocket obtains depends on the fuel consumption rate. The faster the fuel is consumed, i.e., the smaller time t, the greater will be velocity v.

We can picture a case where the rocket velocity will be equal to zero. This happens, e.g., if the fuel burns gradually with a low flow-rate per second, and the engine produces a thrust not exceeding the weight force.

In some way or other we must try to get an engine which will produce maximum thrust. However, the thrust cannot be increased limitlessly. First, an increase in thrust is accompanied by an increase in the weight of the engine, which can result in a considerable burdening of the rocket, i.e., an increase in its terminal weight μ_t.

In addition, great thrusts cause great accelerations and, accordingly, great inertial forces on the rocket. This affects its strength and, in final analysis, again results in increased weight of the rocket. If the rocket is designed to carry living beings, the acceleration must be limited, for safety reasons.

In addition to all these, for long-range rockets, a considerable part of whose trajectories lies outside the atmosphere, a rapid increase in velocity would lead to considerable losses to aerodynamic drag. For such rockets it is more suitable to have a moderate range of velocities so that the lower dense atmospheric layers will be traversed at low speeds (with low aerodynamic drag). The upper layers of the atmosphere, however, can be crossed at high speeds.

All these factors must be taken into consideration when designing specific rockets.

3. The External Efficiency of a Rocket Engine

Qualitatively, the efficiency of a rocket engine is analogous to that of any other heat engine, and is defined as the ratio of useful output to total input.

The losses of a rocket engine consist of heat losses caused by internal processes occurring in the engine, and losses specifically associated with the departing gas jet which removes part of the kinetic energy of the system.

The internal efficiency of a rocket engine is the ratio of the kinetic energy of the exhausted gases to the calorific value of the fuel. Its value is determined by the properties of the fuel and the operation of the burning process, and varies between 0.3 and 0.5 (in the best case). The factors which affect the value of the internal efficiency will be discussed in the section devoted to processes in the combustion chamber and the nozzles. Here we shall discuss only the external efficiency. By this we mean the ratio of the useful output of a gas jet to its kinetic energy.

Let us set up an expression for the external efficiency of a rocket engine. Let us assume that in time Δt, fuel mass ΔM burns; this is then exhausted from the nozzle with velocity $\underline{w}$. This mass has kinetic energy

$$\Delta M w^2/2.$$

Before combustion, the fuel had energy

$$\Delta M v^2/2,$$

where $\underline{v}$ is the rocket velocity at a given moment.

Thus, the mass leaving the rocket absorbed energy

$$\Delta M w^2/2 + \Delta M v^2/2. \qquad (1.15)$$

The second term corresponds to the preliminary expenditure of energy in order that mass ΔM acquire velocity $\underline{v}$. Before the fuel is ignited, some kinetic energy is imparted to it due to the burning of other fuel particles. Because of this, the mass of fuel becomes "more expensive" in power rating.

Let us now see how much of this energy is useful energy.

Mass ΔM exhausted in time Δt creates thrust

$$(\Delta M/\Delta t)w$$

[we disregard the thrust component $S_a(p_a - p)$]. On path $v\Delta t$ the thrust produces work

$$(\Delta M/\Delta t)wv\Delta t = \Delta Mwv. \tag{1.16}$$

This work will be useful work obtained from the total kinetic energy of mass ΔM. Accordingly, the external efficiency of a rocket engine

$$\eta_e = \frac{\Delta Mwv}{\frac{\Delta Mw^2}{2} + \frac{\Delta Mv^2}{2}},$$

or

$$\eta_e = \frac{2\frac{v}{w}}{1 + \frac{v^2}{w^2}}. \tag{1.17}$$

The external efficiency, as we have seen, is a function of flight velocity $\underline{v}$ (Fig. 1.6). When $v = w$ it attains its maximum value $\eta_{e\ max} = 1$. This is understandable. The ejected particles do not have velocity relative to the starting point, and all the kinetic energy of the jet becomes rocket energy. However, from this we cannot draw far-reaching conclusions as to the correspondence between the velocity of the planned rocket and the exhaust velocity of gases. But the fact of the matter is that the efficiency power rating of the quality of the machine is a relative estimate or machines designed to solve only similar problems. The rocket is a unique apparatus which makes it possible to attain cosmic velocities. If this problem must be solved and there is no choice of methods for solving it, the question of external efficiency must naturally be relegated to second place.

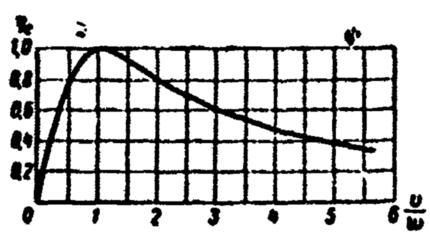


Fig. 1.6. External efficiency of a rocket vs. the ratio of flight velocity to gas exhaust velocity.

However, if in the future it will be possible to create an engine with various exhaust velocities, ones which differ sharply in magnitude, the selection of the sequence of operation of these engines must be linked to a program for changing the velocity of the rocket.

We should mention that (1.17) gives the "instantaneous" value of the external efficiency. To determine the efficiency for the entire flight with a working engine we must, evidently, use not the ratio of (1.16) to (1.15), but the ratio of the integrals of these values throughout the flight.

CHAPTER II

TYPES OF ROCKET VEHICLES AND FUNDAMENTALS OF THEIR CONSTRUCTION

1. Pilotless Rockets and Their Purpose

At present we use the word "rocket" in its broadest sense to indicate the general class of so-called pilotless vehicles, i.e., vehicles which do not have a pilot on board to control the movement. We say "at present" because we know that the day is not far off when man will have advanced so far in rocketry that he will be able to use rockets as transportation media and for space flights.

In rocketry, as in any other rapidly developing field, there is still no established generally-accepted terminology. Often the same type of vehicle has different names, while for some types there is no generally-accepted nomenclature whatsoever. We do not intend to give here any sort of general classification of types of rocket vehicles. Such a classification would be very difficult at present and, no matter how carefully done, would contain many disputable points. Therefore, we will treat the problem of rocket classification only to that extent which is necessary to preserve a specific sequence in describing the general types of vehicles.

The shape of a rocket, its dimensions, and construction are determined, in final analysis, by its mission. Therefore, we will give a survey of the existing types of rockets, adhering to the principle of the mission of the rockets. Let us first examine ground artillery devices, which we will divide into short-range, long-range, and extralong-range missiles. Then we will examine the basic principles of antiaircraft rockets, aircraft rockets, and glide bombs, and we will give a brief description of special rockets. In conclusion we will give a survey of present-day meteorological and geophysical rockets and discuss the basic problems in the creation of artificial earth satellites.

It is not entirely possible to classify rockets according to their purpose. We can divide them according to their target homing and guidance principles. We can classify them according to the type of fuel used, structural features, engine types, etc.

Depending on the place of launch and the target, military pilotless vehicles designed to destroy this target are often divided into the following classes: "surface-to-surface," "surface-to-air," "air-to-air," "water-to-water," etc. The first word indicates the point of launch and the second — the location of the target.

This gives rise, however, to certain inconsistencies. For example, it is not clear how to classify vehicles which can be launched from various places toward the same type of target, or from the same launch site toward various types of targets.

Despite the various types and designs of rockets, and pilotless vehicles in general, they all have certain characteristic structural elements.

Each such vehicle has a propulsion unit which creates thrust according to the reaction principle. The propulsion unit contains the engine chamber (combustion chamber with nozzle, and a cooling system), a system for supplying fuel to the combustion chamber, and an engine control system.

All rockets have fuel containers; in some cases, e.g., the solid-fuel rocket, the combustion chamber itself serves as the fuel-storage device.

To assure correct flight we must have a guidance system to stabilize the rocket in flight. Therefore most vehicles have control or stabilization systems. These contain instruments which react to the attitude of the rocket in space, and can be located either in the rocket or on the ground (if the rocket is ground-controlled). The control system also contains auxiliary, amplification, and servo mechanisms. Finally, we have the executive controls: the control actuators with aerodynamic and gas-jet vanes, rotating engine chambers, and their drivers. If the rocket is unguided, it is provided with either aerodynamic stabilization using rear fins, or gyro-stabil[illegible]on by rapid rotation about the longitudinal axis.

Every vehicle must have a payload compartment to carry to the target. If the rocket is designed for military purposes, this is usually a warhead. If the vehicle is to be used for research purposes, the payload is the research apparatus and radio-transmitting devices.

All the parts of the vehicle must be firmly connected by the structural housing; in man[illegible] cases this may be the walls of the fuel tanks, while in solid-fuel rockets it can also be the walls of the combustion chamber.

Finally, we must not forget the ground equipment complex. For small unguided rockets the ground equipment includes the launchers, a system of simple runners mounted on a truck or an airplane. Long-range liquid-fuel rockets are launched from a launching pad using a complex system of auxiliary launch units for transporting the rocket to the pad and lifting it into place, fuel transportation and service, etc. Ground equipment also includes apparatus for checking the launch of the rocket, and controlling and observing it in flight.

These vehicle elements can be seen from the examples to be given in this chapter. Subsequent chapters will be devoted to a more detailed examination of the individual elements. The operation and design of the propulsion units, and the working processes in the engine, will be examined in Chapters III-VI. General information on the forces acting on a rocket, and flight conditions, will be given in Chapters VII and VIII. Chapter IX contains the general principles for rocket control and stabilization during flight. Finally, Chapter X describes the ground equipment.

In this book we examine rocket vehicles exclusively. Pilotless vehicles equipped with jet engines, e.g., self-propelled missiles, constitute a separate branch of rocketry.

Let us now turn to a more detailed examination of typical rockets.

2. Short-Range Ground Artillery Missiles

Ground artillery rocket vehicles can be divided into short-range, long-range, and extralong-range missiles.

For the destruction of targets located up to 50 km away, i.e., at distances attainable by ordinary artillery, unguided rockets are usually used; because of their simplicity and their concentrated firepower, they can conduct effective area fire. The fuel for these rockets is rocket powder, a variety of solid propellant widely used in rocket engines for many years.

In our terminology, this is a short-range rocket.

The typical weapon of this type is the Soviet "Katyusha," used in World War II. This type also includes the various powder rockets used in other countries during the war. Figure [illegible] shows a cross section of a typical powder rocket projectile.

The rocket has combustion chamber 1 containing the powder rocket charge. This

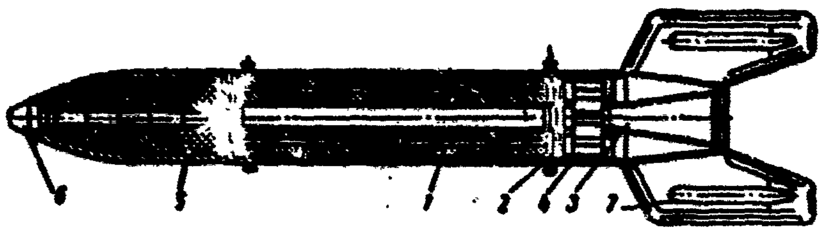


Fig. 2.1. Powder rocket projectile. 1) combustion chamber with rocket charge; 2) igniter; 3) nozzle; 4) diaphragm; 5) warhead; 6) fuze; 7) tail fin.

charge usually consists of several grains and is ignited by special igniter 2. The gases formed by the burning of the powder exhaust through nozzle 3, causing thrust. So that during the combustion process the grains are not shifted lengthwise and bits of powder pass out through the nozzle, the rear of the combustion chamber contains a metal grid, diaphragm 4. The powder rocket projectile has tail fin 7 to give it flight stability.

This is not the only way of keeping the rocket stable. Short-range powder rockets are very often stabilized by giving the rocket great spin. In this case the rocket retains its flight stability much in the manner of an artillery shell, due to the gyroscopic effect. Such a rocket is called a spin-stabilized rocket. The rocket in this case has no tail assembly; the rapid rotation is caused by the action of inclined nozzles. Figure 2.2 shows the structure of the spin-stabilized mine for the six-barrel German mortar of World War II, and a 380-mm spin-stabilized rocket.

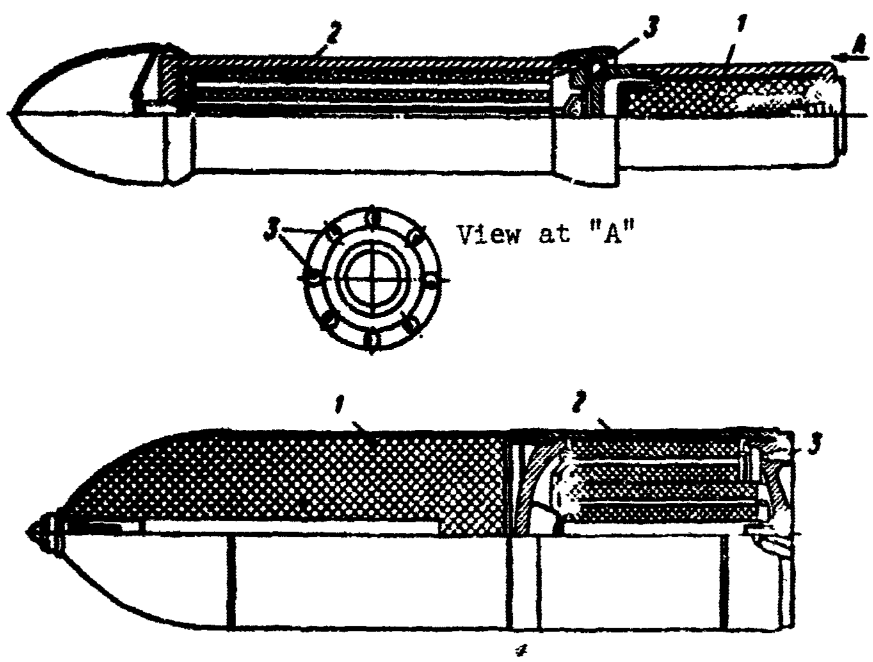


Fig. 2.2. Spin-stabilized rockets. 1) warhead; 2) moving charge; 3) nozzles.

Both missiles have inclined nozzles. It is interesting that in the first of these the gases are exhausted through side nozzles, with the warhead behind.

This was done in order that the nozzle be as close as possible to the center of gravity of the missile, and thus decrease the influence of the random destabilizing moments which occur due to geometric irregularities in the nozzles. In addition, the placement of the warhead behind the rocket charge was intended to increase the combat effect, since the explosion occurs at a certain height above the ground. However, such a design did not work out in actual practice.

Rocket artillery shells have a warhead and a fuze with a detonator, designed according to the purpose of the missile. The rockets shown in Figs. 2.1 and 2.2 were intended to be used against enemy troops.

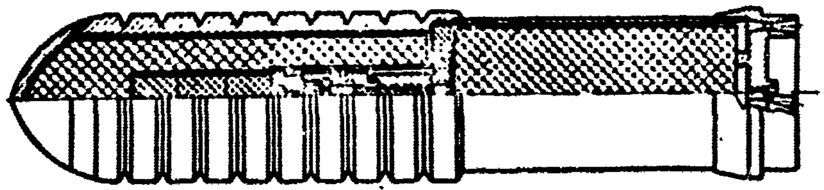


Fig. 2.3. Fragmentation missile.

The fragmentation effect of rockets can be improved by using a massive body with a longitudinal incision as is shown by the spin-stabilized missile in Fig. 2.3. This produces more uniform fragmentation during the explosion.

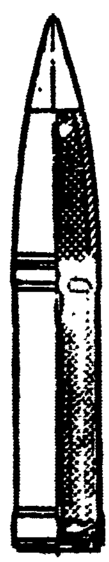


Fig. 2.4. High-explosive fragmentation missile.

Figure 2.4 shows a high-explosive fragmentation rocket. The warhead is included in the massive body, making it able to penetrate thick plating.

The head of the high-explosive incendiary missile shown in Fig. 2.5 contains the high-explosive incendiary elements. The head of the incendiary rocket contains a fuel mixture (Fig. 2.6) for a purely incendiary effect.

Let us mention the main characteristic of the rocket shown in Fig. 2.5. The combustion chamber is made of refractory ceramics, wound on the outside with a thin layer of piano wire. Such a design has a certain weight advantage, since the cold-drawn wire has noticeably high strength compared with ordinary pipes. However, because of technological complications, the high cost of the wire, and the unsatisfactory reliability, such chambers were not used.

Figure 2.7 shows the construction of two anti-tank rockets, often called grenades. Their characteristic feature is the shape of the warhead, which has a depression in front. This is the so-called shaped charge.

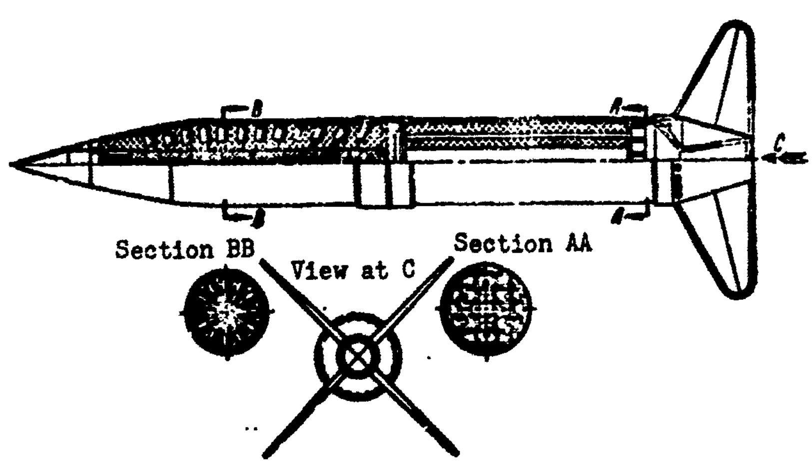


Fig. 2.5. Fragmentation incendiary rocket with the combustion chamber reinforced with fine wire.

The form of the blast wave depends greatly on the shape of

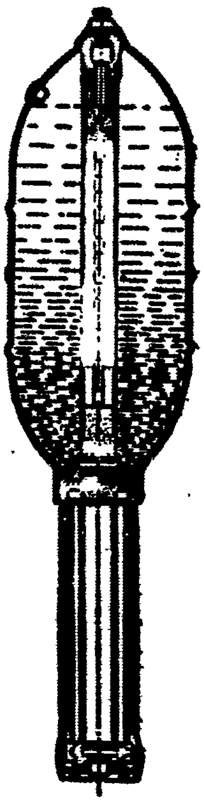


Fig. 2.6. Incendiary rocket.

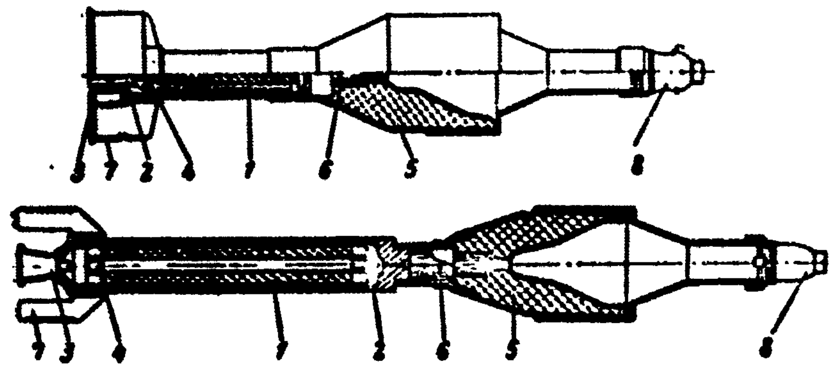


Fig. 2.7. Anti-tank rockets. 1) combustion chamber; with rocket charge; 2) igniter; 3) nozzle; 4) diaphragm; 5) warhead; 6) base detonator; 7) stabilizer; 8) nose fuze.

the charge and the strength and mechanical characteristics of its covering. When the shaped charge explodes, the wave from the depression is concentrated in the form of a thin beam with great armor-piercing ability. The front part of the casing, covering the depression of the shaped charge, is crumpled upon impact and forms the so-called pestle. This pestle, shown in Fig. 2.8 by the dashed lines, is propelled together with the explosion products at a very high speed, and intensifies the destructive power of the shaped charge.

When examining the various types of military rockets, we must not forget those with atomic (nuclear) warheads. The nuclear charge, as we know, is relatively heavy; therefore, large-calibre missiles are needed to deliver it. One of the powder rockets designed for this purpose is the Honest John rocket (Figs. 2.9 and 2.10).

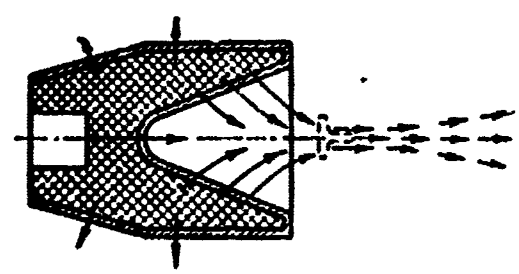


Fig. 2.8. Explosion of shaped charge.

The rocket is designed for direct support of ground troops and has a range of 32 km. The launch weight of the rocket is 2700 kg, the nose with the warhead weighs 630 kg, and the fuel weighs 900 kg. Before launch, the rocket is heated by means of a jacket which contains electric heating elements. This is necessary to decrease the range deviations which could occur with a change in the properties of the soild propellant if it assumes the variable temperature of the surrounding medium (see Chapter IV). Figure 2.9 shows the rocket at the moment the

Fig. 2.9. The Honest John rocket.

heat-control jacket is being removed.

All the powder missiles examined above are unguided. The homing on the target is accomplished by the launching device similar to that when an artillery shell is fired. In this case it is quite natural that such a guidance system should have been selected. For a short range and relatively large targets there is no need for the rocket to be guided, to be equipped with expensive apparatus; this would complicate the launching device and all the other equipment. When a rocket is not expensive, firing inaccuracies can be compensated by number of projectiles, which, in final analysis, is more suitable.

Fig. 2.10. Launching of the Honest John.

However, in certain cases, even for short-range firing, it is desirable to use a guidance system for more accurate homing of the rocket on the target and to correct its flight. This makes sense when firepower is directed toward a moving target (e.g., a tank) or an object which requires more than one hit to destroy. In these cases it is difficult to compensate for errors in unguided missile fire by the number of rounds, not to mention the fact that, e.g., there is usually no time for systematic area coverage during a tank attack.

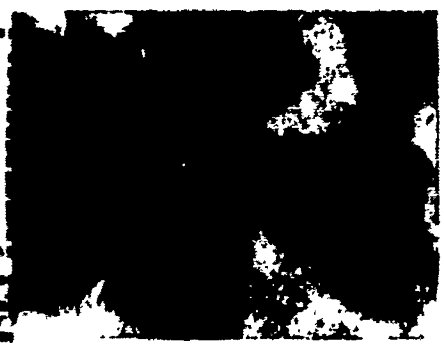


Fig. 2.11. The Dart, a [illegible]

Figure 2.11 shows the Dart, a modern guided short-range anti-tank powder rocket; its design is based on a German missile of World War II.

The missile has +-shaped wings and x-shaped stabilizers, [illegible] kg, [illegible] 1.6 m long, has a flight range of 4.8 km, and a maximum effective range of 900-1800 m.

The rocket receives signals during flight not by radio, but by two very strong thin steel wires which unwind from spools located beneath the wings. This type of signal transmission, as opposed to a radio system, requires somewhat less complicated apparatus and is impervious to external random artificially-created radio interference. At the same time, the system has a certain unreliability, in that the wires may break due to jamming of the spools or if a shell fragment hits the wires. The wire may also break because of the blast wave.

At great distances, a guidance system is completely necessary even for area fire. Long-range rockets are two expensive for quantity to compensate for the large scatter of unguided rockets. Therefore, rockets with a range of more than 50 km are guided, as a rule.

3. Long-Range Rockets

Ways of Increasing the Range of Powder Rockets

All classes of rocket missiles with a range of up to several hundred kilometers are usually called tactical missiles or vehicles. Longer-range missiles are classified as strategic weapons.

Each of these classes can be further subdivided into short-, intermediate-, and long-range rockets. True, such a division at present is not standardized, and therefore the terms "short," "intermediate," and "long" range are usually used in a comparative sense. However, for the most part the available literature uses the term long-range rocket to indicate all rockets with a range of more than 100 km. Military rockets with a range greater than 6000-8000 km are called intercontinental ballistic missiles (ICBM's).

An increase in terminal velocity, i.e., rocket flight range, without changing the weight of the payload, unavoidably leads to an increase in the fuel supply. This increase is the greater, the heavier the payload. There is a simultaneous increase in the time required for the fuel supply to be consumed.

We have seen that for all powder rockets, as well as for all solid-fuel rockets in general, the combustion chamber is also the container for the entire fuel supply. On the one hand, this is very convenient, since the rocket design is as simple as possible, which also simplifies its operation. On the other hand, such a design places a definite limitation on the amount of fuel and its burn-out time. The fact of the matter is what a large amount of fuel requires a larger combustion chamber. But the chamber is at high pressure and operates under adverse temperature conditions. It may be that an excessive weight increase results if a larger chamber is to be strong at high temperatures and pressures.

In addition, another essential fact is that the walls of the combustion chamber and the walls of the nozzle of the powder engine are not capable of being cooled. When the burning time of the charge is short, the walls do not heat up very much and they retain their strength almost completely. As the burning time increases, it may be necessary to cool the walls of the combustion chamber and nozzle, or at least prevent them from heating, which would lead to a complex adverse structure, and it would lose its prime advantage.

Therefore, with large fuel supplies and long duration of engine operation, it is more expedient to use a liquid, rather than solid, fuel. Another advantage of this is that a liquid fuel usually has a higher calorific value.

Thus we can say that with an increase in the payload of a solid-fuel rocket (compared with a liquid-fuel rocket) there is a sharp limitation on the terminal velocity, or the flight range. We can also say that with an increase in terminal velocity, a solid-fuel rocket must have a smaller payload.

Solid-fuel rocketry has advanced considerably in the past few years. Fuels with high calorific values have been developed which burn stably at relatively low pressures [of the order of 20-40 atm (tech.)], while pyroxylin- and nitroglycerin-base powders require a pressure of 60-80 atm (tech.). Methods have been developed for thermally insulating the walls, making it possible for a rocket engine to operate without cooling for several tens of seconds.

This has led to the fact that new solid-fuel engines have begun to be used in such rockets as previously used liquid-fuel engines. These include tactical rockets with ranges up to 500 km, guided antiaircraft missiles, the second (and particularly the third) stages of geophysical and long-range ballistic missiles, and, finally, the carrier rockets for artificial earth satellites. Solid-fuel rockets are also used as launch boosters of various types. The solid-fuel charges in the Polaris, for example, weigh up to several tons.

Nevertheless, if we are speaking of the relative possibilities of increasing the payload with a simultaneous increase in terminal velocity, the above limitations which [illegible] imposes on the range or payload of a rocket remain valid, to a

Fig. 2.12. Long-range multi-stage powder rocket.

considerable extent.

Within the limitations imposed by solid fuel, the range can be increased considerably by using multi-stage rockets. The four-stage unguided powder rocket shown in Fig. 2.12 is an example of such a rocket.

The rocket is launched from inclined rails. When the fuel in the first stage is expended, the first (rear) engine is dropped and the three remaining stages continue the flight. Then the second and third stages are dropped, in succession.

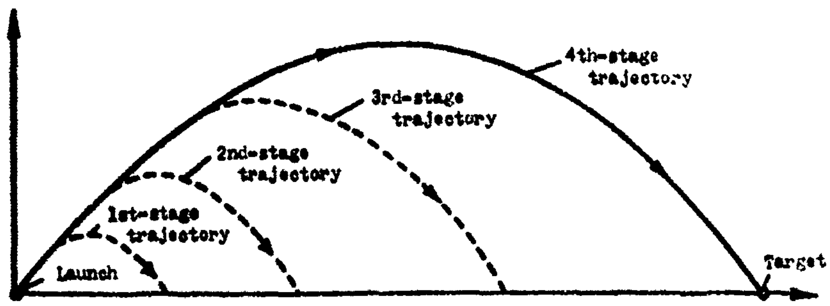


Fig. 2.13. Trajectory of a four-stage rocket.

Figure 2.13 shows the trajectory of a rocket. The range of the rocket is approximately 170 km. Only the nose of the rocket, the fourth stage carrying 40 kg of explosives, reaches the target. The total weight of the powder charge which assures the indicated range is 585 kg.

Since the flight is unguided, its error over a considerable range must be very great, and it does not produce the desired effect. Therefore, it is not accidental that this rocket, created during World War II, was never used.

The flight range of a rocket can be increased by a combination of the weapons and reaction principles of firing.

Figure 2.14 shows the so-called active-propulsion missile. This is fired from ordinary weapons. The missile acquires additional velocity because of the combustion of the rocket charge located in the base.

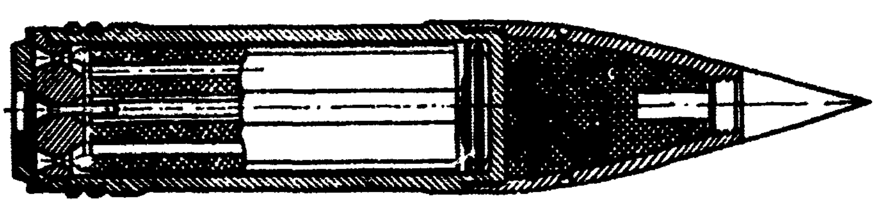


Fig. 2.14. Active-propulsion missile.

This method makes it possible to noticeably increase the range, although at the expense of the military effect (compared with an artillery shell of the same weight).

The use of rocket fuels with higher calorific values than powder makes it possible to attain greater range, with a simultaneous increase in the weight of the payload. Present-day rocketry is developing along these lines.

Long-range rockets use liquid fuels with maximum calorific value: ethanol, gasoline, and kerosene, oxidized by nitric acid, liquid oxygen, or special types of oxidizers.

A long-range vehicle should have considerably greater absolute dimensions than short-range vehicles. This is quite evident from the fact that more fuel is required for greater ranges with the same warhead. In addition, with an increase in range, the warhead itself should be enlarged. Clearly, it makes no sense to use a rocket with a range of 1000 km, and load it with only 10-20 kg of explosives. As for a nuclear warhead, there must be a limit to the payload.

At present, the launch weight of long-range vehicles is several tons, while that of ICBM's is tens of tons. For example, the initial weight of the Atlas rocket is greater than 100 tons. The weight of the warhead of such a rocket varies from 200-300 kg to several tons.

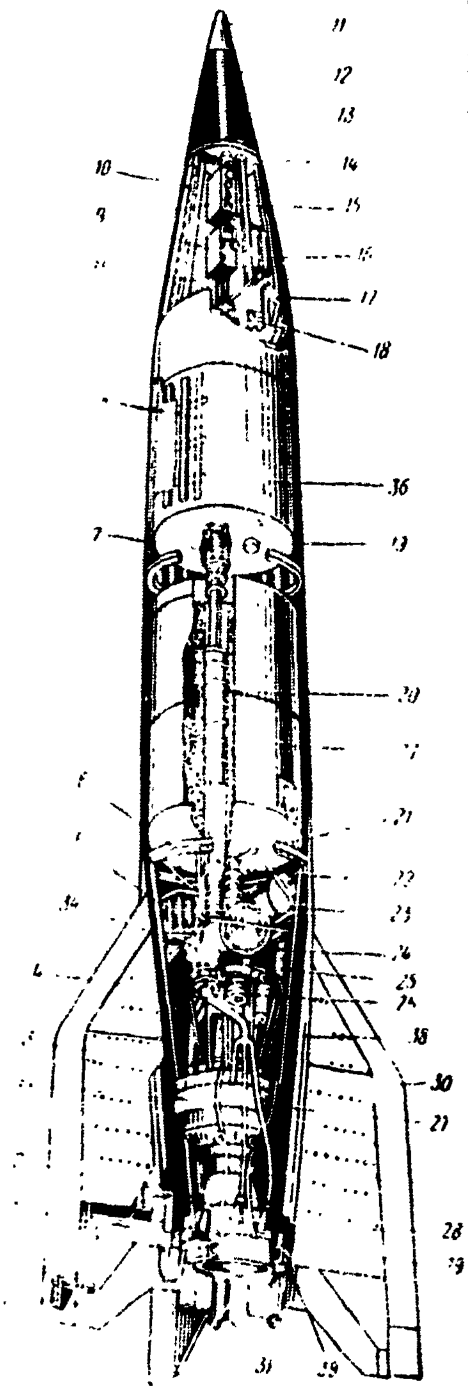


Fig. 2.15. Long-range ballistic missile. 1) chain transmission to the rudders; 2) rudder motor; 3) precombustion chamber; 4) ethanol pipeline to combustion chamber; 5) air cylinders for the pneumatic system of the propulsion unit; 6) rear ring; 7) servo valve; 8) fuel compartment case; 9) guidance system instruments; 10) pressure-feed pipeline for the ethanol tank; 11) tip with nose fuze; 12) warhead; 13) tube with detonator; 14) base fuze; 15) cruciform plywood panel; 16) cylinders for filling ethanol tank; 17) front ring; 18) gyroscopic devices; 19) ethanol jettison tube; 20) pipeline for feeding ethanol to the turbo-pump assembly; 21) LOX filler pipe; 22) bellows; 23) hydrogen peroxide tank; 24) thrust frame; 25) permanganate tank (the steam generator is located behind this); 26) main oxygen valve; 27) ethanol feed pipes for internal cooling; 28) ethanol jettison tube; 29) control actuators; 30) stabilizers; 31) graphite vanes; 32) air vanes; 33) combustion chamber and nozzle; 34) turbo-pump assembly; 35) control instrument bay; 36) ethanol tank; 37) LOX tank; 38) after-section housing; 39) vane ring.

Long-Range Ballistic Missiles (LRBM)

We use the term <u>ballistic</u> to describe rockets whose free-flight trajectories are the same as those of a free-falling body. In this sense the powder rockets discussed above are also ballistic; however, the name "ballistic" is applied only to large rockets with a range of the order of hundreds of kilometers.

The simplest LRBM is the German V-2 (official nomenclature — A-4), shown in Fig. 2.15.

The liquid-propellant components for this rocket are aqueous ethanol (fuel) and liquid oxygen (LOX) as the oxidizer. The calcrific value of this fuel was much higher than that of powder.

The diagram of the V-2 shows clearly the various components characteristic of LRBM's. The propulsion unit was located at the tail of the rocket. Its basic components are the combustion chamber with nozzle 33, and the turbo-pump assembly 34, to force-feed the liquid fuel components to the combustion chamber.

The turbo-pump assembly consists of two centifugal pumps connected to a turbine. The turbine is activated by the decomposition products of the hydrogen peroxide (steam + oxygen) which form in a special steam generator (not visible in the figure). The hydrogen peroxide is fed to the steam generator reactor from tank 23, and dissociates with a catalyst — sodium permanganate — fed from tank 25. These substances are expelled from the tanks by compressed air contained in cylinders 5.

The propulsion unit contains pipelines and pneumatic valves to control the operation of the engine, and also pipelines and valves to feed the fuel components to the combustion chamber and the engine cooling system. The combustion chamber and all the units of the engine assembly are attached to the thrust frame 24.

The propulsion unit is enclosed by a thick-walled reinforced shell, the after-section housing 38, which is capped by strong vane ring 39.

The thrust of the engines of LRBM's reaches tens of tons and is approximately twice the launch weight of the rockets. For the V-2, with a launch weight of 12.9 tons, the thrust at the ground is 26 tons and aloft, beyond the atmosphere, it is 30 tons.

The operation time of the engine is usually 50-200 seconds (for the V-2 — 64 seconds).

We will examine in more detail in the next chapter the engine design and its operational features.

The rocket contains two tanks for fuel components (see Fig. 2.15): tank 36 for ethanol and tank 37 for LOX.

Before launch the rocket contains about 3.5 tons of ethanol and about 5 tons of LOX. The ethanol enters the turbo-pump assembly through pipe 20 which passes through the oxygen tank.

The warhead of the rocket is in the nose and contains about one ton of explosives. The charge is activated by fuzes 11 and 14 when the rocket strikes the ground. In a research rocket the explosives are replaced by measuring, recording, and transmitting devices.

All the basic units of the rocket are connected by strong housing 8, a rigid frame with longitudinal and transverse reinforcements (the former are called stringers, the latter – ribs) covered with plate steel. The airframe of the strong housing is welded. Thrust frame 24 is attached to rear ring 6; instrument bay 35 is attached to front ring 17. The alcohol tank is suspended by special rods from the front rib; the oxygen tank is attached to the rear rib by bearing rods.

The rocket has four stabilizers 30 attached to the tail cone. The airframe of the stabilizers, as that of the strong housing, is welded, and consists of stringers and ribs covered with plate steel.

Ballistic rockets are controlled and have automatic stabilization for steady flight in a given direction. The flight direction of the V-2 is controlled by gyros in instrument bay 35. Other control-system instruments are also located there, on cruciform panel 15.

The executive units of the control system are gas and air vanes. Gas vanes 31 are located in the stream of gases emerging from the engine, and are attached to their drivers – the control actuators on the vane ring. As these vanes turn, the gas stream is partially deflected, thus causing the rocket to veer in the required direction. Since the gas vanes must operate under the most adverse temperature conditions, they are made of graphite, the most thermally stable material. Air vanes 32 play an auxiliary role. They are used only for flights in which the atmospheric layers are sufficiently dense and the rocket velocity is high enough. For flight in airless space, only the graphite vanes are used to guide the rocket.

A detailed examination of the operation of the guidance system and the stabilization of a ballistic rocket can be found in Chapter IX.

Structurally, such a rocket, despite its basic simplicity, is an extremely complex apparatus. It suffices to state that the V-2 shown in Fig. 2.15 contains over 30,000 parts, many of which must be precision-made.

A ballistic rocket is launched vertically from a launching pad, on which it stands by means of special supports.

As it gains speed, the rocket gradually turns around and its trajectory becomes inclined. At the moment the rocket attains its set velocity with a given flight direction, the engine cuts off, after which the rocket continues its flight — as an unguided missile. The maximum velocity of the V-2 was approximately 1500 m/sec, and the angle of inclination of the velocity vector to the horizon at the moment of engine cutoff was 42° * (Fig. 2.16).

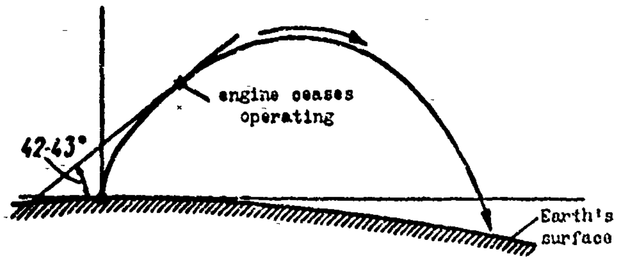


Fig. 2.16. The trajectory of the V-2 ballistic rocket.

The engine of a ballistic rocket cuts off at altitudes where atmospheric resistance no longer affects the rocket's movement. At the moment of engine cutoff the guidance system ceases to function, the rocket becomes unguided, and, depending on the random forces which may influence it at the moment of engine cutoff, there is arbitrary rotation relative to the center of mass. At it enters the relatively dense atmospheric layers the V-2, thanks to its tail stabilizers, is oriented with respect to flight direction, and by the end of the fall it again moves with its nose forward, having a velocity of 700-800 m/sec at the moment of impact.

The destructive power of a ballistic rocket does not come from the warhead alone. When it impacts, the rocket has great kinetic energy. In addition, the fuel tanks contain residue of fuel which explodes on impact. Figure 2.17 shows a crater caused by a V-2. The crater was formed because of the high velocity of the rocket at impact and because of the

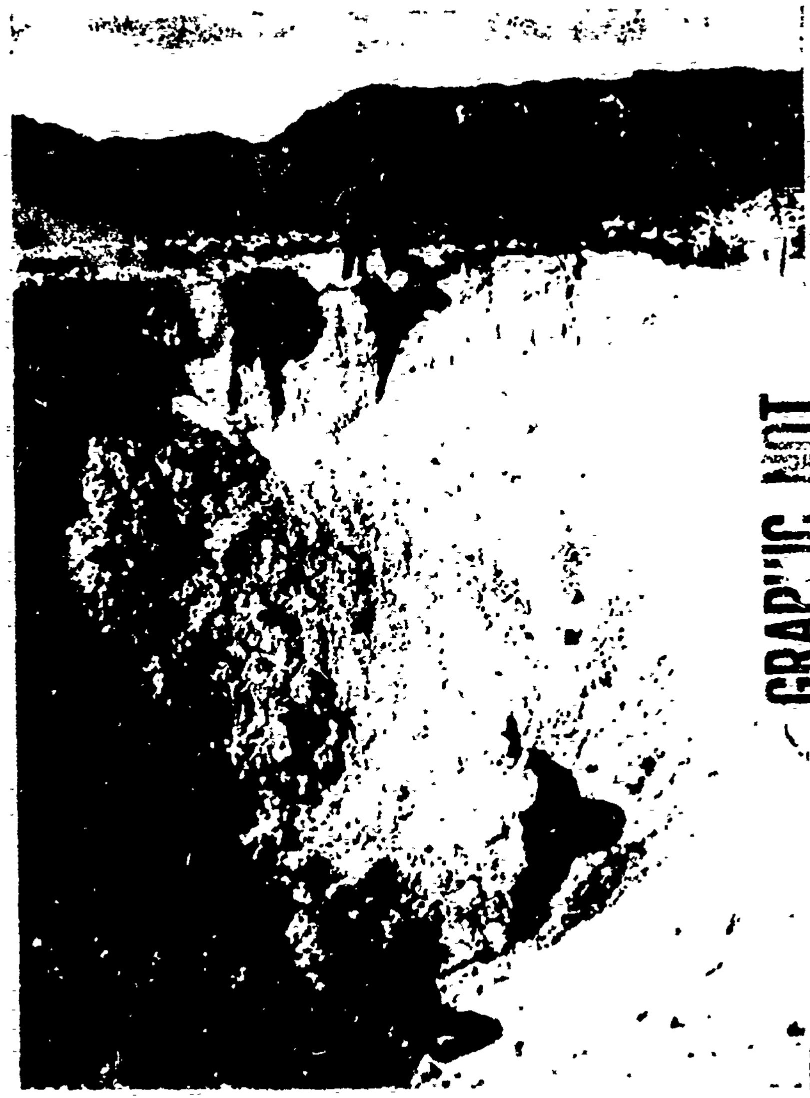


Fig. 2.17. Crater formed by a V-2.

* The program of turning the rocket, and the terminal value of the angle of inclination of the velocity vector to the horizon will be examined in more detail in Chapter VIII.

residue of fuel; the rocket contained no explosives.

It should be pointed out that an unstabilized rocket (e.g., one not having tail fins) rapidly loses speed during its fall, because of irregular rotation. When it impacts, no crater forms and the residue of fuel generally does not explode.

The V-2 rocket had a particular structural feature; the axial compression forces from the propulsion unit were transmitted to the nose of the rocket through the external strong housing. Such a design is called a mounted-tank system. In this case the fuel tanks are freely mounted within the housing and receive no inertial compression forces from the nose of the rocket.

The opposite case, an integral-tank system in which the walls of the fuel tanks are simultaneously the structural element, receiving axial inertial forces from the rocket nose, is shown in Fig. 2.18. The integral-tank system is more reasonable and makes it possible to attain better structural characteristics for the rocket, i.e., to lower the value of μ_t.

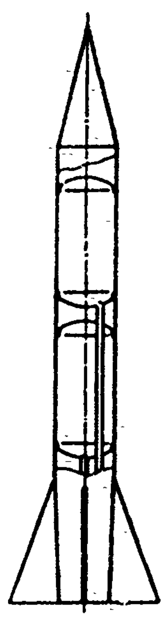


Fig. 2.18. Diagram of a rocket with integral tanks.

The structure of the rocket housing can be changed noticeably, depending on how the fuel components are to be fed to the combustion chamber (see Chapter III). In the V-2 the fuel is fed using a turbo-pump assembly. In this case the tanks are held at a slight pressure and can be made quite thin and light. If we use so-called pressure feed, where the fuel components are fed to the combustion chamber by compressed air pressure, the tanks must be made quite strong and with relatively thick walls, since they are under high pressure.

The best weight characteristics for long ranges are found in those rockets which use turbo-pump feed, with light integral tanks as the housing. However, the difficulty arises of maintaining the durability of the housing during stabilization when it enters the relatively dense atmospheric layers. Light tanks cannot withstand large transverse inertial and aerodynamic forces under conditions of strong initial heating, and a ballistic rocket will inevitably be destroyed on the descent trajectory at a great height, not reaching the target at all.

One way to avoid this difficulty is to use a detachable nose cone. In this case only the nose cone, containing the warhead, must reach the target, not the entire missile. In order that the nose cone have stable flight as it approaches the target, it has its own stabilizer.

Figure 2.19 shows a diagram of the flight of a rocket with a detachable nose cone. Rockets of this type include the Redstone, shown in Fig. 2.20, and also all types of known foreign ICBM's (Atlas, Titan, etc.).

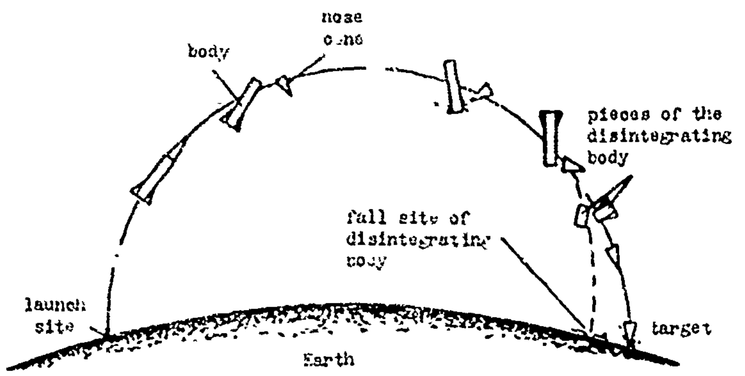


Fig. 2.19. Diagram of the flight of a rocket with a detachable nose cone.

Fig. 2.20. Redstone rocket in flight.

The described principle of a detachable nose cone makes possible a design with a high weight ratio. It has the disadvantage, however, that when it strikes the target most of the kinetic energy of the rocket mass is lost. However, when a nuclear warhead is used, this drawback becomes completely inconsequential.

If the rocket has a detachable nose cone, there is not much sense in having tail stabilizers on the rocket itself. The main purpose of tail fins on a rocket such as the V-2 is to stabilize the body as the target is approached. If the body does not reach the target, it does not need stabilizers.

From what has been said it follows, in particular, that much can be said about the operational principle of a ballistic rocket by examining the outside. For example, the Redstone ballistic rocket (Fig. 2.20) has practically no stabilizers. What tail assembly there is serves mainly to project the air rudders. Since there is no stabilizer, the nose cone must be detachable. Obviously, in this case the rocket uses integral tanks and a turbo-pump feed system. If there were pressure feed, the tanks would have to be

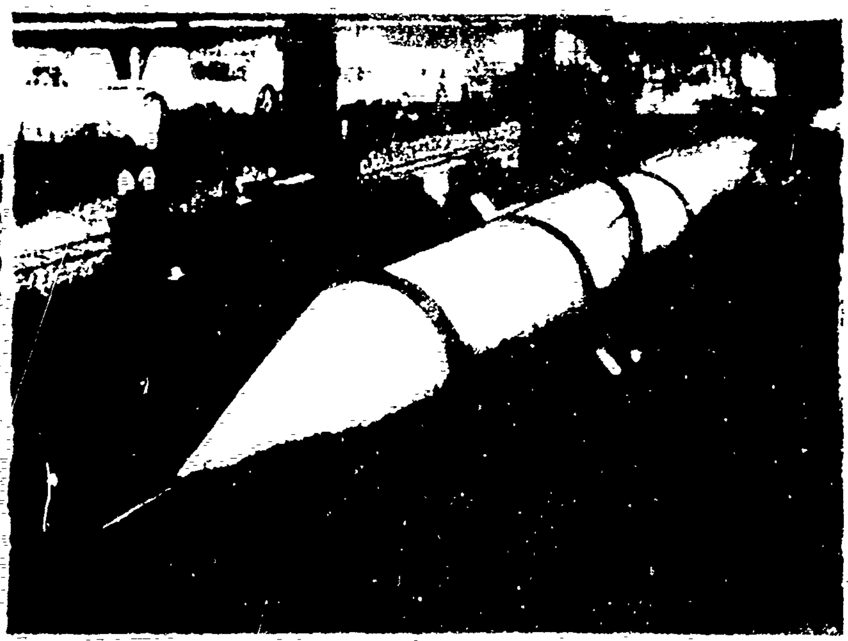


Fig. 2.21. The Redstone on the assembly line.

durable, and [illegible] need for removing the nose cone and, consequently, the rocket [illegible] stabilizers.

Available published information on the Redstone confirms this. The rocket has a detachable nose cone with its own air rudders which operate as the target is approached. The rocket's range is 800-900 km.

Fig. [illegible]. The Corporal ballistic rocket.

Figure 2.21 shows the Redstone on the assembly line.

Let us examine another American ballistic missile, the Corporal (Fig. 2.22). This rocket has a [illegible]loped tail [illegible]. Therefore, we can state, quite reliably, that the rocket has an [illegible]. Consequently, this rocket uses either a mounted-tank system with turbo-pump feed, or an integral-tank system and pressure feed. The relatively small diameter of the rocket (0.76 m) favors the second assumption. In addition, the presence of communication lines on the outside of the body is quite convincing proof. With pressure feed, the tanks are under high pressure. To avoid complicating the structure and engineering of the tanks, the cables and pipes do not enter the tank through its base but pass to the outside and are covered with thin deflectors, easily seen in Fig. 2.22. Available information on the Corporal confirms this. The rocket has a launch weight of about 5 tons and a range of 80-120 km. Figure 2.23 shows this rocket in the launch position.

Fig. [illegible] launch[illegible] position.

Long-range ballistic missiles similar to those described can attain, at present, ranges up to 3000-3500 km (e.g., the Thor). For greater ranges, multi-stage rockets with detachable nose cones must be reverted to. Modern rocketry is actually in the process of accomplishing this in the

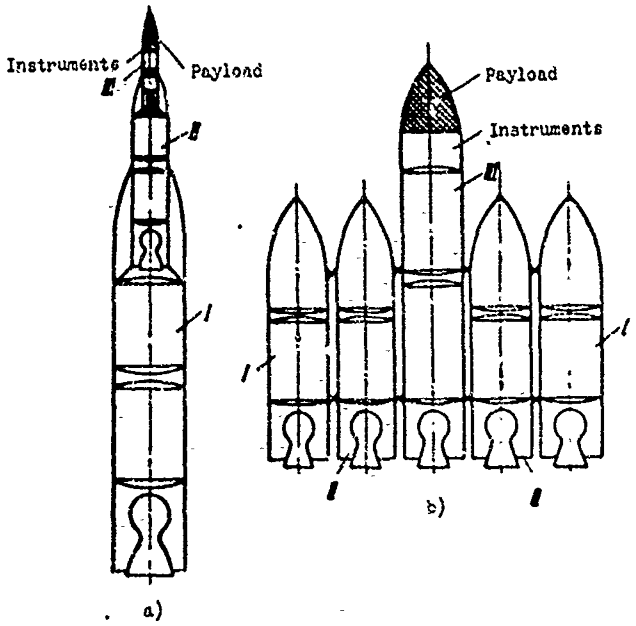


Fig. 2.24. Multi-stage rockets with transverse and longitudinal partitions. a) with transverse partition; b) with longitudinal partition. I, II, and III — the stages of the rocket.

creation of ICBM's.

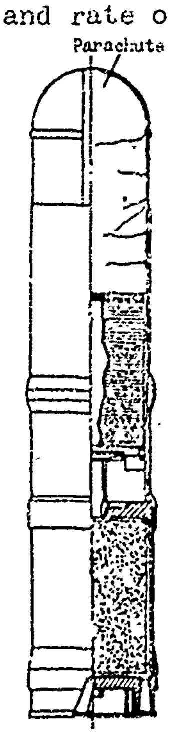
Multi-stage rockets have a great diversity of structural systems; of these let us mention those with so-called transverse and longitudinal partitions.

A transverse-partition multi-stage rocket has a structure in which all subsequent stages can be considered as the payload of the previous one (Fig. 2.24a).

Figure 2.24b shows a system with longitudinal partition. In such a system the propulsion unit of the last stages can be used for the previous stages. If there are more than two stages, there can be a system with combined longitudinal-transverse partition.

4. Antiaircraft Rockets

Antiaircraft (AA) rockets are a means of air defense. They are designed to combat flights of enemy aircraft. Compared with ordinary antiaircraft artillery, rockets are considerably more effective with respect to range of fire and rate of climb, and in addition to these, guided rockets are more accurate.



The majority of AA rockets are intended for direct destruction of enemy planes, although there are AA rockets with secondary purposes, e.g., the barrage rocket shown in Fig. 2.25. Just before the raid by enemy aircraft a group of such rockets is launched in the region of the guarded object. At the apex of the trajectory this rocket ejects a cable which drops slowly by parachute. This creates a barrage. Since the parachutes descend slowly, the barrage remains effective for several minutes and, if necessary, can be supported by the launching of additional rockets.

Direct-hit AA rockets are divided into two types: guided and unguided.

Fig. 2.25. Antiaircraft barrage rocket (85 mm).

Unguided-rocket fire is in many ways similar to AA fire. The fire accuracy of such rockets is low and is compensated by concentration of fire.

The advantage of rocket missiles over artillery shells, in the given case, is their greater range and overall firepower. The fact of the matter is that at the great altitudes of present-day planes, high-calibre shells must be used, which require more powerful weapons; this is quite difficult, considering the great number of weapons needed to guard large objects. The launching mounts for AA unguided rockets are, in all cases, simpler, lighter, and more maneuverable.

Fig. 2.26. Solid-fuel AA unguided rocket.

Figure 2.26 shows a solid-fuel version of a single-stage AA unguided rocket from World War II. Its design needs no explanation. Figure 2.27 shows the liquid-fuel version of the same rocket. The fuel components come from tanks 1 and 2 and are fed to the combustion chamber by the pressure of the gases which form during the combustion of grains 3 located behind warhead 4.

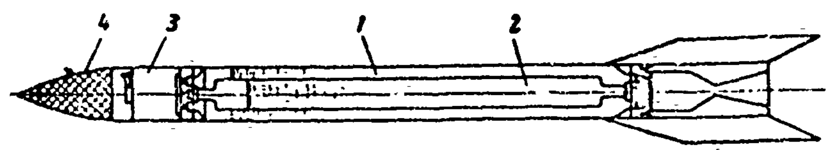


Fig. 2.27. Liquid-fuel AA unguided rocket. 1) fuel tank; 2) oxidizer tank; 3) slow-burning charge; 4) warhead.

For increased range the AA rockets can be made in two stages.

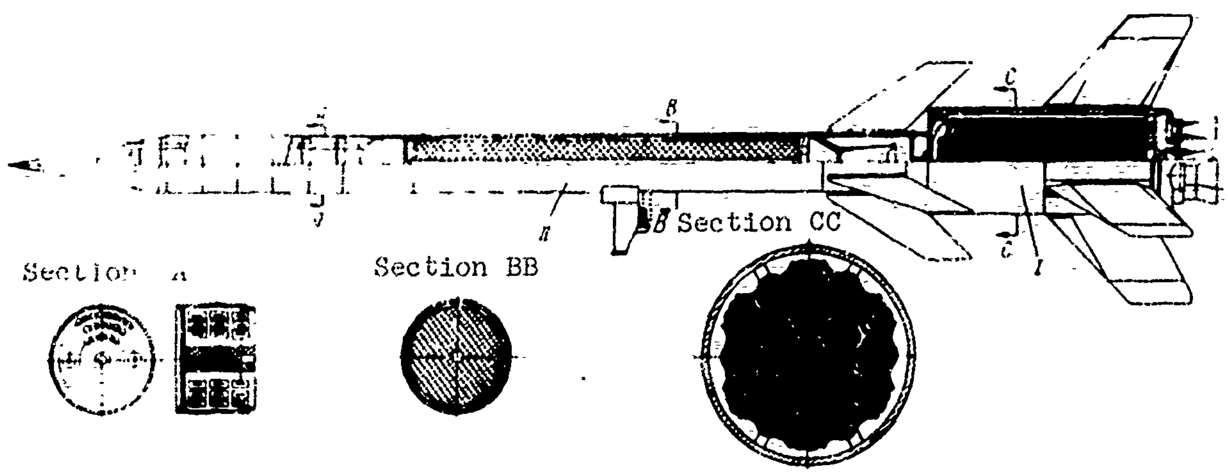


Fig. 2.28. Two-stage solid-fuel AA unguided rocket. I and II — stages of the rocket.

Figure 2.28 shows a solid-fuel two-stage AA unguided rocket. The first stage I has a moving charge in the form of several grains; the second stage II has a moving charge in the form of one grain. The warhead (section AA) contains the fragmentation-[illegible] for increased destructiveness.

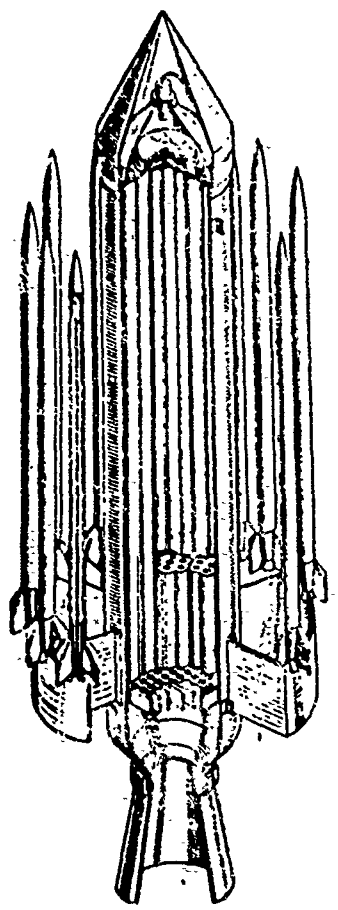


Fig. 2.29. Solid-fuel AA rocket with a group of small rockets as its second stage.

The rocket shown in Fig. 2.29 is quite unique. This is an unguided rocket-carrier; its second stage consists of several small rockets which, as they continue to move, cover a certain amount of space. This gives fire somewhat in the form of "rocket buckshot" (Fig. 2.30).

Unguided AA rockets, although they have certain advantages over ordinary AA weapons, do not provide effective air defense because of the low fire accuracy (particularly at high altitudes). In this sense, all the afore-mentioned types of unguided rockets do not satisfy the ever-increasing requirements of fire accuracy.

At present, one of the basic methods of air defense is the creation and widespread use of AA guided rockets.

Guided rockets can be divided into two types, on the basis of their target homing: ground controlled and target-seeking. Radio communication is used to send and receive commands from the ground. Target-seeking systems can consist of transmission and reception of radio signals reflected from the target, i.e., the radar principle (active homing).

The rocket can be guided to the target by the reception of the acoustic or thermal (infrared) waves emitted by the target (passive homing). Finally, we have the so-called semi-active homing, in which the receiver on the rocket receives signals reflected from the target; the source of these signals is located on the ground.

The division of guided rockets

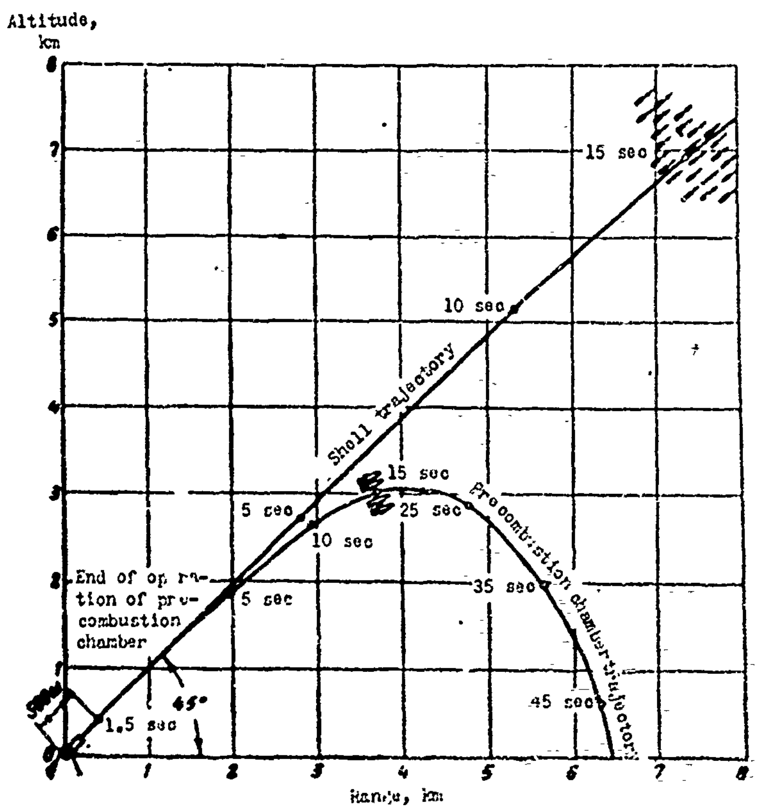


Fig. 2.30. Flight trajectory of a rocket-carrier with small rockets.

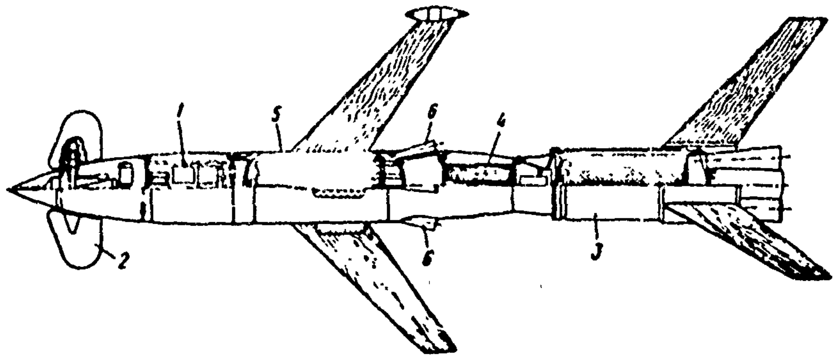


Fig. 2.31. Solid-fuel AA guided rocket. 1) guidance devices; 2) air rudders; 3) precombustion chamber; 4) warhead; 5) moving rocket charge; 6) rocket nozzles.

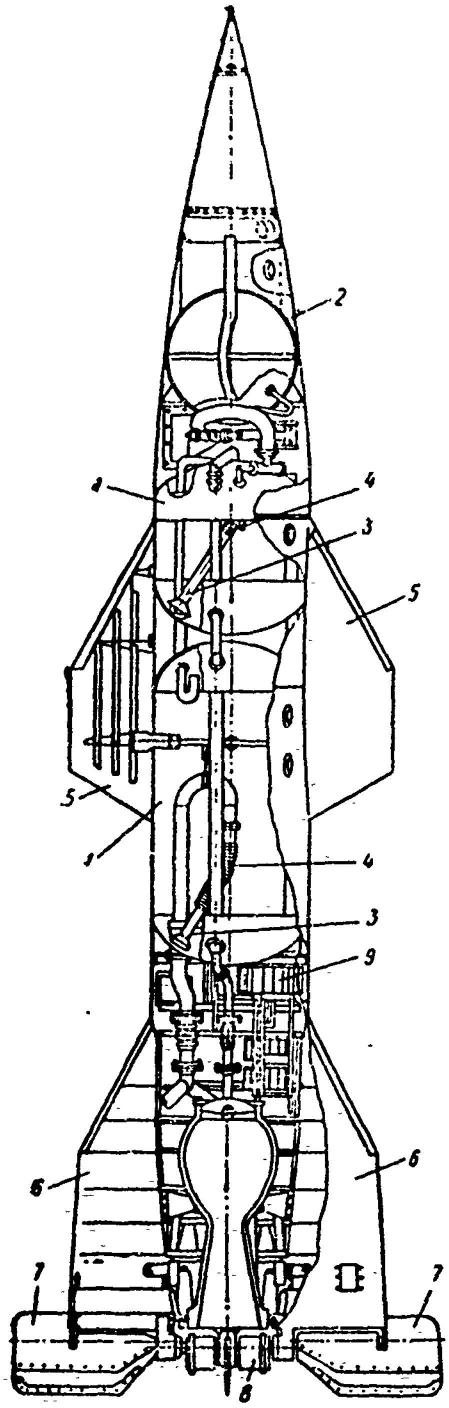


Fig. 2.32. Diagram of a liquid-fuel AA guided rocket. 1) fuel tanks; 2) spherical high-pressure container; 3) intake; 4) flexible joint; 5) wings; 6) stabilizers; 7) air vanes; 8) graphite vanes; 9) guidance instruments.

into target-seeking and ground-controlled is arbitrary, since there are possible designs which use both methods simultaneously (complex homing method). Here the rocket is ground-controlled to a certain point, and at the end of the trajectory, when the target is near and can be "noted by the rocket," the target-seeking process takes over.

This homing method is most widely used at present, mainly in long-range AA rockets.

Homing methods will be discussed in greater detail in Chapter IX.

Solid and liquid fuel can be used equally as well in AA guided rockets.

Figure 2.31 shows a ground-controlled solid-fuel rocket with a booster. This is the German Rheintochter, one of the first guided rockets.

The guidance devices 1 are located in the nose of the rocket. The rocket is controlled by four air vanes 2 located on the nose. This is the so-called "canard" aerodynamic system.

The rocket has six equi-positioned wings, at the ends of two of which there are ailerons, preventing the rocket from rolling during flight.

Four wings are attached to precombustion chamber 3. One and one-half seconds after lift-off the booster is jettisoned and the main rocket engine cuts in. It is curious that warhead 4 is located behind moving charge 5, and that the stream of gases exhausts through side nozzles 6.

During flight, the rocket is guided to target by radio. The trajectory is observed throughout the flight: with good visibility, an optical system of theodolites is used; with poor visibility, it is observed by radio.

The German Wasserfall rocket of World War II (Fig. 2.32) can serve as an example of a liquid-fuel AA guided rocket.

In this rocket, the fuel from tanks 1 is fed to the engine combustion chamber by air pressure from spherical container 2.* The fuel from the tanks enters special tracking intakes 3 suspended from flexible joints 4 which make it possible for the intakes to turn and follow the liquid during rocket maneuvers. The rocket has four wings 5, and four stabilizers 6 with air vanes 7 and graphite vanes 8.

One characteristic of this rocket, and of most AA guided rockets in general, is the symmetry of the aerodynamic shape of the rocket with respect to two planes passing through its axis. The rocket has, as a rule, four wings and four control elements. Such a system makes it very convenient to guide the rocket. As opposed to airplanes, lateral maneuvers can be made without rolling, i.e., without turning relative to the longitudinal axis; this decreases the maneuver time. The control command signals are fed to the guidance elements of each of the two planes individually, and are independent. This simplifies the control system considerably.

Because of engineering defects and asymmetry of shape, moments may occur relative to the longitudinal axis. These are counteracted either by special ailerons at the ends of the wings or by unequal movement of one of the pairs of rudders.

A study of the presently developed AA guided rockets used by various countries indicates a trend toward the use of two-stage rockets. An ordinary solid-fuel engine is used as the first (booster) stage. The second stage has, as a rule, a liquid-fuel rocket engine; in certain designs this may be a ramjet engine. For a range of the order of 25-35 km and altitudes of 3-18 km the AA guided rockets have a launch weight of the order of 1000-1500 kg, with a warhead of 100-150 kg.

Figure 2.33 shows the Nike rocket; this has a solid-fuel booster and a second-stage liquid-fuel rocket engine. The rocket is 10.7 m long overall, with a total launch weight of 1100-1200 kg (weight of second stage – 680 kg). At engine cutoff

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* The structure of the engine of this rocket will be examined in Chapter III.

Fig. 2.33 The Nike on a launcher.

the rocket has a velocity of about 700 m/sec. The maximum destructive target altitude is 18 km with a slant range of 30–40 km.

The Nike is launched from guides and flies practically vertically up to the moment the first stage is jettisoned (Fig. 2.34), and then it is guided to the target by a radar beam (see Chapter IX).

The shortcomings of this missile are its relatively short range, the poor interference-killing ability, and the complexity of the guidance system.

Figure 2.35 shows a battery of Terrier AA rockets on the deck of a warship. The rockets have solid-fuel boosters. Figure 2.36 shows a photograph of the takeoff of the French Larea AA guided rocket, which has four solid-fuel boosters.

The Swiss Oerlikon (Fig. 2.37) is an example of a single-stage AA guided rocket.

Attempts to increase the effective range of AA guided rockets lead to an increase in the absolute dimensions and weight of the rocket. For example, the Nike-Hercules (Fig. 2.38) has a launch weight of 4.5 tons and a range of 80 km. This rocket has a radar homing-system, with a ground-controlled initial flight stage.

Of the "surface-to-air" guided missiles, particular attention has

Fig. [illegible] Take-off of an AA guided rocket, [illegible] the launch position.

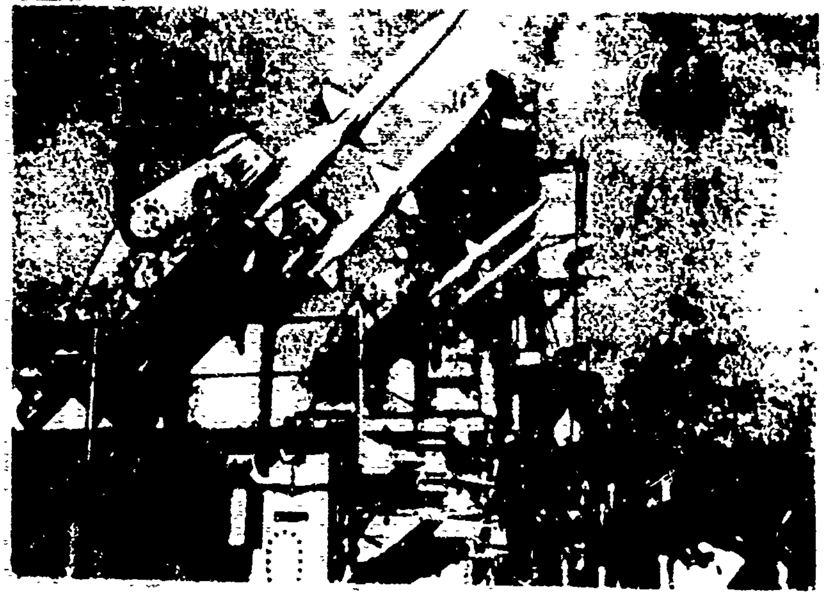

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Fig. 2.35. Shipboard AA rocket installation.

Fig. 2.36. Take-off of an AA rocket using solid-fuel boosters.

Fig. 2.37. Oerlikon guided rockets.

Fig. 2.38. The Nike-Hercules AA rocket.

been devoted by the foreign press to missile-interceptor rockets, the so-called antimissile missiles, designed to combat enemy ballistic missiles. Such AA rockets are very difficult to create, and at present are in the initial development stage.

An antimissile missile should be a missile with a very high engine thrust, so that it will have an effective rate of climb. It is proposed that the antimissile missile be equipped with a nuclear warhead. Possibly, the antimissile missiles will be launched in groups and intercept a ballistic rocket at about 80 km from the defended object. The temperature of the nuclear explosion is considered to be the decisive factor in the destruction of the warhead of an ICBM.

The warhead of an ordinary AA rocket explodes at a distance from the target to insure the maximum fragmentation effect. Since the fragments, in addition to scattering velocity, still have forward flight velocity of the rocket itself, the explosion should obviously occur just short of the target. The time of the explosion of an AA guided missile is determined, as a rule, by the setting of a proximity fuze which operates on the radar principle. For safe handling, the proximity fuze usually has several arming stages: after launch, after some time has elapsed, or after a definite velocity has been attained. In the event of a miss, or if the fuze malfunctions, the abort command is given automatically, to destroy the rocket in midair. This eliminates the danger of the rocket's landing on its own launch positions and defended objects. Certain designs (particularly practice rockets) have a parachute system to retrieve the rocket so that it can be reused.

Certain present-day long-range (more than 50 km) AA guided rockets have nuclear warheads for maximum effectiveness in combatting groups of aircraft.

5. Other Types of Rocket Vehicles

Air-Launched Rockets

Air-launched rockets can be used for firing on ground and air targets.

"Air-to-surface" and "air-to-water" rockets are similar to aerial bombs and are intended mainly for destroying targets by direct hits. They can be either guided or unguided.

Rocket engines are used to increase the flight speed of this type of vehicle. Additional speed is needed, on the one hand, to increase the penetrating force when firing on a target such as a heavy ship. On the other hand, if the missile is guided, the extra velocity is also needed to increase the control efficiency and the effective range, in order that the airplane be able to destroy the target without approaching too close.

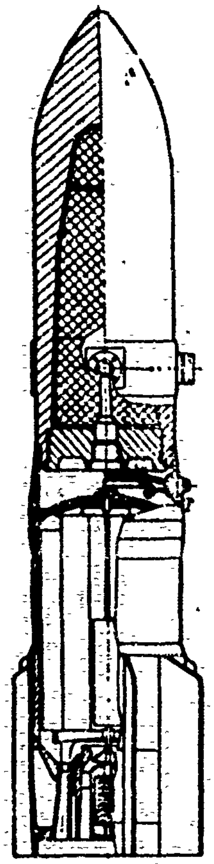

Fig. 2.39. Longitudinal section of an armor-piercing aerial bomb with solid-fuel rocket booster.

Figure 2.39 shows the longitudinal section of a 1000-kg armor-piercing aerial bomb with a solid-fuel rocket booster. It can be seen that the relatively small rocket charge cannot assure a great flight range for such a heavy bomb; it is merely the booster which increases its piercing power.

The additional velocity which such aerial bombs receive from a rocket charge is, on the average, 100 m/sec.

A feature of the solid-fuel engine in this design is the nozzle pressure-regulator which helps decrease the spread of the ballistic characteristics when the powder charge has various temperatures (see Chapter IV for powder characteristics).

Typical "air-to-surface" and "air-to-water" vehicles are guided aerial bombs and aerial torpedoes, designed to destroy warships and large ground installations. Here, as for the AA rockets, there may be either self-homing or target-homing from a bomber.

Figure 2.40 shows one of the guided missiles (torpedoes) of World War II. This missile has a rocket engine 1 which operates on hydrogen peroxide. The warhead 2 is in the nose of the vehicle. The control instruments are in the rear of the body 3. The missile has ailerons 4, elevators 5, and a rudder 6. The figure also shows the flare 7, a rear beacon which makes the torpedo visible at night to the tracker on board the bomber.

Fig. 2.40. Aerial rocket torpedo. 1) engine; 2) warhead; 3) rear of the body containing the control instruments; 4) ailerons; 5) elevators; 6) rudder; 7) flare.

Air-to-air missiles are also categorized as guided and unguided. The unguided AAM's can be further subdivide into single-fire or salvo-fire missiles.

Air-launched salvo-fire missiles

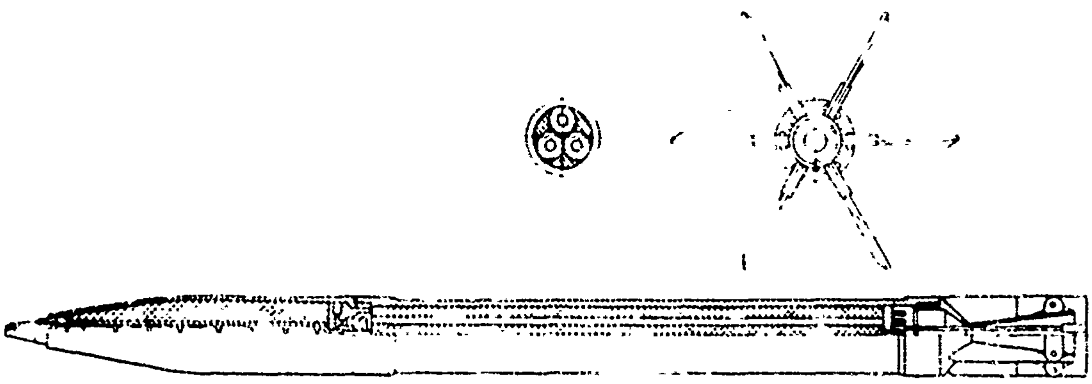

Fig. 2.41. Air-launched missile with unfolding fin assembly.

Fig. 2.42. Salvo of rockets from launchers on the ends of the plane's wings.

Fig. 2.43. Rocket salvo from a plane. At the right can be seen the fired rockets.

are principally identical to short-range ground artillery shells.

Figure 2.41 shows the structure of a direct-hit air-launched missile with unfolding fin assembly. While the rocket is in the launching tube the fin assembly is folded; it unfolds as the rocket leaves the tube.

Figures 2.42 and 2.43 show a rocket salvo from jet planes.

The use of even such relatively simple rockets noticeably increases the firepower of airplanes.

It is impossible to install large-calibre artillery weapons on a plane because of the great weight of these weapons, and the recoil when fired. The launching devices for rocket fire are extremely simple and light, and there is no recoil when the rockets are fired.

Figure 2.44 shows an air-launched unguided fragmentation missile for firing on bombers from a fighter. When the warhead explodes at a certain distance from the target, the fragmentation-incendiary elements scatter forward in the shape of a cone with an apex angle of about 30°.

The present state of the development of air-launched armament of foreign armies is characterized by the transition to plane-to-plane guided rockets (Fig. 2.45).

The launch weight of such rockets is [illegible]0-200 kg. Such rockets have solid-fuel engines.

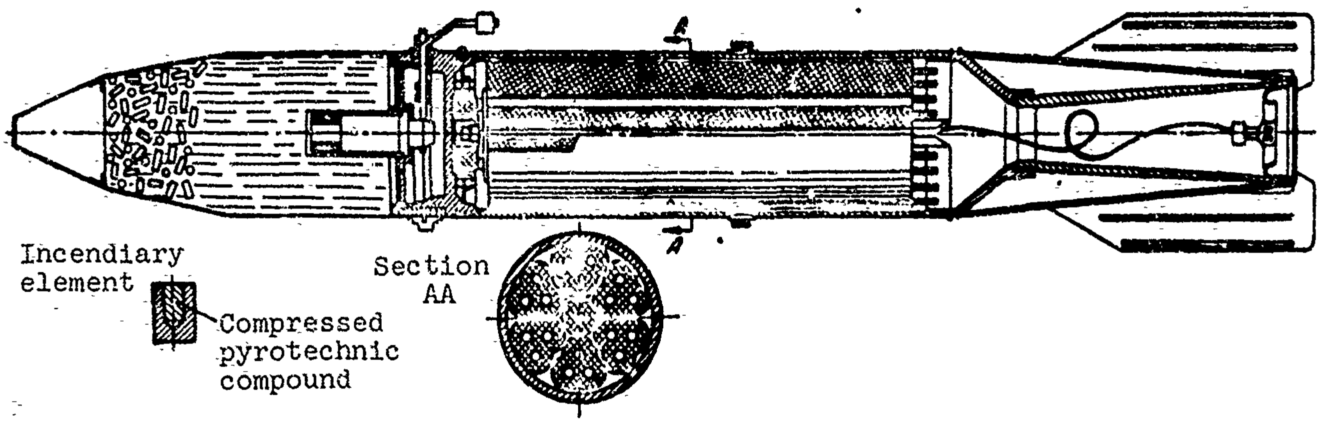

Fig. 2.44. Air-launched fragmentation missile.

Fig. 2.45. Air-launched guided rocket.

Naval Rockets

Above we have examined ground rocket-artillery missiles and air-launched rockets. There also exist naval rockets.

The specific feature of these vehicles is the transportation condition. They must be able to be stored compactly on board ship. They must contain fuel which can be stored for long periods of time. The launching devices must also have certain features.

The firing on above-water targets does not differ much from firing on ground targets. True, there are possible exceptions.

Figure 2.46 shows a spherical air-launched skip rocket. Such a missile, equipped with a rocket engine, is ejected from a plane at a low height toward a ship, falling short. After the fuel has burned out, the moving charge separates from the warhead. Then the spherical charge, equipped with contact and depth fuzes, ricochets (skips) along the surface of the water, making up to 12 jumps, and, if dropped correctly, destroys the ship.

Rockets can also be used for underwater firing. Figure 2.47 shows an underwater missile for submarine firing on the lower parts of ships. An interesting feature of this missile is the gas shroud (pocket) on the surface of the missile, to reduce

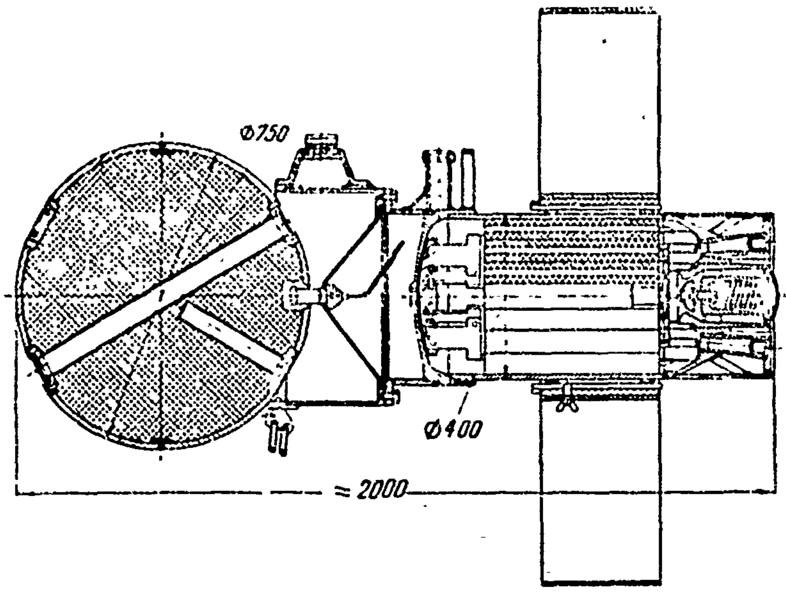

Fig. 2.46. Above-water skip rocket.

friction. This shroud results from the ejection of a small amount of the solid-fuel gases through a frontal aperture; these gases then gradually envelope the missile.

Meteorological and Geophysical Rockets

To conclude our survey of the existing types of rockets, let us examine the so-called meteorological and geophysical rockets.

The value of such rockets to contemporary science hardly needs to be mentioned. Rockets have made it possible to conduct direct physical studies of the entire atmosphere of the Earth, and near cosmic space. Through the use of rockets we have obtained completely new data on a great many meteorological, geophysical, and astrophysical problems when investigating the Earth's atmosphere and near cosmic space.

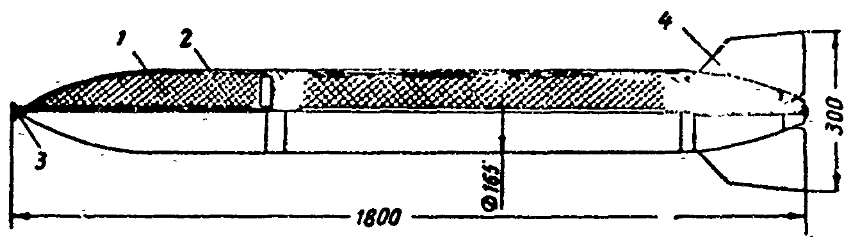

Fig. 2.47. Underwater rocket projectile. 1) warhead; 2) tube for forward ejection of gases to decrease body friction in the water; 3) aperture for emitting solid-fuel gases; 4) main nozzles.

The heights attained by present-day geophysical rockets could not be reached by any other means.

The ascent of experimental animals in geophysical rockets, and their subsequent recovery, can be considered the first step in solving the problem of manned space flight.

The Soviet Union has done much in the creation of special geophysical rockets. These rockets have made possible active Soviet participation in the International Geophysical Year; flights made using these rockets, including the record flight of a single-stage rocket to an altitude of about 500 km, have yielded valuable scientific results.

Geophysical rockets differ considerably from long-range military rockets, not only in the features of the installation and location of the measuring and recording scientific-research apparatus, but also in the vast number of scientific requirements placed on such rockets. Here we should first mention the system for recovering individual groups of equipment, jettisoned from the rocket at specific times according to the scientific program, the stabilization system for free flight after engine cutoff, etc.

The Soviet Union has created several highly-effective specialized guided geophysical rockets, and has instituted a vast program of scientific investigations of the upper atmosphere, outer space, and the physical conditions of rocket flight.

May 1949 saw the launching of the first vertical rocket to an altitude of 110 km; the scientific apparatus weighed 120-130 kg. The research program has been continually expanded since that time. In May, 1957, a rocket was launched to an altitude of 212 km (Fig. 2.48). The scientific apparatus and experimental animals (dogs) weighed 2200 kg; these were later successfully recovered. In February, 1958, a rocket containing 1520 kg of scientific apparatus was launched to an altitude of 473 km (Fig. 2.49), setting a world altitude record for rockets of this class.

In August 1958 a Soviet single-stage geophysical rocket reached a computed altitude of 450 km; it contained two dogs, in addition to the instruments for complex upper-atmosphere studies. The geophysical apparatus, radio-telemetry devices, power supplies, hermetically-sealed cabin with the experimental animals and auxiliary systems, together with the instrument compartment, weighed 1690 kg. The experimental animals showed no ill effects upon landing.

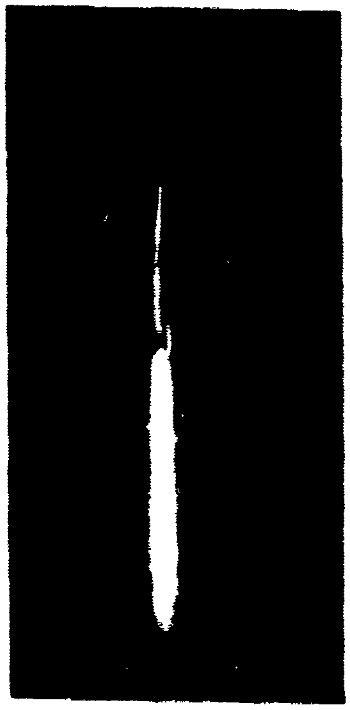

Fig. 2.48. Flight of a Soviet geophysical rocket to an altitude of 212 km. The projecting instrument containers can be seen on the side of the rocket.

Fig. 2.49. Soviet geophysical rocket launched on February 21, 1958.

In recent years, many specialized versions of meteorological and geophysical rockets have been created by foreign countries. The first of these were the small

Fig. 2.50. The Wac-Corporal meteorological rocket.

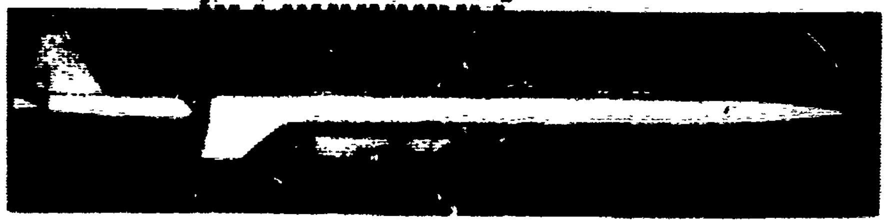

Fig. 2.51. The Aerobee meteorological rocket.

Fig. 2.52. The Véronique meteorological rocket.

unguided meteorological rockets, such as the Wac-Corporal (Fig. 2.50) and its sister rocket, the Aerobee (Fig. 2.51). These are liquid-fuel rockets; the fuel is fed to the combustion chamber by compressed gas pressure. The Aerobee is a two-stage rocket; a solid-fuel engine is used in the first stage. The Wac-Corporal and the Aerobee are unguided and launched vertically from launching rails (see Fig. 2.50). The maximum attainable altitude of these rockets is 80-120 km, depending on the weight of the payload (100-70 kg). The French meteorological rocket Véronique (Fig. 2.52) has approximately the same characteristics. The photograph shows a cutaway of this rocket; we can see the engine with the fuel tanks, and the gas-generator pressure-feed system for feeding the components to the combustion chamber. The detachable nose cone of the rocket contains the research apparatus and the parachute recovery system. The fact that the nose cone is detachable simplifies the problem of recovering the scientific equipment. The rocket is launched from a launching platform, not rails. During the first part of the flight the rocket is stabilized by special cables attached to it. The cables are paid out from a drum on the ground. Because of this, the rocket appears to be on "reins." At a specific time after the launch, when a velocity sufficient for stable flight has been attained, the rocket is cut loose from the cables.

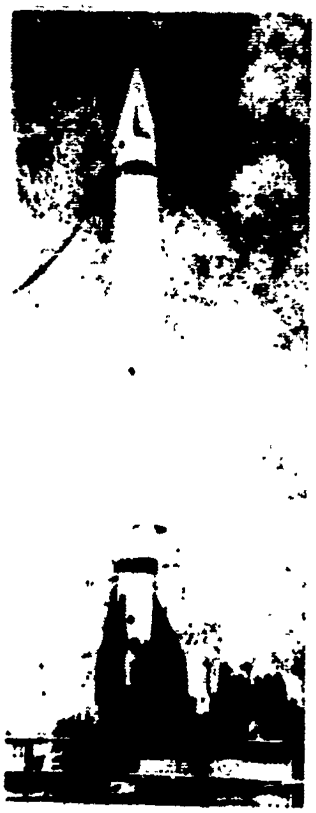

Fig. 2.53. The [illegible]king meteorological rocket.

Figure 2.53 shows the Viking, an example of a guided meteorological rocket. Rockets of this type have attained altitudes of about 250 km with a payload of the order of 400 kg.

The propulsion unit of the Viking has a turbo-pump feed system, and integral low-pressure tanks. A structural feature of this rocket is that it is guided not by graphite vanes but by the turning of the combustion chamber in two planes. The angular orientation of the rocket during roll is restored by means of special nozzles which use part of the exhaust gases from the turbo-pump unit.

Meteorological rockets may also be of multi-stage construction. Figure 2.54 shows a two-stage bumper meteorological rocket in flight. A V-2 is used as the first stage, a Wac-Corporal as the second. This rocket attained altitudes somewhat above 400 km. The payload, however, weighed only several tens of kilograms.

Observational data obtained by meteorological rockets are recorded by instruments during the flight and usually transmitted by radio to the ground. In this case the same type of radio-telemetry systems are used as in the various military rockets (see Chapter X).

The scientific apparatus recovered from the launching of meteorological rockets is very valuable. This apparatus includes the results of investigations, recorded on film, spectrograms, and photographs (see, e.g., the photograph of the Earth's surface in Fig. 2.55), all kinds of samples of high-atmosphere air, etc. In this case, measures must be taken to decrease the rate of fall of either the rocket or the container with the instruments, i.e., a recovery system must be organized.

The simplest recovery system for unguided rockets, which only partly solves the problem, is to jettison the tail assembly of the rocket when it reaches maximum altitude. Then, when it returns to the Earth, the rocket will be unstabilized and, because of its irregular roll, its rate of fall will sharply decrease. In this case, when small rockets are launched it is possible to retain some of the instruments and records.

Geophysical rockets are designed, as a rule, with detachable nose cones containing the equipment to be recovered. Special braking units are provided in the parts to be recovered. For example, the braking unit of the nose cone of the Véronique consists of four discs connected by cables. Braking flaps can also be used to decrease the rate of fall; these unfold in the tail section of the nose cone after separation from the body of the rocket (Figs. 2.56 and 2.57).

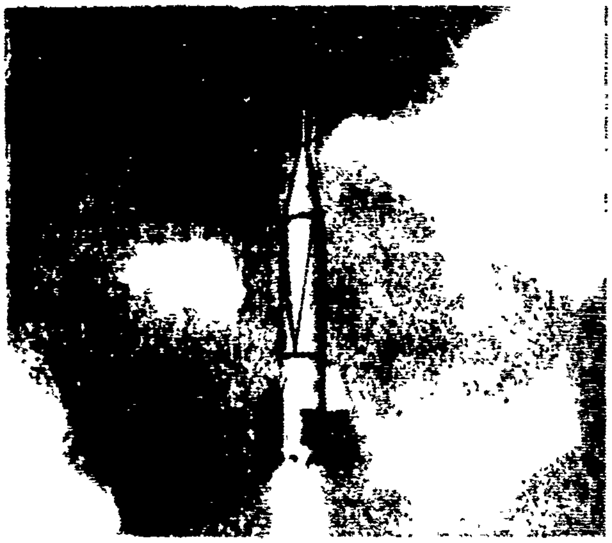

Fig. 2.54. [illegible]

[illegible]. 2.55. Photograph of the Earth's sur- [illegible] from an altitude of 255 km.

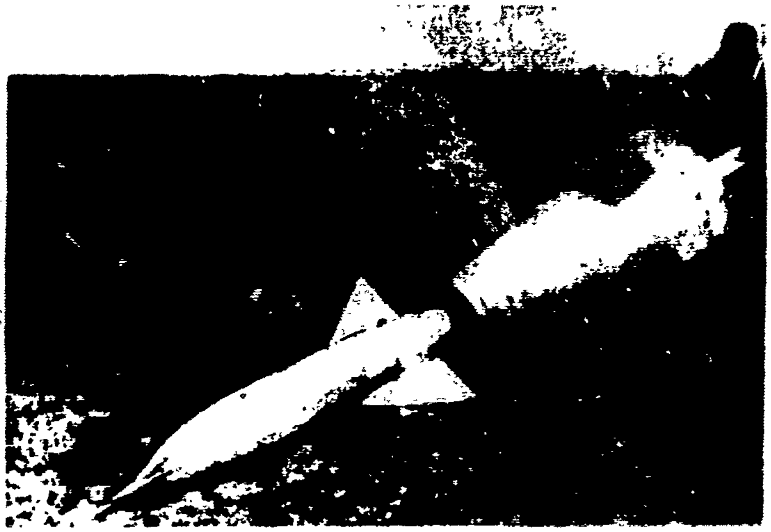

Fig. 2.5[illegible]. N[illegible] container of a Sovi[illegible] physical rocke[illegible] at the moment the [illegible] mental animals were released.

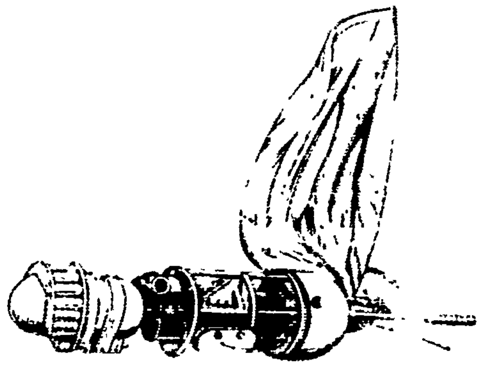

Fig. 2.58. The nose section of a Pyorrki rocket, equipped with a recovery system for prolonged buoyancy.

Fig. [illegible]

Fig. [illegible]. Frame from a movie film taken during the flight of a rocket, showin[illegible] the be- [illegible]

GRAPHIC NOT REPRODUCIBLE

Fig. 2.60. Dog in a space suit.

The nose section, falling from a great altitude, first acquires considerable velocity, and is then braked in the atmosphere by flaps, so that the rate of fall near the ground is about 100 m/sec. A parachute can be opened at such a velocity (this is not shown in Figs. 2.56 and 2.57).

If the nose cone lands in the water, after braking a recovery system comes into play (instead of the parachute); this assures that the nose cone or container with the instruments will float for quite some time.

Figure 2.58 shows the nose section of the Pyorrki rocket, equipped with a collapsible air tank, which fills with air immediately upon impact with the water. The air is contained in a metal cylinder directly ahead of the containers with the apparatus.

When the nose section lands on the ground, the parachute system brakes the container to a velocity of several meters per second. A thin-walled nose tip, which crumples on impact (see Fig. 2.57) can be used for a soft landing.

The described combined recovery system makes it possible to recover even the most delicate instruments. In addition, it makes it possible to conduct experiments on the activity of organisms at great altitudes.

The nose section shown in Fig. 2.56 dropped from a height of 100 km (1950). Two dogs were launched to this height. During the entire flight the activity of the organism of the animals was recorded by instruments. In addition, their behavior was recorded on movie film. Figure 2.59 shows one of the frames from the film taken during the rocket ascent.

The nose section shown in Fig. 2.57 was successfully recovered from a height of more than 200 km (1957). The container also held two dogs. Because of the high rate of fall, the nose section shows quite clearly the thermal effects of braking in the atmosphere. The photograph shows that the flaps were greatly heated.

Complete development of a recovery system is a necessary condition for manned rocket flight to great altitudes. In particular, an emergency recovery system must be provided, so that the person can escape from the rocket at great altitudes at any time during the flight. Such investigations have been going on for many years in the Soviet Union. The first experiments were conducted using dogs; wearing special light space suits they were ejected from rockets at various altitudes and reached the Earth safely. Figure 2.[illegible] shows a dog in a space suit before being launched to an altitude of 100 km (1[illegible]).

The natural continuation of the studies made using geophysical rockets is the launching of artificial earth satellites.

6. Artificial Earth Satellites and the Mastery of Interplanetary Space

The General State of the Problem

At present, perhaps no other technical problem has so captured the imagination of mankind as that of the mastery of interplanetary space.

Much literature is presently devoted to this problem. However, the majority of this literature consists of only the general principles of rocket flight and rocketry in general as applicable to interplanetary flights. These works contain many speculations on the hypothetical conditions of manned rocket flight. They discuss the presence of such dangers as encountering meteorites and the fatal effect of cosmic rays on the human organism. They even give ways of feeding a man under conditions of weightlessness. From time to time, descriptions are published of the so-called "projects" for reaching the Moon, Mars, and Venus using rockets guided by "special instruments," etc. Such questions are usually raised by lone-wolf-enthusiasts, who usually attempt to solve them theoretically, with complete disregard for the actual design of rockets, and therefore most of their works are worthless.

Considerably less often in the literature does one find works which treat specific technical problems, whose solution actually results in the vast sea of difficulties which separates the dream of manned space flight from its actual realization.

Rocketry has long since emerged from the stage of preliminary quests in which it formerly found itself. The design and creation of the present-day long-range rocket, not to mention the space rocket, is possible only by large organized groups, with many thousands of members, armed with modern science, laboratories, and computer technology, and having a serious experimental-productive background.

At present, operations on the creation of intercontinental and space rockets, and the carrier rockets for artificial earth satellites, have become so broad in scope that the lone-wolf-enthusiast who attempts to independently pose and solve the problems which to him are the most important and interesting generally finds himself in an unproductive position, at best repeating what has been done long before by the various groups.

A typical problem published from time to time in the works of individual authors is that of determining the so-called "optimum" flight conditions. For example, they discuss the question of finding the "most favorable" flight trajectory from the Earth to the Moon, the "most favorable" weight ratios between the n-th and the (n + 1)-th stages of a rocket, etc.

Such works, however, do not take into account the actual design of the rocket itself. Meanwhile, the solution of such problems actually begins with the preliminary selection of not one or two, but many interdependent design parameters of the rocket, mainly associated with its weight characteristics and the energy-producing possibilities of the propulsion unit. Variation of these parameters makes it possible to determine their most advisable ratios, allowing the creation of a design which can actually be produced. The selection of the rocket parameters involves a vast number of questions associated with the establishment of the most suitable trajectory for placing it in orbit, and the optimum parameters of the orbit itself. All these questions can be answered by considering the mission of the projected rocket. In this case, we must consider not only the flight, or ballistic, characteristics of the rocket, but many other factors as well, e.g., storage, transportation, the organizing of launches, etc. The economic side of the picture also plays an important role.

After preliminary selection of the parameters when planning and designing a rocket, all computations are repeatedly refined in accordance with the specific design.

Naturally, the basic computational operations in the planning of a rocket are carried out by high-speed electronic computers, since the use of ordinary computation methods would require a span of time which would considerably exceed the lifetime of a generation of people.

As we can see, in planning a rocket, practical demands at present immeasurably exceed the capabilities of one man. Naturally, this does not eliminate the theoretical value of the works of individual authors, but merely places them in a subordinate position.

The Orbit of an Artificial Satellite

The basic force which determines the movement of a satellite is the force of gravity. The laws of motion of a satellite, just as the laws of motion of any celestial body, are defined by celestial mechanics, with Newton's law of universal gravitation as its base.

The simplest orbit for an artificial satellite will be a circular one, where the satellite moves with constant velocity at a constant distance from the center of the Earth. It is easy to calculate the velocity which must be imparted to the satellite in order that it attain such an orbit. This is the so-called first cosmic, or circular, velocity. In moving in a circular orbit, gravity should be counterbalanced by centrifugal force (Fig. 2.61):

$$M_{sat} g_r = M_{sat} v_{cir}^2 / r, \qquad (2.1)$$

where r is the radius of the circular orbit: $r = R + h$;

h is the height of the orbit above the Earth's surface;

R is the mean radius of the Earth: $R = 6371$ km;

g_r is the acceleration of gravity at distance r from the center of the Earth;

M_{sat} is the mass of the satellite;

v_{cir} is the circular velocity.

Let us establish a relationship between the acceleration of gravity and distance r. According to the law of universal gravitation, the force of attraction of the satellite by the Earth:

$$G = f(M_E M_{sat}/r^2),$$

where f is the gravitational constant;

M_E is the mass of the Earth.

Considering that $G = M_{sat} g_r$, we get

$$g_r = fM_E/r^2.$$

The numerator of this expression is the constant of the gravitational field. Let us designate this value by k^2. For the Earth's gravitational field,

$$k^2 = fM_E = 398{,}620 \text{ km}^3/\text{sec}^2.$$

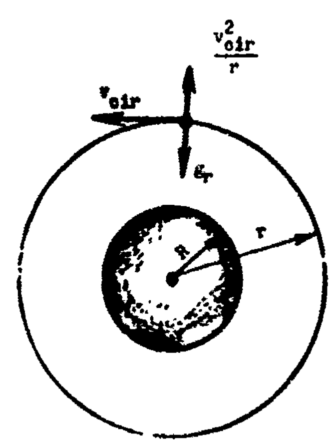

Fig. 2.61. Circular orbit of a satellite.

Thus,

$$g_r = k^2/r^2. \tag{2.2}$$

On the basis of relationships (2.1) and (2.2) we get

$$v_{cir} = \sqrt{g_r r} = k/\sqrt{r} . \tag{2.3}$$

Data on the density of the Earth's atmosphere, obtained from the launching of geophysical rockets and artificial satellites, have shown that a height of the order of 300 km for a circular orbit is sufficient so that as a satellite moves it does not undergo noticeable aerodynamic drag and could exist for several weeks. Formula (2.3) for a height of 300 km gives a value of 7730 km/sec for circular velocity v_{cir}.

If at the moment the satellite is inserted into orbit the rocket has a velocity greater than circular (for the given insertion height), the orbit will be not circular but elliptical, and the <u>maximum</u> distance of the orbit from the center of the Earth (the so-called <u>apogee</u>) will always be higher than the insertion point (Fig. 2.62). This has happened in the case of all artificial earth satellites created to the present time.

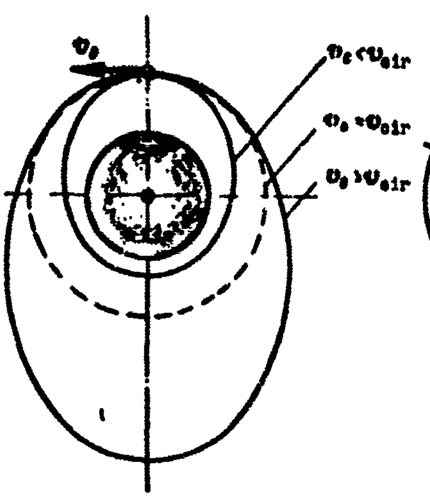

Fig. 2.62. Change in the orbit of a satellite vs. the insertion velocity.

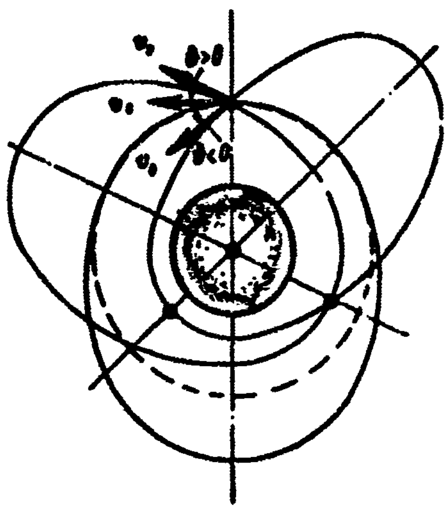

Fig. 2.63. Change in the orbit of a satellite vs. direction of insertion velocity vector.

If the terminal velocity of the rocket is less than circular, the <u>minimum</u> distance of the elliptical orbit from the center of the Earth (the so-called <u>perigee</u>) will always be less than the insertion height (Fig. 2.62). With low release heights this will cause the satellite to enter the dense layers of the atmosphere, brake, and be destroyed.

This means that for minimum heights the velocity with which the satellite is inserted into orbit can never be less than circular, corresponding to the insertion height.

It is important to note that for a given insertion height and velocity, a deviation of the angle of inclination of the velocity vector to the local horizon at the insertion point ϑ to either side of the zero value will cause the perigee of the orbit to decrease (Fig. 2.63). Therefore, for a low perigee the satellite is best inserted into orbit with a zero angle of inclination of the velocity vector, where the insertion point coincides with the perigee of the orbit.

The time of one revolution of the satellite around the Earth, i.e., its orbital period, for a circular orbit of radius r, is defined as

$$T_{sat.cir} = 2\pi r/v_{cir} = (2\pi/k)r^{3/2}. \qquad (2.4)$$

With a satellite flying at 300 km, its orbital period is approximately 90 minutes. As the height increases, the period also increases. At a height of 35,830 km the orbital period of a satellite is 23 hours 56 minutes, i.e., one sidereal day. Such a satellite, launched in the equatorial plane in an easterly direction (with respect to an observer on the ground), would appear to be suspended in space, without moving.

For elliptical orbits, the orbital period is determined by a relationship which is completely analogous to (2.4); instead of r we must substitute a, the semimajor axis of the ellipse (see Chapter VIII):

$$T_{sat} = (2\pi/k)a^{3/2}; \qquad (2.5)$$

here, as we know, the semimajor axis of an ellipse

$$a = (r_\pi + r_\alpha)/2,$$

where r_π is the radius of the perigee;

r_α is the radius of the apogee.

For example, the first Soviet artificial earth satellite, launched on October 4, 1957, was inserted into orbit with a perigee of 228 km and an apogee of 947 km. Its orbital period at the outset was 96.2 min, which corresponds to the result given by using relationship (2.5).

Although the density of the Earth's atmosphere at several hundred kilometers is quite insignificant (at 300 km the mean free path of molecules is of the order of 70-150 m), the drag of even such a rarefied atmosphere affects the movement of the satellite, with enormous velocity and considerable flight time. Aerodynamic braking results in an energy loss, which leads to a gradual decrease in the mean radius of the orbit (see Chapter VIII). Expression (2.5) shows that in this case the orbital period must unavoidably decrease, i.e., the satellite begins to orbit the Earth more rapidly.

Such a phenomenon, which at first glance appears paradoxical, can be graphically illustrated by the first and third Soviet satellites, separated from the body of a carrier rocket. The rocket was decelerated more rapidly, quickly lost altitude, and overtook the satellite.

It should be pointed out that with drag present, the mean radius of the orbit decreases mainly due to a decrease in the apogee, as a result of which the elliptical orbit becomes increasingly circular. As the height of the circular orbit decreases, the satellite velocity will increase, as follows from (2.3).

Braking progressively increases as the height of the orbit decreases. At about 100 km the aerodynamic drag is so great that a further increase in velocity is impossible. The velocity becomes less than circular and the satellite begins to drop rapidly. Intense heating occurs, and the satellite or carrier rocket is destroyed; they can be likened to artificial meteors.

The lifetime of a satellite depends on the degree to which it is braked in the atmosphere. Clearly, the further the orbit is from the Earth, i.e., with an increase in the perigee and apogee, the greater will be the lifetime. In addition, the degree of braking due to aerodynamic drag, as we shall see later,* will depend on the shape of the flying object and on its weight per unit area cross section perpendicular to the velocity vector, the so-called transverse load.

The lifetime of the first artificial satellite was 92 days. In this time it made about 1400 revolutions about the Earth. Subsequent satellites had considerably longer lifetimes.

The orbital parameters of the Soviet satellites which also determine, in particular, their lifetime, were selected on the basis of the scientific problems posed by the research program.

The orbital planes of the first Soviet satellites were inclined 65° to the equatorial plane (the so-called orbital inclination). This means that the projection of the orbit onto the Earth's surface, i.e., the satellite track, is located from 65°S to 65°N; consequently, it passes practically between the north and south polar circles. Such an orbit has the advantage that the scientific apparatus on board the satellite can take measurements within a broad range of latitudes.

Because of the diurnal rotation of the Earth, each successive revolution of the satellite should be shifted to the west by an angular distance equal to the rotation of the Earth during one revolution of the satellite. This would take place if the orientation of the satellite's orbital plane remained constant with respect to fixed

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* See Chapter VII, § 3: "Aerodynamic Forces."

stars. Actually, however, because the Earth's gravitational field does not have central declination (the influence of the Earth's flattening at the poles), the satellite's orbital plane will slowly turn about the Earth's axis, with so-called precessional movement. The rate of precession of the orbit of a satellite is not very great: the angular shift of the orbit is only several degrees per day. The precession decreases for orbits with greater inclinations, and also for orbits which are further out.

The direction of the precession depends on the insertion azimuth, i.e., the azimuth at which the satellite was inserted into orbit:* with eastern azimuths, the precession occurs opposite to the Earth's rotation; with western azimuths, the precession occurs in the same direction as the Earth's rotation. In the first case, the additional movement of the track for each revolution will be in a westerly direction, in the second case — in an easterly direction.

Figure 2.64 shows the daily track for the first earth satellite.

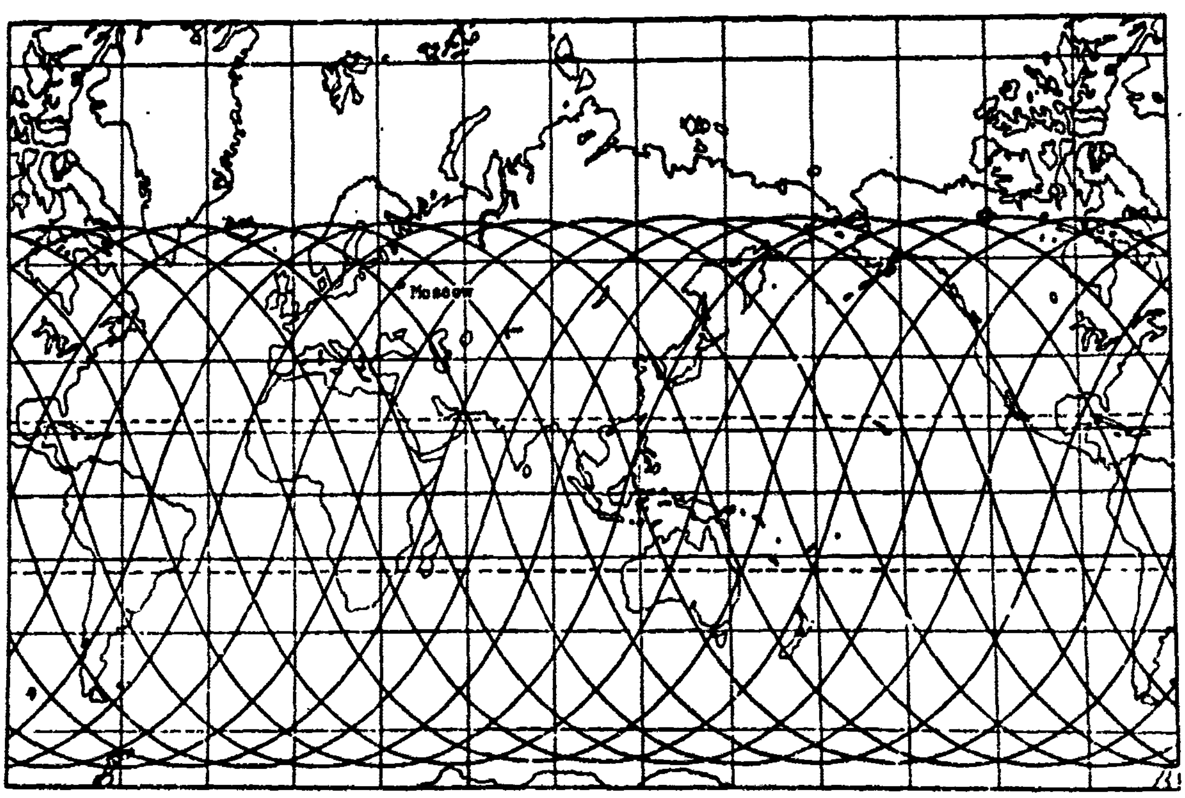

Fig. 2.64. Diagram of the movement of the first satellite in one day.

Inserting a Satellite into Orbit

In order to insert a satellite into orbit, the carrier rocket must lift it to a sufficiently great height, and at the same time accelerate it to the velocity required

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* The insertion azimuth is the angle between the meridian and the orbital track at the insertion point (usually reckoned clockwise from north).

for it to move in a given orbit.

We must bear in mind that a satellite is inserted into orbit from a rotating Earth; therefore, the absolute terminal velocity of the carrier rocket will consist of its relative velocity, i.e., the velocity relative to a fixed Earth, and the translational velocity caused by the Earth's diurnal rotation. An increase in the terminal velocity due to the Earth's rotation depends mainly on the orbit inclination, i.e., it is determined by the insertion azimuth. The smaller the orbit inclination, the greater the velocity increment. With zero inclination, i.e., when launched in an easterly direction in the equatorial plane, this increment becomes maximum: the peripheral velocity of diurnal equatorial rotation is 465 m/sec (Fig. 2.65).

This is why it is more difficult to insert a satellite into orbit with greater inclination. For example, when the Soviet satellites are launched, only 200 m/sec of the Earth's diurnal rotation is utilized. If a satellite is inserted into orbit with westerly azimuths (orbit inclination greater than 90°), it would even be necessary to overcome the peripheral velocity of the Earth's rotation.

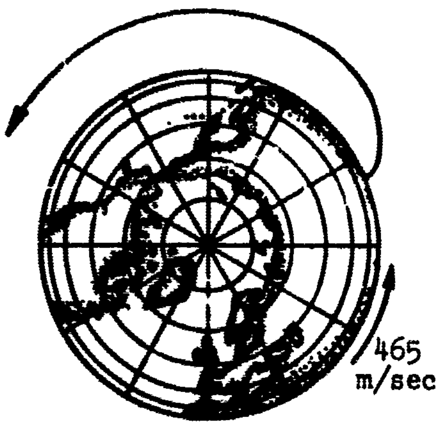

Fig. 2.65. Launching a satellite, making maximum use of the Earth's diurnal rotation.

We have seen that the terminal velocity of a carrier rocket, necessary to sustain the satellite, decreases with increasing insertion height. However, this in no way indicates that it is easier to insert a satellite at a greater height than a lesser one. The fact of the matter is that as the carrier rocket rises, there are increased losses in velocity due to overcoming gravity (see Chapter I). The increase of these losses with increasing altitude is more significant than the decrease in required velocity. Therefore, increasing the insertion altitude involves additional difficulties. Clearly, insertion into orbit is done most simply at the perigee.

Great significance is placed on the program for turning the carrier rocket when a satellite is inserted into orbit. Depending on this program, the insertion trajectory and the parameters of the final orbit will change. In the initial stage of the flight, the trajectory of the carrier rocket is analogous to that of an LRBM. The vertical launch phase and the relatively steep initial phase of the trajectory assure, on the one hand, maximum simplicity of the launch apparatus and most favorable flight conditions immediately after launch, and, on the other hand, as rapid as possible emergence from the dense layers of the atmosphere.

The satellite launch program has its own special features, determined by the requirements demanded of the satellite orbit.

In the simplest case, to produce an orbit with a relatively low perigee the turn program should assure a zero angle of inclination of the velocity vector to the local horizon $\vartheta_A = 0$ at the moment a given velocity is reached at the given height (Fig. 2.66a).

An analysis of the problem has shown that to obtain an orbit with a maximum perigee, the insertion program should be associated with the law of a change in mass along a trajectory.* In other words, the continual change in mass as a result of the engine operation, as well as the sporadic change in mass for multi-stage rockets, must have some specific correspondence to the trajectory parameters. We can, e.g., at point C of the trajectory (Fig. 2.66b) cut off the engine, leaving a specified amount of fuel unexpended. Since at point C the angle ϑ_C retains its positive value, the rocket will continue to ascend with decreased velocity, moving along some transitional ellipse. If we use the remaining fuel supply, and use it up in the region of the apogee of the transitional ellipse so as to obtain a velocity greater than circular at a given height, we get an orbit whose perigee is at the apogee of the transitional ellipse. We can also assume that various stages of the multi-stage rocket will be used in the phase where the satellite is inserted into the transitional ellipse, and in the burn-out phase in the region of the apogee of the transitional ellipse. Naturally, the individual examples discussed by no means cover all possible cases of a change in the thrust of the engine and the separation of the stages of a multi-stage rocket along the trajectory. The thrust can be continually changed by changing the operational regime of the engine according to a standard law.

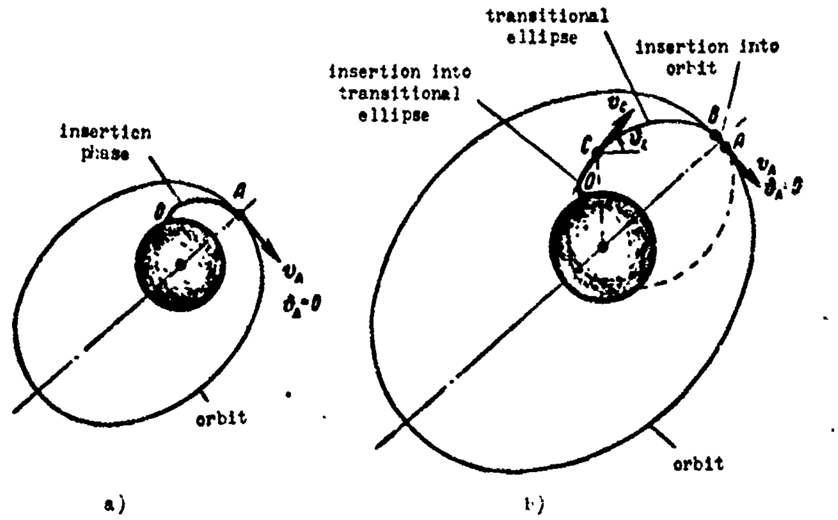

Fig. 2.66. Various means of inserting a satellite into orbit. a) with a low perigee; b) with a high perigee.

As has already been stated, the optimum insertion programs and engine operation regimes are selected together with all the other design parameters of a projected rocket. In particular, one should never judge the measure of applicability of some regime or other without first taking into consideration changes in the weights of the design elements as a function of insertion conditions.

* Tsiolkovsky, in his work "A Rocket to Outer Space" (1903) was the first to mention these characteristics of rocket ascent along an inclined trajectory.

The problem of inserting satellites into orbits is extremely complex from an engineering standpoint, and the difficulties increase as the satellite is made heavier. However, this is just the direction being taken by the development of contemporary rocket structure, since only by the creation of large artificial satellites can we solve the problem of the mastery of outer space.

The Design Characteristics of Satellites

In recent years we have quite clearly demonstrated the possible design variants and characteristics, both of carrier rockets and the satellites themselves.

Before going on, we should state that cosmic velocities can be attained at present only through the use of multi-stage rockets. In order to insert a relatively heavy satellite into a stationary orbit, the rocket must be of high structural quality.

To obtain maximum specific thrust, the engine of the carrier rocket must use those fuels which have the highest calorific values.

The scientific apparatus of the satellites conducts a great many scientific investigations, the value of which is increased by the fact that they are carried out simultaneously. The main advantage of satellites over geophysical rockets is that they make possible prolonged cycles of physical investigations.

The design of the satellite can be quite varied, depending on the scientific apparatus. Power supply and thermal control systems are common to all satellites.

Electrochemical sources (accumulators) and semiconductor solar batteries, which convert the energy of the Sun's rays directly into electrical energy, serve as the power supply for the measuring, recording, and transmitting apparatus on satellites.

The thermal control system should assure a stable temperature regime for the entire time that the apparatus is in operation.

As it orbits, a satellite is periodically subjected to sharp variable heat effects (flight in the Sun's rays and the Earth's umbra, the thermal effect of the atmosphere). In addition, a certain amount of heat is generated by the apparatus operating inside the satellite. The maintenance of the required heat regime in the satellite for prolonged space flight is a relatively new and quite complex problem. The temperature regime can be controlled either by varying the forced circulation of a special gas coolant inside the satellite, or by changing the value of the inherent radiation and solar-radiation absorption factors of the outer surface of the

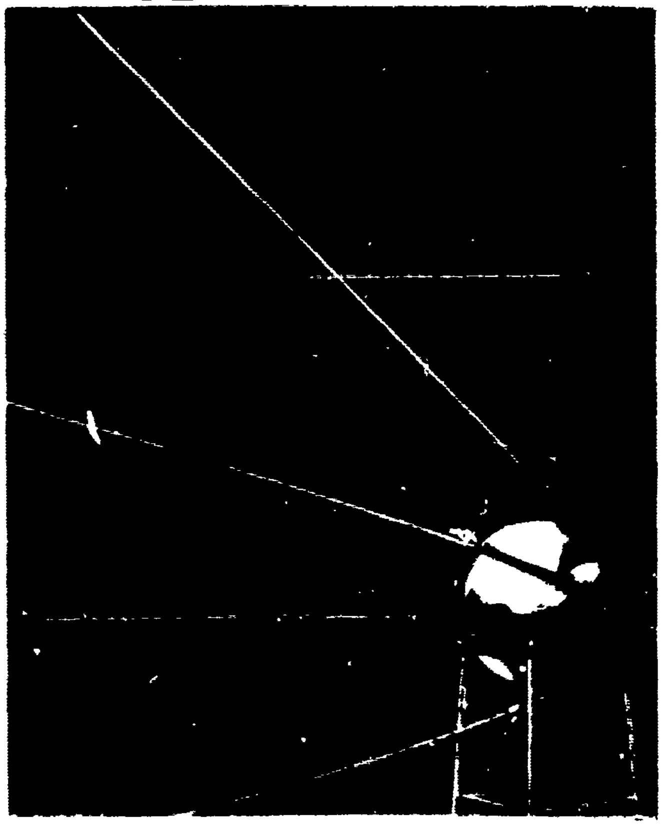

Fig. 2.67. ... earth satellite.

satellite.

The first artificial satellite (Fig. 2.67) was a sphere 58 cm in diameter with four projecting antennas which were unfolded after the satellite separated from the carrier rocket. The satellite weighed 83.6 kg. An airtight housing made of aluminum alloy with a specially prepared surface was filled with gaseous nitrogen, which was force-circulated to control the heat regime (the filler valve can be seen in Fig. 2.67). Powerful radio transmitters in the satellite emitted signals which were coded to correspond with the readings of the apparatus in the satellite.

Another possible version includes satellites which do not separate from the carrier but ... the ... stage of the rocket. Such was the second Soviet satellite ... Layka. The total weight of the scientific-measurement and ... apparatus, the airtight container with the experimental animal, and ... was 508.3 kg.

A considerable ... in the improvement of the design of satellites was the creation ... artificial satel... on May 15, ...

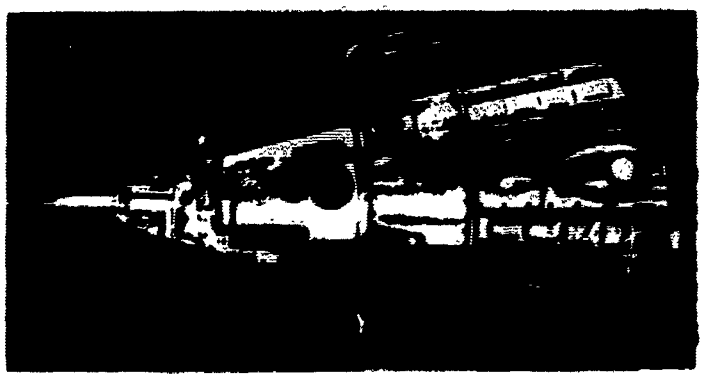

Fig. 2.68. General view of the third Soviet artificial earth satellite.

... a true space ... number of ... The great ... the satellite ... up of the ... the mutual ... instruments as much as possible.

... container, 3.57 m tall, 1.73 m base diameter, ...

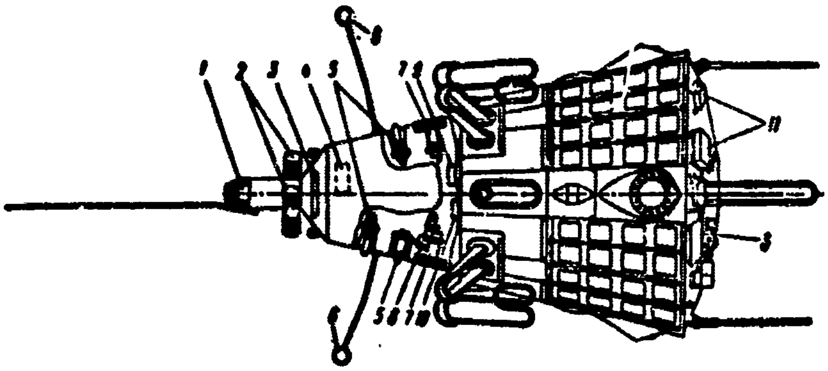

Fig. 2.69. Scientific apparatus of the third Soviet artificial earth satellite.

1) magnetometer; 2) photomultipliers for recording solar corpuscular radiation; 3) solar cells; 4) cosmic-ray photon recorders; 5) magnetic and ionization magnetometers; 6) ion traps; 7) electrostatic fluxmeters; 8) mass spectrometer tube; 9) cosmic-ray heavy nuclei recorder; 10) device for measuring the intensity of primary cosmic radiation; 11) micrometeor sensors. The electronic units for the scientific apparatus, the radio measurement system, the programmer, and the electrochemical power supplies are contained within the satellite.

Figure 2.69 shows the positioning and composition of the scientific apparatus in the satellite.

In addition to its chemical power supply, the satellite was equipped with groups of solar cells which assured normal operation of the apparatus for many thousands of hours.

The temperature control system was considerably improved over the first satellites. Changes in the reflecting and absorbing properties of the surface were automatically controlled by the opening of special folding shutters, whose electric motors were controlled by the temperature control system (four shutter flaps can be seen on the right in Figs. 2.68 and 2.69).

The satellite was equipped with improved radio engineering measuring apparatus for accurate measurement of its orbital movement, and radiotelemetry apparatus to continually record the results of the scientific measurements. These results were "stored" and then transmitted when the satellite passed over special stations in the USSR.

A special programmer guaranteed automatic functioning of all the scientific and measurement apparatus of the satellite.

The apparatus in the satellite made wide use of semiconductor elements.

Basic Trends in the Development of Space Flights

The solution of the problem of creating satellites has opened broad vistas in the explanation of the physical conditions of outer space, and the time is fast approaching when man will make his first space flight.

On the whole, the problem of mastering interplanetary space is unusually complex. All those interested in this question would do well to remember the words of Tsiolkovsky:

"Stellar navigation cannot be compared with flying in the air. This latter is mere childsplay compared with the former.

"...If they know the difficulties involved, many of those now working with such enthusiasm would back away in horror.

"But how rich will be the rewards!"

He goes on to say:

"Man will not remain eternally on Earth, but in pursuit of light and space he will at first timidly reach beyond the atmosphere, and finally conquer all near-solar space."

Time has proved Tsiolkovsky completely correct. He was right in stating not only that the problem of space flights is very complex, but also that man would successfully solve it. We are today witnessing the first fruits of science and engineering along these lines.

The space age was opened on January 2, 1959. The first Soviet space rocket became the first in history to overcome the Earth's gravitational field and become an artificial planet of the solar system (Fig. 2.70). The rocket approached within 6000 km of the Moon. Radio-telemetry communication with the rocket was maintained for a distance of over 500,000 km from the Earth.

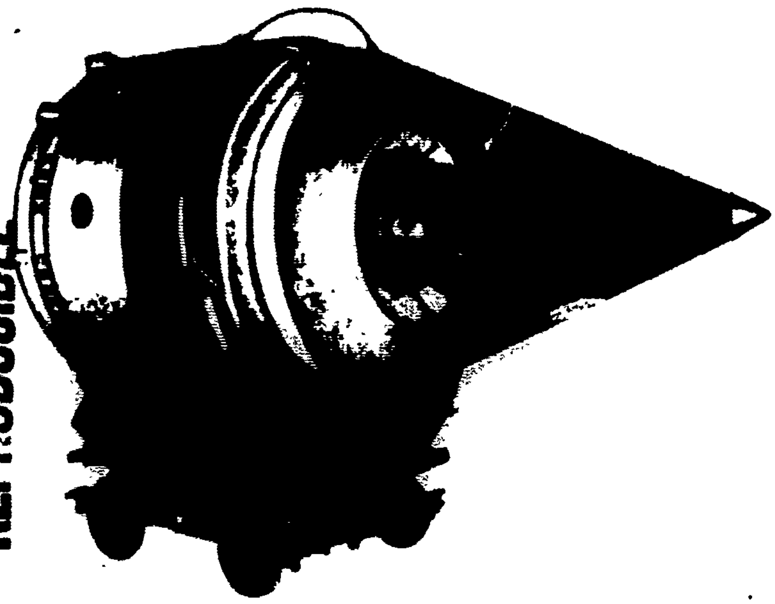

Fig. 2.70. Mock-up of the last stage of the first space rocket on the assembly stand. Half of the nose cone has been cut away, showing the spherical instrument container with the apparatus.

The next noteworthy step toward mastering of space was the launching of the second Soviet space rocket toward the Moon. A banner with the shield of the Soviet Union was placed on the Moon on September 14, 1959. For the first time a space flight had been made from the Earth to another heavenly body.

On October 4, 1959, the Soviet Union successfully launched the third space rocket. This rocket was intended to solve a number of problems on the study of outer space. The most important of these was the photographing of the far side of the Moon, never seen on Earth.

An automatic interplanetary station (AIS) (Figs. 2.71 and 2.72) was created to study and photograph the Moon; this was a multi-stage rocket which was exceptionally accurately inserted into orbit around the Moon (Fig. 2.73).

The space station was equipped with an orientation system used to train the

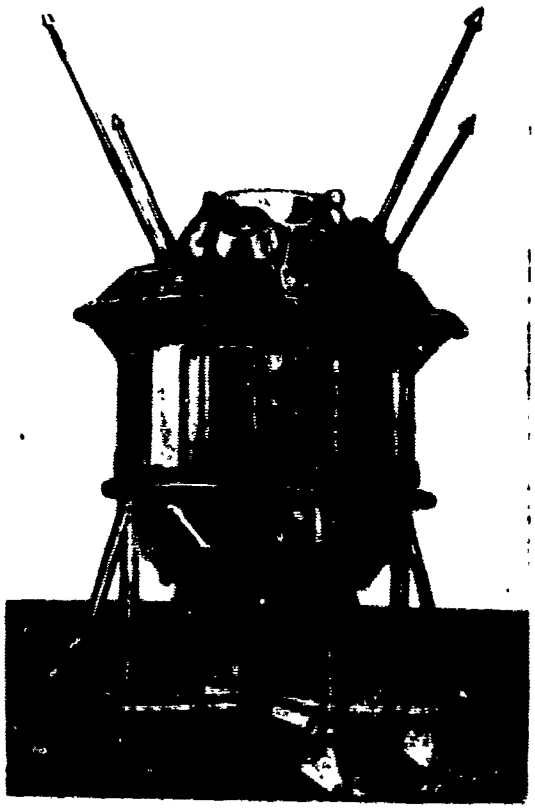

Fig. 2.71. Automatic interplanetary station on its assembly truck.

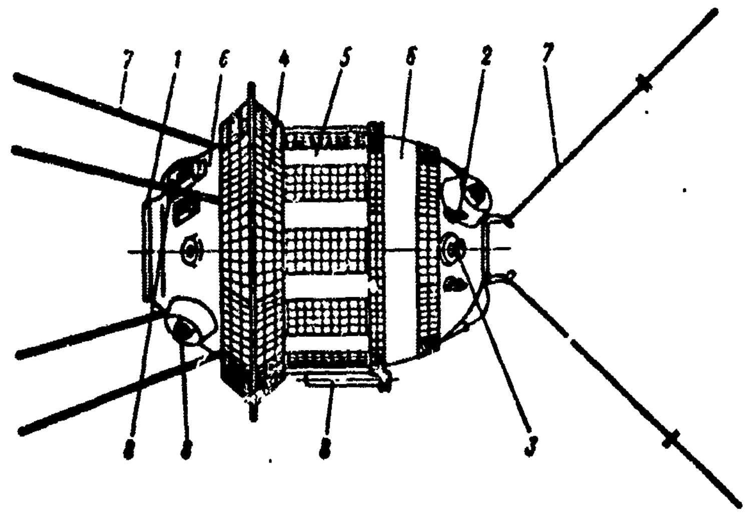

Fig. 2.72. Diagram of the automatic interplanetary station. 1) porthole for the photographic apparatus; 2) orientation system motor; 3) solar transducer; 4) sections of the solar cells; 5) temperature-control system shutters; 6) heat shields; 7) antennas; 8) scientific research instruments.

photographic apparatus on the Moon in order to photograph it. The film was then developed and fixed automatically. Then, as the AIS approached the Earth the photographs were transmitted via a radio-television system. Thus, the first television transmission was made from space. Now scientists had at their disposal photographs of the part of the lunar surface never seen by ground observers. Simultaneously, photographs were taken of that part of the lunar surface which is visible on the Earth at a very small angle and which, till now, had not been studied sufficiently. A special commission from the Academy of Sciences of the USSR confirmed the nomenclature of the reliably established formations on the far side of the Moon. Table 2.1 gives a resume of the basic data on the first Soviet artificial earth satellites and the first Soviet space rockets.

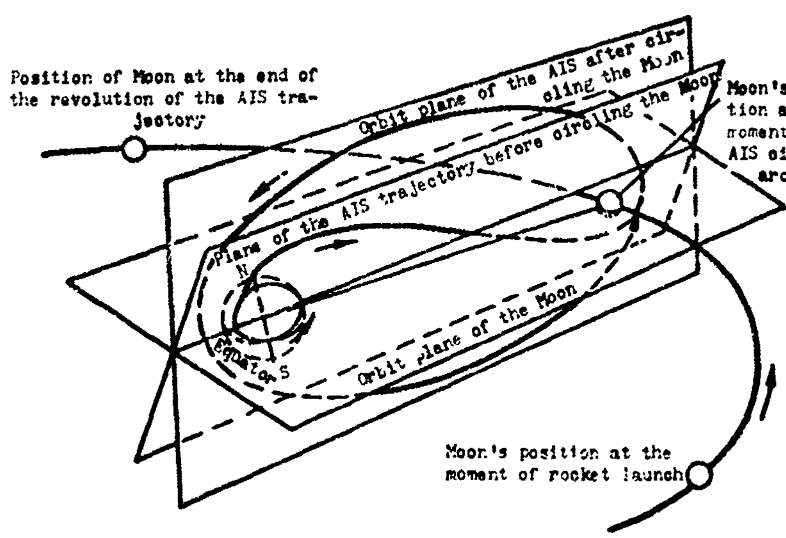

Fig. 2.73. Diagram of the flight trajectory of the automatic interplanetary station (AIS).

Table 2.1

Name	Launch date	Weight of last stage of rocket without fuel, kg	Weight of scientific apparatus and power supply, kg	Orbit parameters (initial)				Lifetime	Remarks
				Apogee, km	Perigee, km	Inclination to equatorial plane, degrees	Orbital period		
First Soviet artificial earth satellite	Oct. 4 1957	Weight of satellite 83.6 kg	—	947	228	65	96,17 min	92 days	Circular velocity attained for the first time; artificial earth satellite created
Second Soviet artificial earth satellite	Nov. 3 1957	—	508,3	1671	225	65	103,75 min	162 days	First flight of a living creature — the dog Layka — on an earth satellite
Third Soviet artificial earth satellite	May 15 1958	Weight of satellite 1327 kg	968	1881	226	65	105,95 min	More than 20 months	Largest space laboratory in the world
First Soviet space rocket	Jan. 2 1959	1472	361,3	Aphelion $197 \cdot 10^6$ km	Perihelion $146 \cdot 10^6$ km	1° to orbit plane of Earth	450 days	Eternal solar satellite	Escape velocity attained for the first time
Second Soviet space rocket	Sept. 12 1959	1511	390,2	—	—	—	—	—	First rocket to reach the surface of the Moon; September 14, 1959, $0^h2^m24^s$
Third Soviet space rocket	Oct. 4 1959	1553	435	485,000 km from center of Earth	47,500 km from center of Earth	—	About 21 days	11-12 revolutions about the Earth	First flight around the Moon, photography of the far side of the Moon

Processing of the observational data from Soviet artificial earth satellites and space rockets, and also of the numerous records of the instruments contained therein, produced results of fundamental scientific value on investigations of the upper layers of the atmosphere and outer space; made it possible to discover many new, previously unknown, regularities in the study of radiation about the Earth and in outer space, study of the upper atmosphere, investigation of interplanetary gas, study of the magnetic fields of the Earth and the Moon, study of meteor and micrometeor dangers, and biological investigations, which are gradually becoming a new science — space biology — one of whose main tasks is to assure safe manned space flight.

The results of these investigations have already made it possible to predict the trends of further space flights.

The first of these is the systematic flight of satellites near the Earth. Further operations on the creation of artificial earth satellites, besides expanding the scientific studies already conducted, can also be used to solve a number of applied technical problems, through the use of satellites.

It will be advantageous to create artificial satellites oriented in some specific manner in space. Such orientation is required to solve many scientific problems. For a number of investigations involving the Sun, the satellite should be oriented toward the Sun. For investigations involving the Earth and the atmosphere the satellite should be so oriented that one of its axes is directed toward the Earth, and the other coincides with the direction of the orbit. For astrophysical studies it will evidently be necessary to have a satellite which will have a fixed position relative to the fixed stars.

An important step in the development of space flights will be manned satellite flights; this requires the solution of a great many problems involving flight safety, including the development of absolutely reliable propulsion units, systems for stabilizing and guiding the rocket in flight, and a recovery system for the descent. We must also have the necessary life-support systems during takeoff and landing, under conditions of high g-forces, and also during the orbital flight, during weightlessness. The experiments with test animals, conducted using the second Soviet artificial earth satellites and the flights of numerous geophysical rockets (using these same animals), have given the first significant results along these lines, producing scientific data on the effects of space flight on a living organism.

The idea has often been expressed of the possibility of using a system of special satellites for the rebroadcasting of television transmissions; this would allow long-range transmission on the usw band without resorting to radio relay lines and cables.

Using satellites, a service can be established for the constant observation of solar corpuscular radiation; it will be able to forecast the most important phenomena occurring in the upper layers of the atmosphere.

It is still difficult to foretell all the possible scientific and practical uses of satellites, just as at the dawn of aviation it was impossible to foresee the diverse spheres of applicability and the vast progress of aviation to the present.

A second direction in the development of space flight is the solution of problems associated with lunar flights and investigations.

We can expect that in the future it will be possible to send a flight to the Moon, land, and then return to Earth. The problem of landing equipment on the Moon is quite complex, as is the problem of the subsequent launching from the Moon and the return to Earth.

In the even more distant future it might become possible to establish special stations on the Moon, similar to those scientific stations which have been organized in the relatively inaccessible regions of the Earth, e.g., in the polar regions. In this regard we should mention the extreme complexity of such an undertaking. It can be done, but only as a result of further progress in rocketry, and the solution of a number of scientific and technical problems. But it may turn out that projects which today seem completely fantastic and impossible will be realized considerably sooner than would appear at first glance.

A third group of problems which forms an independent direction in the development of space flights consists of those problems associated with the study of near-solar space and the planets of the solar system.

One of the purposes of flights in the solar system is the direct study of the interplanetary medium. The probing of interplanetary space using scientific apparatus will make it possible to establish the density of interplanetary gas at various distances from the Sun, determine the chemical composition of interplanetary gas, obtain new, extremely interesting, data on the intensity distribution and the composition of cosmic radiation in various regions of the solar system, investigate various forms of solar radiation, and study the Sun's magnetic field and its influence on phenomena in the interplanetary medium.

Of particular interest is the investigation of the planets of the solar system, primarily Venus and Mars.

By sending rockets equipped with automatic instruments to these planets we can study their magnetic fields and the radiation belt, and obtain detailed pictures of their surface. It will b. possible to investigate the atmosphere of the planets: determine its density, chemical composition, and degree of ionization, and also investigate the structure and temperature of the surface of the planet. Finally, the prospect of studying forms of life on other planets is intriguing. Manned flight to the planets belongs to the future, but this day is fast approaching.

The development of space flight and rocket vehicles has brought to life a new branch of contemporary technology — space technology — which should make possible space flights, flights to the Moon, and flights to other planets.

Space technology will make use of various devices with scientific-measuring, recording, and transmitting apparatus, which will automatically fulfill the observational program and transmit the results to Earth.

An important problem in space technology is that of long-range and extralong-range radio communication which is necessary for determining the trajectory parameters, transmitting the measurement results and information on the operation of the apparatus to Earth, and also for transmitting commands from Earth.

For flights within the solar system we must have radio communication and image transmission of the order of tens and hundreds of millions of kilometers. Therefore, of fundamental importance is the problem of creating light, small, and very economical radio apparatus, and also powerful transmitting and receiving devices on Earth.

All the apparatus of space rockets and aggregates must be not only as light and economical as possible, but must also be extremely reliable, capable of operating faultlessly for many months and even several years. Such time periods are characteristic of flights within the solar system; this is not surprising, if we remember the length of the rotation periods of the planets. The specific nature of the operation of the apparatus in space is also determined by the effects of cosmic radiation and the presence of the deep vacuum which surrounds a space ship. Another important factor is the need for prolonged maintenance of a specific temperature regime. Space technology also requires the development of reliable and light power sources, using primarily solar energy.

C H A P T E R III

ROCKET ENGINES, THEIR CONSTRUCTION AND OPERATIONAL FEATURES

1. Power Conversion

Power Conversion System

To create thrust in a direct-reaction engine, velocity must be imparted to the exhausted mass of the combustion products. This velocity is obtained as a result of the conversion of the chemical energy of the fuel into the kinetic energy of the gas jet.

Let us examine the sequence of the processes occurring in a direct-reaction engine (Fig. 3.1), using a liquid-fuel rocket engine as an example (Fig. 3.2).

The propellant of the majority of rocket engines consists of two different substances — the oxidizer and the fuel. These are called the propellant components.

The oxidizer contains a large amount of oxygen or some other oxidizing element. The fuel consists mainly of fuel elements which are capable of producing great quantities of heat in chemical reaction with the oxidizing element. Thus, rocket engine fuels are chemical energy carriers (see Fig. 3.1).

The properties of fuel will be examined in more detail in the next chapter.

The propellant components are located in the engine tanks; these form the biggest part of the rocket — the fuel compartment.

The fuel burns and the combustion products* subsequently expand in the thrust

* The term "combustion products," widely used to describe and study heat engines, is also applicable to rocket engines. We should make the reservation, however, that in the latter case it is not accurate, since it infers that the fuel combustion process goes to completion. In rocket engines, particularly liquid-fuel engines, the combustion process, even in the nozzle exhaust area, does not go to completion. Therefore it would be more correct to use "burning products" instead of "combustion products." However, following tradition, we will not change this term, but we will give it a slightly different meaning. This also is true of the term "combustion chamber."

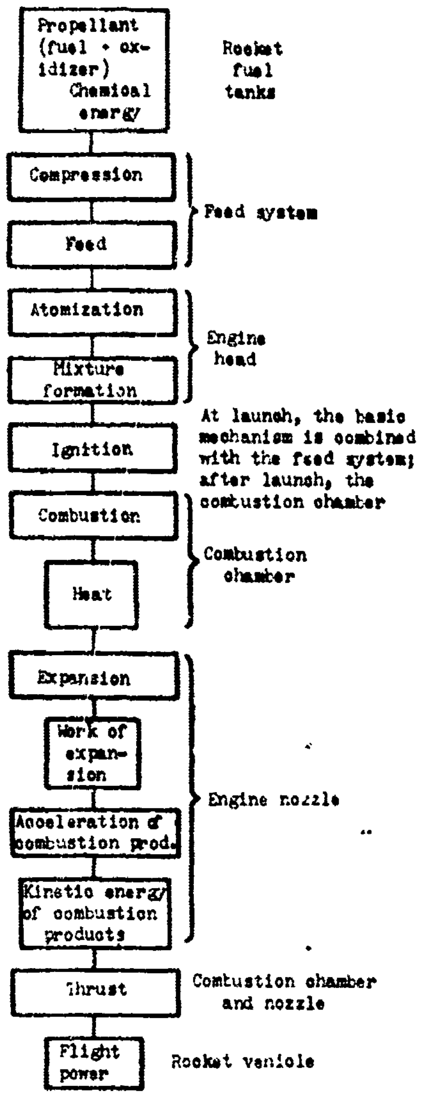

Fig. 3.1. Power conversion system in direct-reaction liquid-fuel rocket engines.

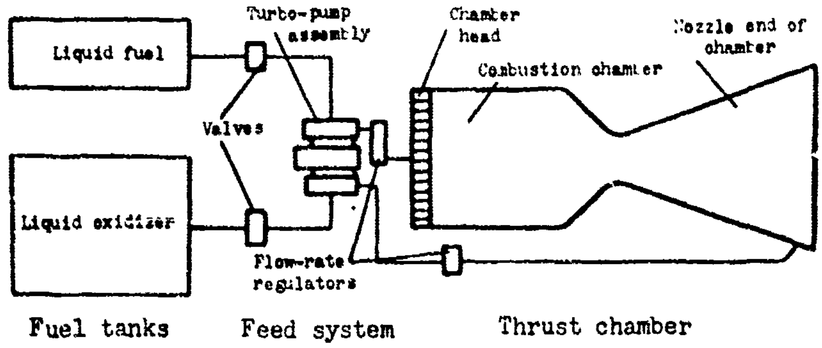

Fig. 3.2. Diagram of a liquid-fuel rocket engine.

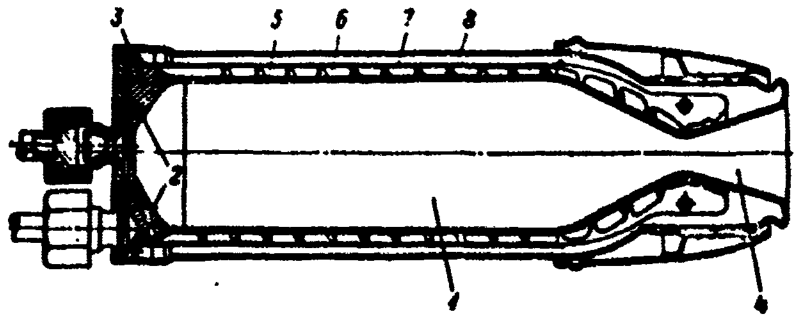

Fig. 3.3. Liquid-fuel thrust chamber. 1) combustion chamber; 2) injectors; 3) chamber head; 4) nozzle end of chamber; 5) inner shell; 6) cooling passage (cooling jacket); 7) outer shell; 8) component feed to cooling passage.

chamber. The thrust chamber consists of the combustion chamber, with fuel injectors in its head, and the nozzle (Fig. 3.3).

Before combustion begins, the fuel must pass through several preparatory stages. First, the fuel should be compressed to a pressure greater than that in the combustion chamber, otherwise it will not enter the chamber. The fuel is compressed and fed to the engine by a feed system, consisting of a unit to produce pressure (e.g., tanks of compressed air, a turbo-pump assembly, etc), and also a system of valves, regulators, and pipes to assure reliable operation and the possibility of controlling the engine.

When fed to the chamber, the fuel should be supplied in a state most favorable for combustion. The fuel and the oxidizer should form a highly uniform vapor. To obtain such a mixture the fuel, accelerated by the nozzles in chamber head 3 (Fig. 3.3), is atomized into small droplets which vaporize as they move through the chamber. After vaporization, the vapors of the propellant components mix. This is the process of mixture formation.

The ignition and burning of the fuel in the combustion chamber are the next necessary steps in the working process of the engine. Upon combustion, a considerable amount of the chemical energy is converted into heat which is used to increase the heat content (enthalpy) of the combustion products.

The main part of the heat evolved in the combustion chamber (i.e., part of the enthalpy of the combustion products) is converted, in a direct-reaction engine, into the kinetic energy of the gases exhausted from the nozzle. This occurs due to the expansion of the combustion products as they move through the nozzle.

The operational efficiency of a rocket engine as a heat machine is determined by the efficiency of the combustion and expansion processes; this determines the amount of chemical energy of the fuel which converts into kinetic energy.

One of the propellant components, e.g., the oxidizer (see Fig. 3.3), passes through a special cooling passage (the cooling jacket), lowering the temperature of the inner shell which comes in contact with the incandescent combustion products. The directed exhaust of gases from the nozzle leads, as has already been mentioned, to the occurrence of thrust. As a rocket vehicle moves the thrust creates work, which is used to create the kinetic energy of the vehicle.

All these processes which lead to power conversion are also accomplished in any other type of reaction engine.

Basic Types of Reaction Engines

The type of fuel used determines, for the most part, the design of the engine and its basic units. Using this criterion, all existing reaction engines can be divided into two basic groups: air-breathing jet engines and rocket engines.

The distinguishing feature of the air-breathing jet engine is its use of oxygen from the air as the oxidizer.

The advantage of such engines is that they make it possible to cut down considerably on the amount of fuel aboard the vehicle (for present-day rocket fuels, the oxidizer constitutes about 60-80% of the weight). This also makes oxidizer tanks unnecessary.

On the other hand, the use of atmospheric oxygen makes air-breathing jet engines inoperative at heights where the oxygen content is low due to rarefaction of the atmosphere.

The second group consists of engines which operate on fuels entirely self-contained in the vehicle. Such engines are called rocket engines [RF]. Their main advantage, as we have seen, is that their operation is completely independent of the surrounding medium. Rocket engines can be used at any height, in particular for flights in interplanetary space.

Rocket engines are usually divided into engines which operate with a liquid propellant — the liquid-fuel rocket engines [LFRE] — and solid-fuel rocket engines [SFRE].

Until recently, rocket engines used low explosives — powder — exclusively as the solid fuel. Therefore, SFRE's were called simply powder engines. At present, they operate not only on powder but on other forms of solid fuel as well (see Chapter IV), and therefore it is more correct to call such engines solid-fuel engines.

Finally, the development of rocket engines has seen the use of hybrid propellants, where the fuel was a solid, located in the chamber, and the liquid oxidizer was fed to the chamber from a tank by means of a feed system. Certain types of proposed nuclear rocket engines can also be classified as hybrid-component engines.

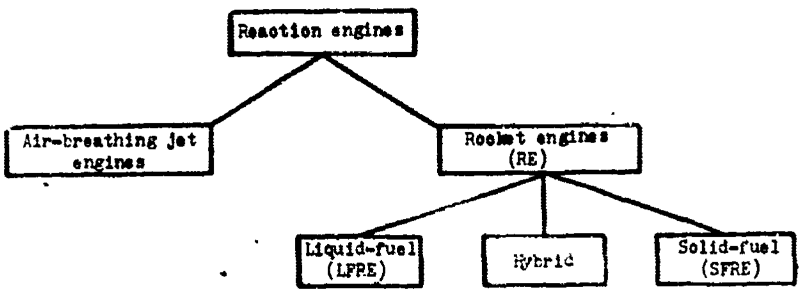

Fig. 3.4. Breakdown of reaction engines into basic groups.

Figure 3.4 gives the breakdown of reaction engines into the basic groups.

Let us examine LFRE's in more detail. We will not discuss the construction of powder engines and other solid-fuel engines, since their design is so simple that it is easily understood without explanation (see the examples of powder rockets in the previous chapter).

Air-breathing jet engines are a special field of study, outside the framework of this book. Much literature exists on the subject.

2. Liquid-Fuel Rocket Engines

Types of Existing Liquid-Fuel Rocket Engines

In classifying LFRE's (Fig. 3.5) we must first consider the type of fuel used in the engine.

Since the features of a certain propellant are determined mainly by the oxidizer, the breakdown of LFRE's should first be made according to the type of oxidizer used (nitric acid, LOX, hydrogen peroxide), and then by type of fuel (kerosene, ethanol, etc.).

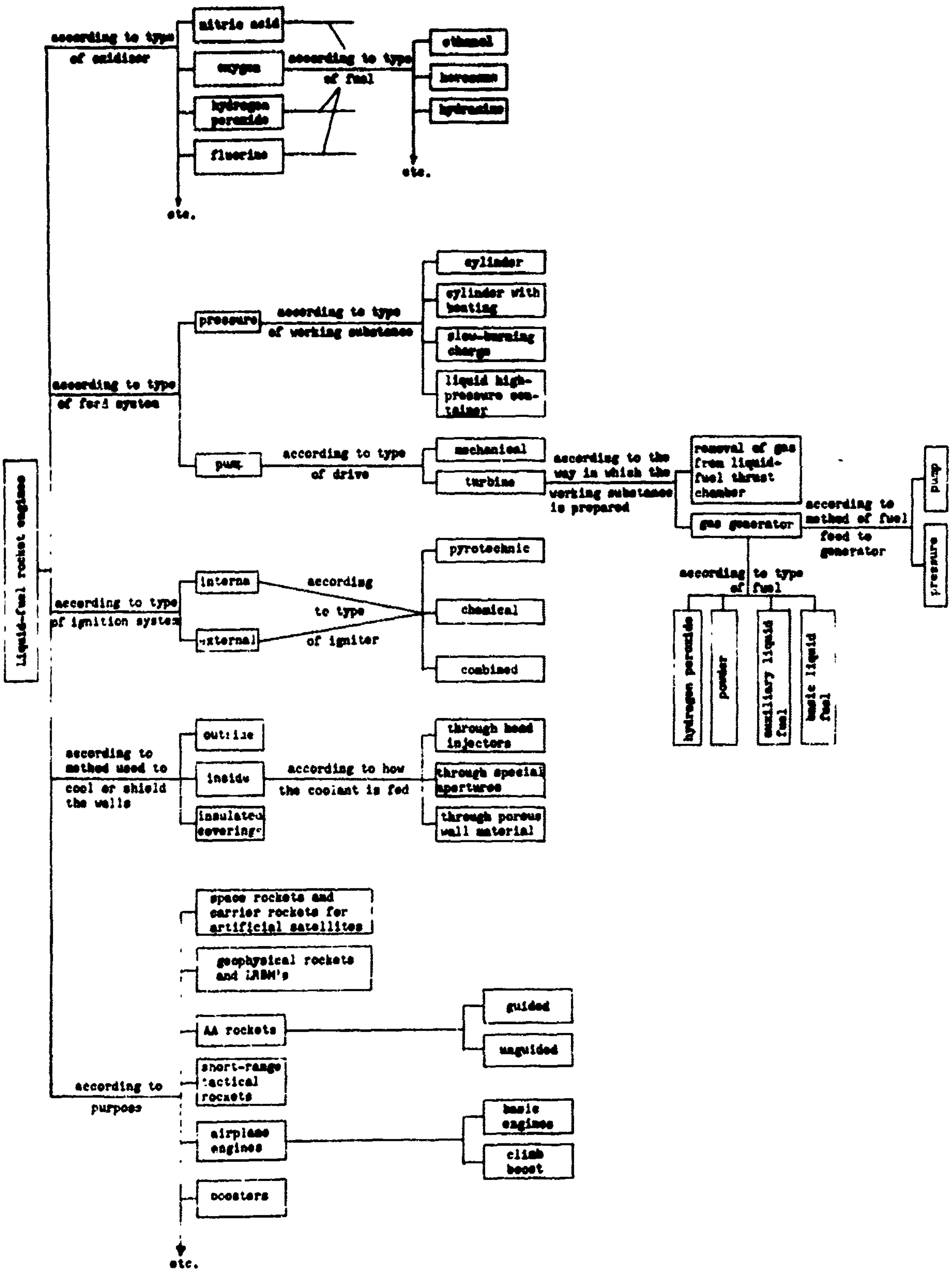

Fig. 5.5. Classification of liquid-fuel rocket engines.

The fuels of LFRE's will be discussed in more detail in the next chapter.

A second characteristic which typifies LFRE design is the method of feeding the propellant. There are two methods: pressure and pump feed. The first is the simpler of the two, and is used in the engines of relatively small rockets. The second is used in LRBM's with thrusts exceeding, as a rule, 20 tons, and also in aircraft engines having relatively long operation times.

With pressure feed the propellant components are fed to the combustion chamber by compressed gas which enters the fuel tanks at high pressure. The gas pressure in this case should exceed the pressure in the combustion chamber. The excess pressure is necessary to compensate for losses in the feed passages (these include the pipes, valves, injectors, cooling passages, etc.).

Thus, using this feed system, great pressure is created in the rocket's fuel tanks. Consequently, the tanks must be quite thick-walled in order that they be strong enough. This causes great weight of the structure, and is a common disadvantage of all pressure-feed systems.

The simplest type of pressure feed, long used in rocket engines, is tank feed (Fig. 3.6). Air, or some other gas, in tank 1, compressed to high pressure (250-300 kg/cm^2) is used in this system for creating pressure in the tanks and for feeding the fuel. The air tank is usually spherical, since the weight of such a tank for a given volume will be minimum, compared with other tank shapes.

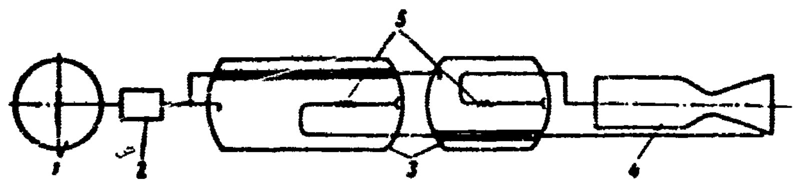

Fig. 3.6. Diagram of tank pressure feed. 1) compressed-air tank; 2) gas pressure reducer; 3) fuel tanks; 4) thrust chamber; 5) flexible elements of the intakes.

An integral part of the tank feed is gas pressure reducer 2. This is an attachment which assures constant gas pressure in fuel tanks 3 and, consequently, uniform fuel feed to chamber 4, necessary to keep the operational regime of the engine unchanged. Without a pressure reducer, the pressure in the tanks at the outset would be very high and then, as the fuel is consumed and the gas expanded, it would drop considerably. We will discuss the design of the reducer below.

As can be seen from Fig. 3.6 and the description given, tank pressure feed is very simple. Therefore it is used in many engines. However, tank feed has one essential drawback, in addition to that inherent (in general) in all pressure feeds (heavy fuel tanks). This is that the weight of the gas required to remove all the fuel is very great. For example to feed 1 m^3 of fuel with a fuel-tank pressure of 35-40 kg/cm^2, about 50 kg of air are required. In addition, to store this air supply on the rocket requires a 150-liter tank, even if the initial pressure in the tank is 300 kg/cm^3. Such a tank (spherical), made of high-grade steel, will weigh about 150 kg and have a diameter of 680 mm.

Thus, tank feed leads to a noticeable increase in weight of the propulsion unit. This disadvantage can be partially eliminated if we heat the air entering the tanks, compressing a small amount of fuel into it. In this case, for the same pressure there is a decrease in the specific weight of the gas entering the tank and, consequently, a decrease in the amount of air required to displace a given volume of fuel. There is also a decrease in the sizes and weight of the tank necessary to store the air on the rocket.

Even better, as far as weight is concerned, is a feed system in which the fuel is displaced by the hot combustion products of the powder charge (slow-burning charge). Such charges have recently been made. Slow-burning charges, per unit time, form a uniform amount of combustion products by weight, which produces uniformity in the displacement of the fuel from the tanks.

Because of the low specific weight of the hot gases which displace the fuel, only 12-15 kg of powder in all are needed to feed 1 m^3 of fuel (considering the inevitable cooling of the gas on its way to the tank, and in the tank itself). The powder, in solid form, has a sufficiently high specific weight, more than 1.5 kg/liter, and therefore the chamber where it burns will be small, about 12-15 liters. In addition, the powder can burn at relatively low pressures, only slightly exceeding those in the tanks. Therefore, the walls of the housing of the combustion chamber will not be very thick, and the housing will weigh only about 25 kg. Therefore, by using the combustion products of the powder charge to displace the fuel we can eliminate about 100 kg of weight for every cubic meter of displaced fuel (compared with tank feed).

In certain LFRE's, the liquid-fuel combustion products serve to displace the fuel. In other words, a small LFRE is used to displace the fuel. The weight of this system, called a feed system with a liquid high-pressure container, approximately equals that of a feed system using a powder charge.

The use of a slow-burning charge and liquid high-pressure containers for fuel feed is complicated by the high combustion temperatures, and the difficulties involved in producing stable uniform combustion of the liquid fuel and powder.

Generally speaking, other methods can be proposed for producing the high-pressure gas necessary to displace the fuel. Therefore, engines with pressure feed should be further classified according to the type of working substance used for displacement (cold compressed gas, hot compressed gas, powder gases, etc.).

In small and medium rocket engines, with not very high pressure in the combustion chamber, the weight of the tanks with pressure-feed systems is not very great. But in large rocket engines, particularly if they have high pressure in the chamber, the weight of the tanks increases inadmissably, and it is more suitable to use the pump-feed system.

In this system, as the name implies, special pumps serve to feed the propellant components from the tanks to the thrust chamber.

The pump-feed system differs favorably from the pressure-feed system in that high pressure need not be created in the engine tanks. There is some slight pressure (2-3 kg/cm^2) in the tanks of an engine with pump feed, in order to assure that the pumps will operate reliably. In this case, the wall thickness of the tanks need be no greater than required for engineering purposes.

Centrifugal pumps are generally used in LFRE's. This type of pump is convenient in that it can handle a great quantity of the components at high pressure, but is of small size. This can be done because of the high peripheral velocities, and the correspondingly high number of revolutions of the impeller (from 5000 to 25,000 rpm).

Some type of motor is needed to activate the pumps. The best motor for driving centrifugal pumps i a turbine capable of creating very high rpm. Thus, there is a single machine, usually having a common shaft and connecting the turbine with the required number of pumps. This is the turbo-pump assembly shown in Fig. 3.7.

The operation of the turbine requires heated gas or steam (the working substance) at high pressure. This working substance can be produced in various ways. The most convenient and widely used method is the decomposition of highly concentrated hydrogen peroxide in a special reactor. The heated mixture of oxygen and steam obtained from this decomposition can drive a turbine of the required power.

The classification diagram (Fig. 3.5) shows the possibility of various ways of producing the working substance for driving a turbine.

Besides the hydrogen peroxide steam generator, we have the liquid-fuel gas generator which operates on the basic propellant used in the thrust chamber. Engines

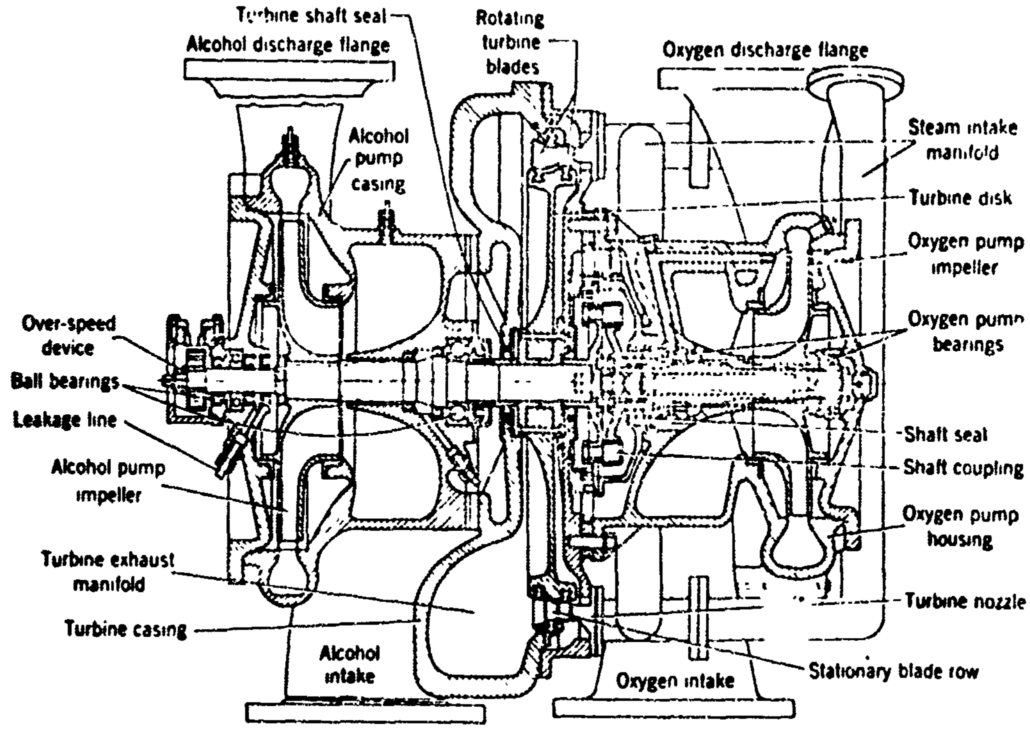

Fig. 3.7. V-2 turbo-pump assembly.

with such a gas generator should contain only two (basic) components, which simplifies their operation.

The propellant is fed to the gas generator using a special feed system which, just as the feed system of the entire engine, can be of two types: pressure and pump. For large rocket engines and airplane engines, the pressure system for feeding the propellant to the gas generator may be too heavy; then it is replaced by the pump system (see, e.g., the description of the engine of an interceptor, and Fig. 3.11).

Of importance to an LFRE is the method of igniting the propellant when the engine is started. The design and type of ignition system depends almost completely on the purpose of the engine.

Ignition can be mainly of two types: external and internal. External ignition is that in which the triggering (ignition) flame is introduced into the combustion chamber from without, through a nozzle. Such a system is used, in particular, when starting the engines of LRBM's. Internal ignition uses devices located on or in the combustion chamber head.

Ignition systems are also classified according to the type of igniter. As igniters we can use, e.g., powder charges ignited by an electric primer and which ignite the liquid propellant. Such an ignition system is called a pyrotechnic system. Often the propellant can be ignited by using hypergolic liquid fuels; this is the <u>chemical</u> ignition system. The simplest chemical ignition is used in liquid-fuel rocket engines that operate on hypergolic fuels. Here ignition occurs with the mere contact of the liquid-propellant components in the combustion chamber. If the basic fuel in nonhypergolic, chemical ignition is accomplished by installing auxiliary tanks containing hypergolic components. Hypergolic fuel is fed to the engine when it is started; it forms an ignition flame, after which the basic, nonhypergolic, fuel begins to enter the combustion chamber. In certain engines the ignition components are located in the basic fuel lines, separated from the basic components by thin diaphragms. As the pressure in the tanks increases, the diaphragms burst and the hypergolic fuel enters the engine, followed by the basic fuel.

Sometimes liquid propellants are ignited by an electric spark. However, this method is not reliable enough since if the engine is large, the electric spark is not powerful enough and there can be a delay or failure in the ignition. Therefore, electric ignition is used only in small engines. A more complex system is used in large engines; a spark plug ignites a small amount of the auxiliary fuel, and the basic fuel is ignited from the small flame that results.

Another characteristic of LFRE's is the method of cooling the chamber. Engine cooling by one of the components fed through a cooling passage (Fig. 3.3) is called <u>external</u> cooling.

In addition to external cooling, <u>internal</u> cooling is often used in engines. Internal cooling consists in the fact that an excess amount of one of the components is fed to the layer of gas at the wall (the wall layer). Therefore, the temperature in the wall layer drops, resulting in cooling of the walls. The excess component can be fed either by special injectors on the head, through a system of openings in the wall, or through the pores of the porous material of the chamber walls.

The use of internal chamber cooling considerably simplifies the problem of creating reliably operating systems of external cooling; therefore, these two cooling systems are often used jointly in the same engine.

With intense internal cooling and additional protection of the inner walls of the chamber by using low heat conducting lubricants, or when heat resistant materials are used in the construction of the chamber, it is generally possible to dispense with external cooling. In this case the engines are called, not quite accurately, <u>uncooled</u> engines.

The type of LFRE and its design features are determined, to a great degree, by the purpose for which the engine is intended. We have the following categories: space rocket engines, engines for satellite carrier rockets, the engines for geophysical rockets, LRBM's ("surface-to-surface"), AA unguided and guided rockets ("surface-to-air"), airplane LFRE's (ordinary and climb boost), various types of boosters, and others.

The engines for space rockets should have high absolute thrust, very high specific thrust, and be relatively light.

Artificial earth satellite carrier rockets, large geophysical rockets, and extralong-range ballistic missiles should have the same engine characteristics.

Propellants with the maximum possible chemical energy are used in the engines of rockets of this type to attain high specific thrust.

In order that the rocket have minimum terminal weight (small relative to the terminal weight μ_t), the engine should be rather light. An increase in the range of rockets requires a considerable increase in the absolute thrust with a simultaneous increase in specific thrust and a decrease in the relative weight of the engine.

We do not have accurate data on the latest engines for long-range missiles, since the development of LFRE's is associated, in the majority of cases, with the development of new types of armament. However, available information on test stands under construction allows us to assume that engines with a single-chamber thrust of 80-100 tons have been built, or will be in the near future. We can also assume that the thrust of a propulsion unit can be increased by using several engines together. Therefore, on the basis of chambers with a thrust of 50-100 tons, we can create engines with a thrust of several hundred tons.

A very important application of LFRE's is AA guided rockets. Such rockets have engines with thrust from 2 to 15 tons and an operational time of 50 seconds. They usually employ a pressure feed system, but may also have pumps.

Liquid-fuel rocket engines for AA unguided rockets and short-range rockets (ranges to 100 km) have approximately the same parameters: thrust from 1 to 5 tons and operational time from 10 to 20 seconds. They must be simple to produce, since they are used for massed fire.

One of the requirements placed on AA rockets is that they must be able to be stored for as long a time as possible, in a fueled condition. This creates problems in selection of the fuel, the materials, and the very design of the rocket.

Liquid-fuel rocket engines can also be used as airplane engines: the main engines and climb boosters.

The main aircraft LFRE's are relatively complex in structure, allow regulation of the thrust within broad limits, e.g., from 1000 to 10,000 kg, have turbo-pump component feed, and have a system for repeated starting. The climb boosters also have pump feed, but are driven by the main engine; they should be capable of being started repeatedly in flight.

Finally, the last group of rocket engines includes the boosters for various vehicles: heavy planes, vehicles with ramjet engines, self-propelled missiles, etc.

The engines of such boosters, as a rule, can be used again after parachute recovery and after the components have been recharged.

Despite the fact that rocketry has developed intensely only in the past twenty years, there are already numerous types of rocket engines, differing from one another in the values of the developed thrust, the fuels used, the feed systems, and the design of the basic units. Many of these engines are in mass production and are currently being used.

Let us turn now to an examination of the structure of some specific engines.

The Construction of a Liquid-Fuel Rocket Engine for a Long-Range Rocket

One of the most complex and ideal types of LFRE's is that used in LRBM's. Let us examine the structure of the V-2 (A-4) rocket engine as an example.

This engine operates on a propellant consisting of LOX (oxidizer) and 75% aqueous ethanol (fuel). The engine has a thrust of about 26 tons (on the ground) with a fuel consumption of approximately 125 kg/sec. The specific thrust developed by the engine is 208 kg-sec/kg, and the exhaust velocity of the gases from the nozzle is greater than 2000 m/sec.

Figure 2.15 showed the general view of the engine installed in the rocket, while Fig. 3.8 shows the engine alone. Figure 3.9 shows the main diagram of the engine and its basic parts.

As these figures show, the main units of the engine are the pear-shaped combustion chamber 4 and the nozzle 1.

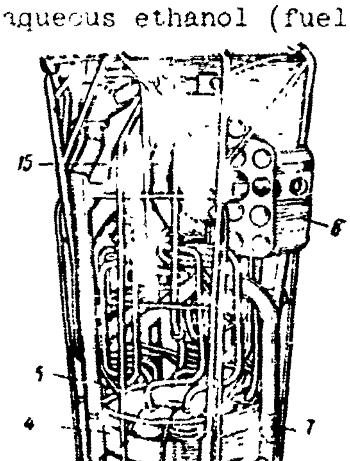

Fig. 3.8. Cutaway view of the V-2 rocket engine. The numbers correspond to those in Fig. 3.9.

When the engine is in operation, there is a pressure of about 15 kg/cm^2 in the combustion chamber. The combustion products are expanded in the engine nozzle to an exhaust pressure of 0.8 kg/cm^2, and acquire great velocity.

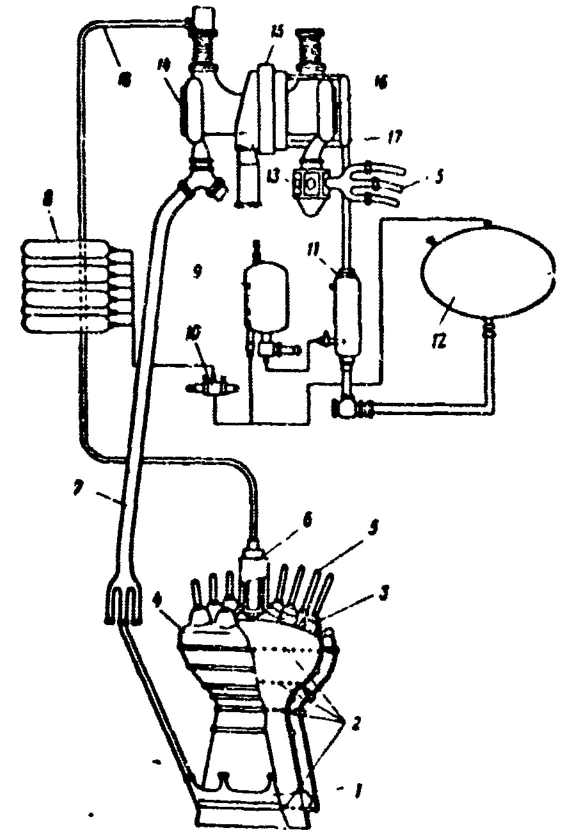

Fig. 3.9. Schematic of the V-2 engine. 1) nozzle; 2) film cooling rings; 3) precombustion chamber; 4) combustion chamber; 5) LOX feed lines to precombustion chambers; 6) main fuel valve; 7) fuel feed to cooling jacket; 8) high-pressure tanks; 9) catalyst tank; 10) air pressure reducer; 11) reactor; 12) hydrogen peroxide tank; 13) main oxidizer valve; 14) fuel pump; 15) turbine; 16) oxidizer pump; 17) turbine steam feed; 18) fuel line to pump when engine is turned off.

The maximum diameter of the combustion chamber is 930 mm, the diameter of the nozzle throat is 400 mm, and that of the exhaust area is 730 mm. The chamber weighs 420 kg.

The base of the combustion chamber has eighteen identical precombustion chambers 3. Each precombustion chamber serves as a place for the installation of the injectors through which the fuel and the oxidizer enter the combustion chamber in finely atomized form. The openings which atomize the oxidizer are located in the injector head, fed with oxidizer through pipes 5. The fuel injectors are on the side walls of the precombustion chamber. The fuel enters the injectors from the upper head chamber; it enters this chamber from the cooling jacket, passing through main fuel valve 6.

The engine is cooled as follows. Most of the ethanol, before entering the precombustion chambers, passes into the cooling chamber through pipe 7; the cooling chamber is formed by the double walls of the combustion chamber and the nozzle. Moving at great speed through this cooling chamber, the ethanol removes the heat from the inner walls of the chamber and nozzle, thus cooling them (external cooling system). Part of the ethanol passes, through special pipes, from the precombustion chamber, is sprayed through system of openings 2 onto the inner ("gas") surface of the wall of the engine and, evaporating from it, creates a vapor mist (internal cooling system).

The propellant in the engine is compressed and fed by centrifugal oxidizer (16) and fuel (14) pumps. Steam turbine 15 drives the pumps. The turbine and two pumps, connected by a common shaft, form the turbo-pump assembly (Fig. 3.7). The turbo-pump is 4[illegible]5 hp and weighs [illegible] kg.

Pipe 18, which connects main fuel valve 6 and the pump, is necessary to displace the air when fueling the system, and to prevent hydraulic shock in the cooling jacket when the engine is turned off. The oxidizer, fed by pump 16, passes through main oxidizer valve 13, whence it enters the precombustion chambers through pipes 5.

The steam to work the turbine is generated in the steam generator by the decomposition of concentrated hydrogen peroxide H_2O_2. When hydrogen peroxide decomposes, enough heat is generated to heat the combustion products to 400-500°C (depending on the dilution of the hydrogen peroxide by water). Hydrogen peroxide decomposes rapidly and dependably in the presence of a catalyst, e.g., a concentrated solution of sodium permanganate $NaMnO_4$.

The steam generator consists of the hydrogen peroxide tank 12 and the tank with the permanganate solution 9. These substances are fed to reactor 11 by displacing them with compressed air. The compressed air supply is contained in tanks 8. Before entering the steam generator tanks the air pressure is reduced in reducer 10.

The steam generator generates 2 kg of steam per second, sufficient to drive the turbine. The steam generated in reactor 11 is fed to turbine 15 through pipes 17.

The generator and its components weigh 148 kg.

The thrust developed by the engine is transmitted to the rocket body by the thrust frame, which weighs 56 kg. Thus, the entire engine weighs 930 kg, i.e., less than 40 g per kilogram of developed thrust.

The engine is started, operates, and is stopped as follows. Before the rocket is launched, after all its apparatus has been checked on the launching pad, it is filled with the necessary fuel, oxidizer, hydrogen peroxide, sodium permanganate solution, and compressed air.

At the starting moment the explosive charges on a revolving fixture in the combustion chamber are ignited. The explosive charges are positioned such that the reactive force of the emerging gases causes the fixture to revolve. Because of this revolution the incandescent combustion products of the explosive charges fill the thrust chamber and heat it. When the required heating has been attained (2-3 sec after start) a magnesium strip located in the coldest part of the chamber begins to burn through. This is a signal for the starting to continue. By means of the electrical-pneumatic system main fuel valve 6 and main oxidizer valve 13 begin to open slightly. The ethanol and oxygen begin to enter the combustion chamber in relatively small amounts merely by means of the pressure from the columns of appropriate fluid (gravity flow). The propellant entering the engine is ignited by the hot powder gas and combustion becomes more intense. Stable combustion is attained in 2-3 sec, and then the propellant feed can be increased. For this purpose, the electropneumatic system turns on the steam generator and turbo-pump assembly. Air from tanks 5 begins to enter pressure reducer 10 and tanks 12 and 9, displacing the hydrogen peroxide and the permanganate into reactor 11. The steam thus formed starts to turn the turbine. The pressure developed by the pumps gradually opens the main fuel and oxidizer valves to full flow. In 2-3 sec the propellant flow rate reaches a nominal value of 125 kg/sec. By this time the rocket has already begun its flight.

In flight, the fuel and oxidizer tanks must have pressure sufficient for reliable operation of the pumps. For this purpose, the fuel tank is fed air from the atmosphere, with a pressure equal to the dynamic pressure, while a small amount of oxygen, evaporated in the heat exchanger by the turbine exhaust gases, enters the oxidizer tank.

The engine is shut down in two stages. First, as the electropneumatic system operates there is a decrease in the amount of hydrogen peroxide which enters the reactor. At the same time, less steam enters the turbo-pump assembly, and the rpm of the turbine decrease. The flow rate of the fuel fed by the pump decreases and the engine thrust decreases to about 30% of nominal. The engine converts, as it were, to the terminal stage. In this stage it operates until the rocket has attained the required flight speed (see Chapter IX). At this moment the hydrogen peroxide ceases to be fed to the reactor and the main fuel and oxidizer valves close. The engine has shut down.

A Liquid-Fuel Rocket Engine with Pressurized Feed

We have examined the construction and operation of a liquid-fuel rocket engine with turbo-pump feed. Let us now examine the construction and operation of an engine having pressurized feed.

Figure 5.1[illegible] shows a diagram of an engine designed for an AA guided rocket.

The engine operates on hypergolic components. The engine thrust is 8 tons and the fuel consumption is 42 kg/sec. The engine operates as follows. At start, [illegible] charge [illegible] is exploded. Piston 4, under the influence of the powder gases, [illegible] diaphragm [illegible], as a result of which the air from high-pressure tank 1 enters [illegible] where the pressure is reduced to operational. At the same time [illegible]

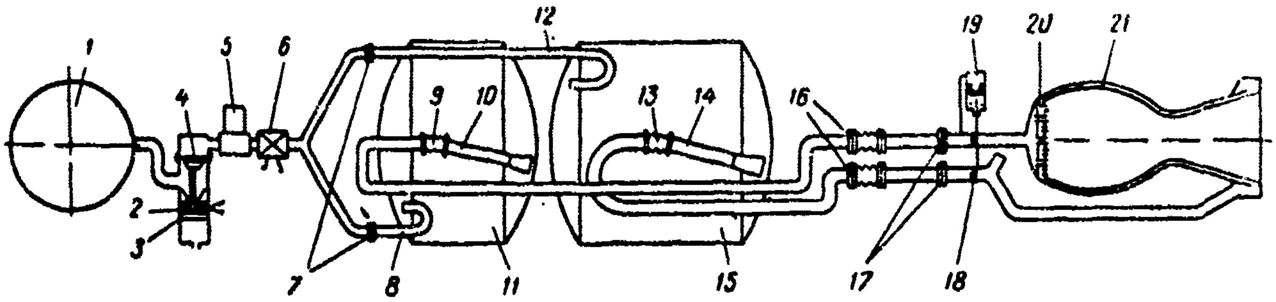

Fig. 3.10. Liquid-fuel rocket engine with pressurized feed.
1) compressed air tank; 2) powder charge; 3) piston; 4) high-pressure diaphragm-valve; 5) gas-pressure reducer; 6) low-pressure pyrotechnic valve; 7) burst diaphragms; 8) air-feed pipe to fuel tank; 9) flexible joint on fuel intake; 10) fuel intake; 11) fuel tank; 12) air-feed pipe to oxidizer tank; 13) flexible joint on oxidizer intake; 14) oxidizer intake; 15) oxidizer tank; 16) bellows box; 17) fuel and oxidizer line diaphragms; 18) throttle shutters; 19) servo-piston for controlling the throttle shutters; 20) head; 21) combustion chamber.

diaphragm 4 bursts, low-pressure pyrotechnic valve 6 opens. The air, having burst diaphragm 7, enters the fuel and oxidizer tanks 11 and 15 through pipes 8 and 12. The propellant is taken in by intakes 10 and 14, suspended on flexible joints 9 and 13 so that the intakes can adjust themselves to the level of the fluid as the rocket climbs.

Under air pressure, the propellant bursts diaphragm 17 in the component pipelines and it begins to enter the engine. To assure smooth starting, the pipelines contain throttle shutters 18. When the engine is started these shutters are closed. After diaphragm 17 bursts, the fuel enters the cylinder of servo-piston 19 which gradually moves, under the pressure of the fuel, opening shutters 18. This assures smooth increase of the fuel feed and smooth starts. From then on the shutters remain open.

The fuel enters the chamber head directly, while the oxidizer passes through the cooling jacket. The fuel and the oxidizer mix, ignite spontaneously, and burn in combustion chamber 21.

The start is much faster than in the above-described V-2; this is necessary for AA rockets. The arrangement of an engine with pressurized feed is also simpler than that of an engine with pump feed.

Aircraft Liquid-Fuel Rocket Engines

In conclusion, let us examine the LFRE of a fighter-interceptor. This engine (Fig. 3.11) is interesting in that its thrust can be controlled in flight by varying the fuel flow-rate within broad limits, from 200 to 1500 kg.

The engine operates on hypergolic components. The fuel is a mixture of hydrocarbons – hydrazine hydrate and methanol – and its oxidizer is hydrogen peroxide.

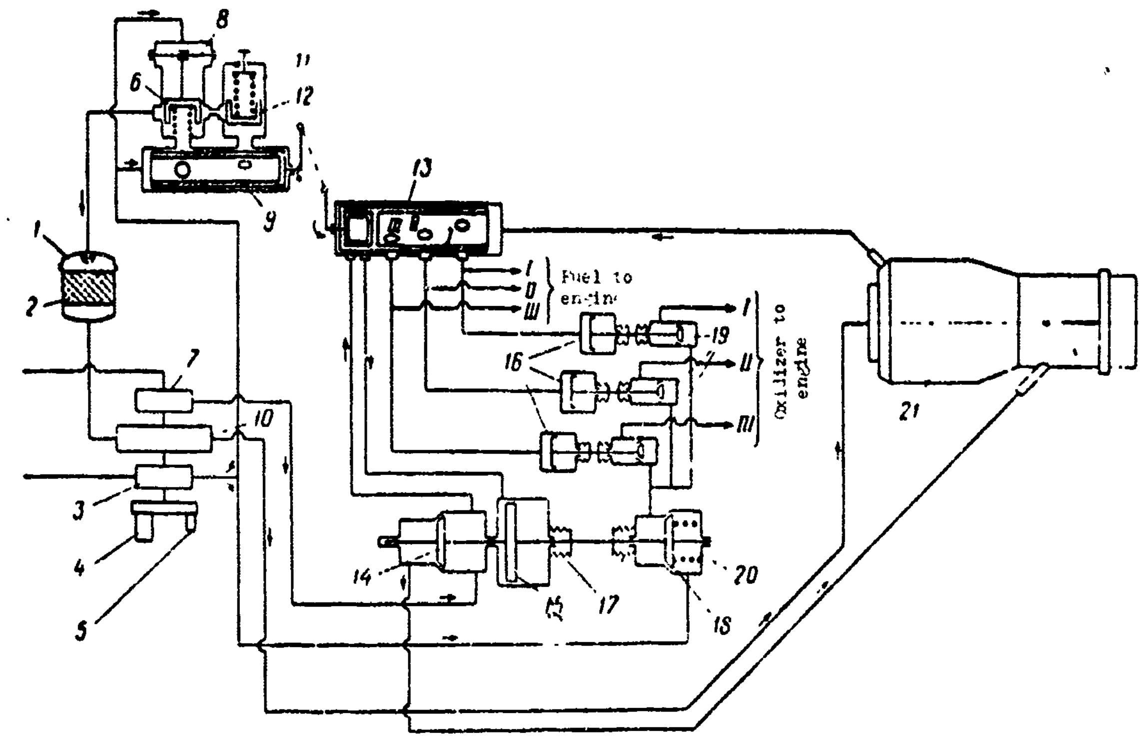

Fig. 3.11. Diagram of the liquid-fuel rocket engine of an interceptor.

1) steam reactor; 2) solid catalyst; 3) hydrogen peroxide pump; 4) electric motor; 5) tachometer transducer; 6) valve for peroxide flow control; 7) fuel pump; 8) diaphragm; 9) peroxide flow control slide valve; 10) turbine; 11) turbo-pump assembly idler control spring; 12) idler control valve; 13) fuel feed control slide valve; 14) fuel valve; 15) servo-piston for main fuel and oxidizer valves; 16) servo-pistons for hydrogen peroxide valves; 17) bellows boxes; 18) main hydrogen peroxide valve; 19) hydrogen peroxide feed valves; 20) chamber for feeding oxidizer to dual main propellant valves; 21) combustion chamber.

The engine has three groups of injectors which are turned on and off in succession when it is necessary to increase or decrease the thrust. Thus the engine has multi-stage-controlled thrust.

The turbo-pump assembly (including hydrogen-peroxide pump 3) is started by electric motor 4. The hydrogen peroxide is pumped to the peroxide flow regulator and, when slide valve 9 is in the appropriate position, raises control valve 12, passes through the bypass channel around valve 6, and enters reactor 1. This engine differs from that of the V-2 in that it uses a solid catalyst 2 which is placed in reactor 1 in packet form. The steam developed in the reactor continues to accelerate the turbine; when the turbo-pump assembly reaches a definite number of rpm the electric motor is disengaged from the shaft by means of a free-play clutch, and it stops. The turbo-pump then idles. The idling rpm is determined by the tension of spring 11 of regulator 12, which changes the flow of peroxide to the reactor.

During idling, the hydrogen peroxide pump 3 transmits pressure to the chamber of dual main fuel valve 20, as a result of which the hydrogen peroxide valve 18 and fuel valve 14 close. This prevents the propellant from entering the combustion chamber. To start the engine, the pilot turns slide valve 9, which also turns slide valve 13 which is welded to it. Slide valve 13 allows the propellant to enter the cylinder of servo-piston 15 through a window in the left-hand chamber. The piston moves to the right, allowing the peroxide to enter three valves 19. In this case the fuel can pass through the cooling jacket of the engine and enter the inner chamber of slide valve 13.

As slide valve 13 is turned, opening I allows fuel to enter fuel-feed line I. At the same time, fuel passes through line I to servo-piston 16 of oxidizer feed valve 19, opening it; hydrogen peroxide passes into peroxide-feed line I. Thus, injector group I begins to operate, spontaneous combustion of the propellants occurs in the chamber, and the engine begins to operate on regime I.

At the same time, because of the turning of slide valve 9, the hydrogen peroxide passes to valve 6 through the appropriate openings; lifting the valve, it enters steam generator reactor 1 in large quantities. As can be seen from Fig. 3.11, the peroxide pressure, depending on the rpm of the turbo-pump assembly, is fed to diaphragm 8 which, bending under pressure, tends to close valve 6 and thus decrease the peroxide flow. This prevents excess flow of peroxide and over-speed of the turbo-pump assembly.

To further increase the thrust it is necessary to continue to turn slide valve 13; this allows the fuel and oxidizer to be fed to injector groups II and III through orifices II and III. During this the peroxide flow control maintains the turbo-pump rpm such that the pressure of the propellant feed remains constant.

The engine is shut down by turning the slide valves in the opposite direction; at the same time, the turbo-pump, during flight, may not be shut down, but can continue to idle.

To prevent the accumulation of propellant components in the combustion chamber due to leakage in the feed system during idling, a slight amount of the steam developed in the turbine is fed to the thrust chamber to decompose the peroxide which has leaked there, and to blow out the thrust chamber.

The dual main propellant valve, slide valve 13, and the three oxidizer valves (I, II, III) are contained in one unit. To avoid the possibility of the propellant components' coming in contact and combusting, all coupling rods from the chamber of one component to the chamber of another are hermetically sealed by bellows 17.

We have examined in brief the construction and operation of only three of the various LFRE's. Modern rocketry utilizes many more types of rocket engines, differing in the propellants on which they operate, their feed systems, the design of the chambers, and the cooling methods. However, it is unnecessary to examine all these types of engines, since the three which we have considered are quite typical, and using them as a basis we can get an idea of the main points of the construction of any LFRE. The construction and operation of the basic unit of the rocket engine (the chamber) will be described in more detail in the following chapters.

In conclusion we should mention that all LFRE's have a characteristically high degree of automation in the starting, operation, and stopping of the engine.

The engine of an AA rocket is started automatically after the electrical signal is fed to powder charge 2 and pyrotechnic valve e (Fig. 3.10). True, this engine does not stop, but operates until all components in the tanks are expended. In an airplane engine, all pilot operations are reduced to a minimum. We have not examined in detail the sequence of operations for the starting of long-range rocket engines. However, in this case as well, only two or three commands are given. All further operations of the rather complex pneumatic-hydraulic apparatus are carried out automatically.

The trend toward maximum automation of engine operation is explained by the basic feature of rocket engines, in particular LPRE's, viz., their brief time of operation — from several seconds to several minutes. In this short time it is necessary to perform all the required operations for reliable fuel combustion, increasing the feed to the nominal flow-rate, keeping this rate constant or changing it in accordance with the operational program of the engine, and, finally, in time the engine must be turned off precisely at the given moment.

We must not forget that the propellant which enters the chamber in combustible form is a typical explosive. Considering the high flow-rate of the propellant per second we can clearly see that the slightest disruption in the operation of the feed system or a delay in the ignition time, resulting in the accumulation of propellant in the chamber, can result in an explosion or lead to considerable variations and increases in pressure which can also lead to destruction of the engine. The same holds true for a chance disruption in combustion or if propellant continues to be fed after the engine has been shut down. In this case, the ignition of the propellant from the incandescent parts of the engine can cause [illegible] combustion and a subsequent explosion.

From what has been said it is clear that the feed system must operate very reliably. This is assured by overall automation and the use of complex multiple interlocking in the design of the feed system. (By interlocking we mean the creation of conditions such that a particular operation in the engine feed system cannot be performed until the previous one has been accomplished.) The automation of the feed systems in present-day LFRE's has reached such a high degree of perfection that all operations in the starting, operating, and stopping of an engine can be performed after only one command signal has been fed to it.

Clearly, such a high degree of automation is also required since LFRE's are installed, for the most part, in pilotless vehicles.

The Basic Units in Liquid-Fuel Rocket Engines

The development and improvement of liquid-fuel rocket engines has required the creation of a great many auxiliary units. The pressure reducer, which plays a large part in the feed system of LFRE's, is one of the most important regulators which keeps the operational regime of the engine constant.

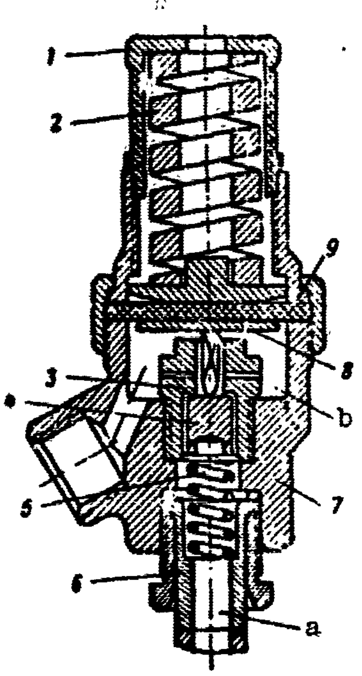

Fig. 3.12. Pressure reducer. a) high-pressure chamber; b) low-pressure chamber. 1) cap; 2) main spring; 3) valve seat; 4) valve; 5) spring; 6) intake; 7) reducer housing; 8) head; 9) diaphragm.

Figure 3.12 gives a cross section of the simplest type of pressure reducer. High-pressure air from the tanks enters the reducer housing through intake 6. Air enters the throttle section of the reducer through the slit between seat housing 3 and reducer valve 4; this section is formed by the gap between the valve seat and the valve itself.

After being throttled in this gap the gas enters low-pressure chamber b and then the exhaust line.

The degree of pressure drop is determined by the degree to which the reducer valve is opened. The further away from the seat the valve is, the smaller will be the pressure sink. As the reducer operates, the valve moves under the influence of applied forces. These forces are as follows: a) the force on the valve of the pressure of the gas entering the reducer; b) the force of the spring 5; c) the force on the valve of the pressure of the gas passing through the slit, d) the force of main spring 2 of the reducer which acts on the valve through diaphragm 9 and valve head 8. The first two of these forces tend to close the valve and increase the throttling of the gas, and the second two forces act in the opposite direction. The force of spring 2 is opposed by the force of the pressure of the gas emerging from the reducer and acting on diaphragm 9. This force decreases the valve opening and increases throttling.

The gas-pressure reducer is a control device. For example, if the gas pressure at the reducer output decreases, the pressure on the diaphragm also decreases, and the spring causes the valve to open somewhat. Because of this, the throttling decreases and the pressure at the reducer output increases. Cap 1 serves to control the tension on the main spring of the reducer. By increasing the tension on this spring we open the valve to a great degree and the pressure at the output will increase. Spring 5 has an auxiliary function. When the tension is released on spring 2, the valve closes and the reducer operates as an ordinary valve.

The valves are the basic elements of the pneumatic-hydraulic automatic control system for the engine. They can be as different as their purpose warrants. Electromagnetic valves (Fig. 3.13) are used with small flow-rates of the liquid or gas. With high flow-rates, when considerable forces are required, the valve is opened by air pressure from the small control valve. In this case we use a valve with a pneumatic or hydraulic servo-piston (Fig. 3.14). Figures 3.15 and 3.16 show two types of check valves; their operation is explained beneath the figures.

The feed systems of LFRE's often utilize

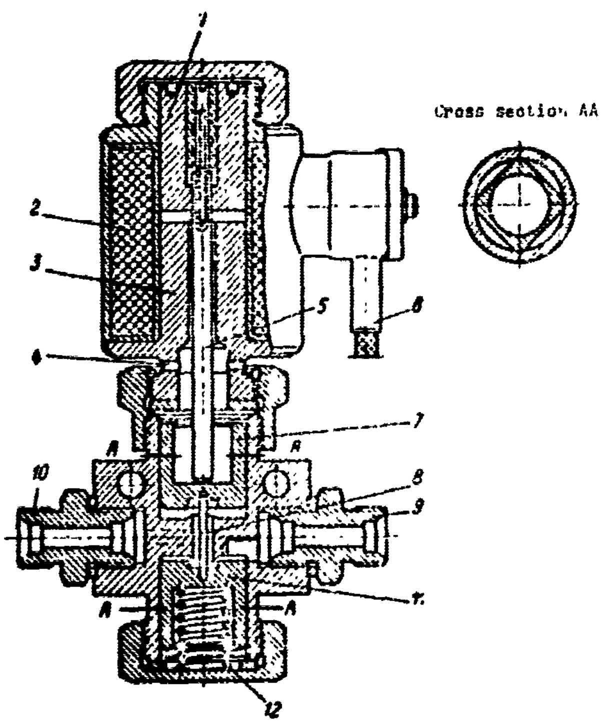

Fig. 3.13. Electromagnetic pneumatic control valve. [illegible]

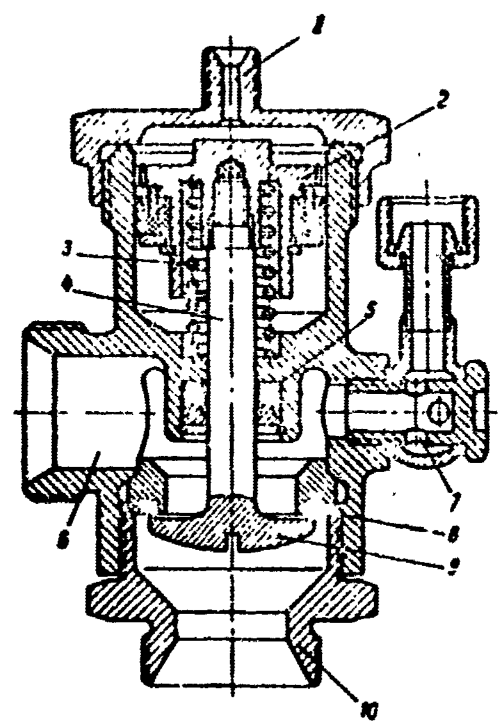

Fig. 3.14. Auxiliary valve with pneumatic servo-piston. [illegible]

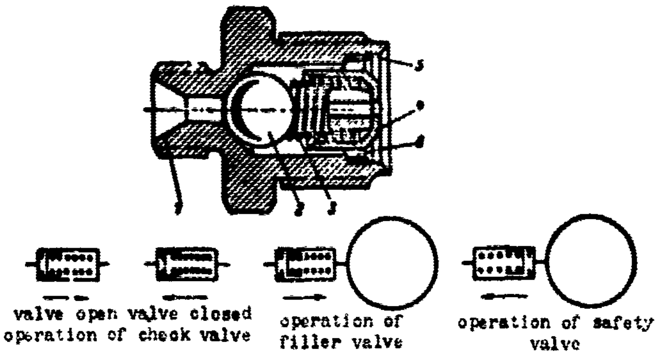

Fig. 3.15. Check valve, and diagrams of the operation of check, filler and safety valves. 1) intake pipe; 2) ball seal; 3) spring; 4) spring support and guide bushings; 5) check ring; 6) openings for passage of the fluid.

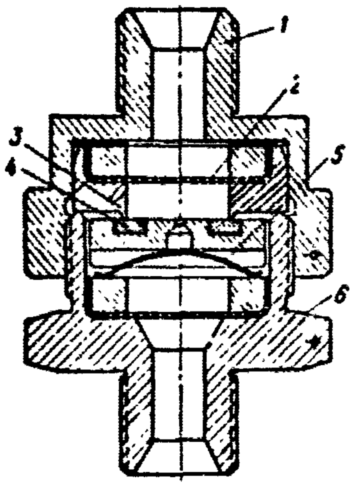

Fig. 3.16. Check valve with plate spring. 1) intake pipe; 2) filter; 3) valve seat; 4) valve shoulder seal; 5) plate spring; 6) exhaust pipe.

the simplest types of burst diaphragms, which burst when the pressure exceeds a certain limit (Fig. 3.17). After bursting, the tabs of the diaphragm fold back and allow the liquid to enter the pipe. If the diaphragm is designed to burst after a certain length of time, instead of by pressure, it is force-burst using a special explosive charge (Fig. 3.18). The diaphragms are single-direction, one-time valves, used only as openers. Sometimes they are equipped with pyrotechnic valves which make it possible to close the pipeline at the required time.

As a pressure sensor, the pneumatic-hydraulic system of LFRE's uses, for the most part, diaphragm devices with electrical contacts, similar to that shown in Fig. 3.19.

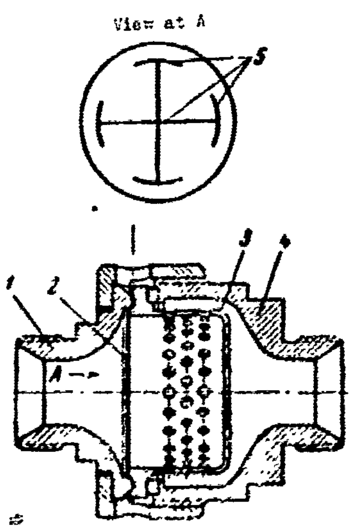

Fig. 3.17. Automatic burst diaphragm. 1) intake pipe; 2) burst diaphragm; 3) filter; 4) exhaust pipe; 5) incisions along which the diaphragm tabs burst and fold back.

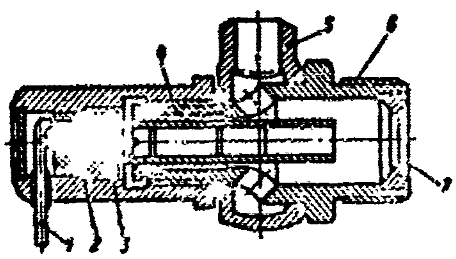

Fig. 3.18. Forced burst diaphragm with explosive charge. 1) electric current to explosive charge; 2) explosive charge burster; 3) ignition charge; 4) pipe to conduct the explosive wave to the diaphragm; 5) intake pipe; 6) exhaust pipe; 7) forced-burst diaphragm.

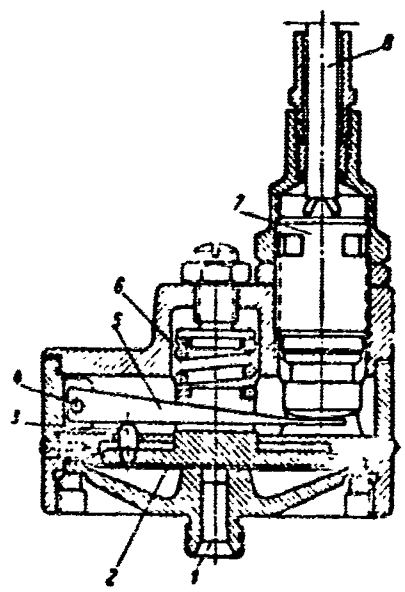

Fig. 3.19. Pressure relay. 1) pressure-feed line; 2) diaphragm which collapses under pressure; 3) pin to transmit the motion of the diaphragm when it collapses; 4) rockshaft; 5) lever to transmit movement to the electric switch; 6) spring acting on the lever; 7) electric switch; 8) electrical outlet.

C H A P T E R IV

ROCKET ENGINE PROPELLANTS

1. The Chemical Energy of Rocket Engine Propellants

Power Sources for Rocket Engines

A rocket engine, just as any other engine, requires some type of energy supply in order to operate.

The energy source widely used at present in rocket engines is chemical energy. As the engine operates, this energy can be released in reactions of various types. The most widespread reaction is combustion. This reaction is used in the operation of the majority of existing heat engines.

Propellants are also used whose chemical energy is evolved by decomposition. Such decomposition is accompanied by the release of heat. The decomposition of hydrogen peroxide is widely used in rocket engines.

If decomposition causes the release of a free oxidizing or fuel element, such a reaction can be combined with the subsequent normal combustion of this element, occurring due to the additional feed of the missing component into the chamber.

We can also conceive of using propellants which generate energy by recombination, e.g., the recombination of atomic substances into molecular ones. As a rule, recombination occurs with the release of great amounts of energy. As with decomposition, recombination can be combined with ordinary combustion.

A rocket engine requires not only a power source but also a mass which the engine repels as it operates. The substance expelled from the rocket is usually called the mass carrier. In present-day engines the propellant is the source of chemical energy, and the combustion or decomposition products are the mass carrier.

Thus, the propellant is first a chemical-energy carrier and then a kinetic-energy carrier.

Nuclear power is a potential source of energy for rocket engines. It is not used at present, but it undoubtedly will be in the near future. Evidently, nuclear propellants will be used only as a power source, while the mass carrier will have to be carried on board the rocket. Sometimes the atomic propellant is called the active mass, and the mass carrier is called the passive mass.

Combustion and Chemical Energy

By combustion we mean the chemical process between two substances, the oxidizer and the fuel, accompanied by the release of great quantities of heat.

The chemical process of combustion can consist of many diverse reactions, but the basic reaction of this process, from an energy standpoint, will be the reaction of the combination (combustion) of fuel and oxidizer atoms. Let us examine this reaction.

The energy level of atoms is determined by the structure of the outer electron shell of the atom. When the outer level is filled, it contains a definite number of electrons, eight in the case of the elements usually used for rocket engine propellants. Hydrogen, whose outer shell contains two electrons, is an exception.

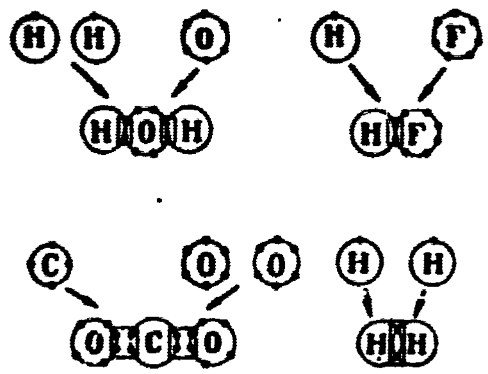

Fig. 4.1. Diagram of the chemical combustion and recombination of elementary propellants. The dots indicate the electrons of the outer shells. The electrons in the rectangles belong to two atoms simultaneously.

The filling of the outer shell with electrons is accompanied by a decrease in the energy level of the atom. An atom has minimum energy level when the outer shell is filled with electrons.

Examining the structure of the electron shells of fuel and oxidizer atoms, we see that their outer shells are not filled. The hydrogen atom (Fig. 4.1) is missing one (of two) electrons, the carbon atom – four, the oxygen atom – two, and the fluorine atom – one (of a possible eight).

During the chemical process of combustion there is no change in the structure of the atom, and the total number of electrons in fuel and oxidizer atoms remains the same. However, when combustion products form, there is a mutual interweaving and transformation of the electron shells of the reacting atoms, so that some of the electrons can simultaneously enter the shells of both fuel and oxidizer atoms.

For example, the formation of water vapor, as a result of the combustion of hydrogen and oxygen, is accompanied by the transformation of electron shells; two groups of two electrons each enter the oxygen and hydrogen atom shells simultaneously. In this case the oxygen atom, becoming a water molecule, will have eight electrons in its outer shell, while the hydrogen atoms will each have two electrons, i.e., the shells of all atoms are filled with electrons.

The same is true of molecules of hydrogen fluoride (HF), carbon dioxide (CO_2), and molecular hydrogen (H_2) (Fig. 4.1); the common possession of electrons results in the fact that the outer electron shells of all atoms in the reaction product are filled with electrons. Therefore, the energy level of a molecule of a reaction product will be lower than that of the atoms before they entered into the reaction. The difference between these energy levels is what determines the amount of chemical energy produced during a given reaction.

Since the energy levels of the initial substances and the combustion products depend solely on the atomic structure of the corresponding substances, the amount of chemical energy produced by the combustion of a known mass of the given substances is always constant, independent of external factors — pressure, temperature, volume, etc.

We have examined the combustion of atomic substances. Actually, however, the fuels and oxidizers used in rocket engines are not in the atomic state, but make up the composition of molecules of simple or, most often, complex substances.

In this case, combustion, from the viewpoint of energy yield, can occur as follows. A substance in molecular form decomposes into atoms of the corresponding elements with the expenditure or generation of chemical energy. Then the atoms combine into combustion products, always with the release of chemical energy. The overall effect of the chemical energy yield is determined by the algebraic sum of the chemical energy corresponding to each of the constituent processes.

Chemical Energy and the Heat of Formation

The chemical energy evolved or expended during the formation of substances, including combustion, can be determined in two ways.

The first method consists of using the data of the spectral analysis of particles (atoms and molecules) to compute the energy levels of these particles before and after the reaction; the amount of chemical energy corresponding to the given transformation of the substance is determined from the difference.

The second method consists in the calorimetric determination of the heats of formation (also called the heats of combustion) of the substance. In this case the required reaction is carried out in a special container (calorimeter); this reaction results in the chemical energy's being transformed into heat used to warm the reaction products. Upon completion of combustion the reaction products are cooled to the temperature of the initial substances before the reaction; the amount of heat removed from the reaction products is measured. The amount of heat thus determined is called the heat of formation of the substance, as opposed to the chemical energy found by the first method.

The heat of formation is in definite correspondence with, but not equal to, the chemical energy.

The difference between the heat of formation ΔH and the chemical energy Q_{ce} occurs because combustion and the corresponding change in state of the substance (the structure of its molecules) are accompanied by a change in the energy reserve which the substance had before and after the reaction, at uniform temperature T.

If the initial and final products are at the same pressure, the difference ΔQ_T of their energy reserve at temperature T equals the change in heat content of the substance* which occurs as the result of the reaction (computed per unit weight)

$$\Delta Q_T = \int_0^T c_{p\ rp}\, dT - \int_0^T c_{p\ ip}\, dT \qquad (4.1)$$

or, if we consider the heat capacities constant,

$$\Delta Q_T = (c_{p\ rp} - c_{p\ ip})T, \qquad (4.2)$$

where $c_{p\ rp}$ is the heat capacity with constant pressure of the reaction products;

$c_{p\ ip}$ is the heat capacity with constant pressure of the initial products.

If during the reaction the pressure is not constant, but, e.g., the volume remains unchanged, the value of ΔQ_T will be different, since the heat capacity of gaseous substances depends on the conditions of their heating or cooling. However, absolute temperature T exerts the most influence on ΔQ_T.

The heat of formation, used in calculating rocket engines, usually has the following conditions: constant pressure, equal to one absolute atmosphere, and normal temperature. In this case the heat of formation is considered as the change in the value of the heat content H of the combustion products, and is designated

* The concept of the "heat capacity of a substance" will be explained in Chapter V.

by ΔH_T, where T is the absolute temperature at which the heat of formation is determined.

Knowing the heat of formation and the heat capacity of the substances participating in the reaction, we can always compute the chemical energy as the difference of the algebraic values

$$Q_{ce} = \Delta H_T - \Delta Q_T.$$

Since when $T = 0$, $\Delta Q_T = 0$, the chemical energy can be defined as the heat of formation at absolute zero, and designated as ΔH_0. This value serves as a measure of the chemical energy, and is widely used in many thermodynamic computations.

In light of the fact that the heat of formation of the combustion products depends on external conditions, we use the so-called standard conditions in determining it. For the most part, the heat of formation of substances is determined at temperatures of 18, 20, or 25°C (291.16, 293.16, or 298.16°K) and a pressure of one atmosphere. In this event, the difference between the chemical energy and the heat of formation of the corresponding products is relatively slight.

Since the heat of formation of a substance depends on the form (atomic, or simple or complex substances) of the elements which make up the given substance, the heat of formation always pertains not only to standard conditions, but also to the standard state of the elements. By this we mean the state of the element as it is usually found in nature. We assume the following as the standard states of the most often used fuels and oxidizers: H_2, O_2, and N_2 molecular gases for hydrogen, oxygen, and nitrogen; β-graphite for carbon; and the crystalline form for metals (the form in which they are most often found in the natural state). The sign of the heat of formation is determined by the fact that the energy lost by the system participating in the reaction is considered negative; hence the heat of formation of combustion products is always negative.

The heat of formation is determined under standard conditions and pertains to the standard state of the elements; in designating this we add the superscript "0." The standard heat of formation of a substance is designated by ΔH_T^0, while the standard chemical energy is ΔH_0^0.

The Calorific Value of a Propellant

By calorific value of a propellant we mean the amount of heat liberated with complete combustion of a weight unit of the propellant. We designate this by K_G, measured in kcal/kg.

The calorific value of a propellant equals the heat of formation of the products of complete combustion of the substances comprising the propellant, taken with the opposite sign, since the chemical energy of a substance lost during the formation of combustion products is completely transformed into liberated heat, according to the law of conservation of energy. Thus,

$$K_G = -\Delta H_T.$$

The heat of formation is usually given in kcal/g-mole of the combustion products, but it is easily converted into the amount of heat per unit weight by using the relationship

$$\Delta H \text{ kcal/kg} = \frac{1000}{\mu} \Delta H \text{ kcal/g-mole},$$

where μ is the molecular weight of the combustion products.

We should point out that a rocket carries both fuel and oxidizer, and therefore these have equal value. We will, therefore, always relate the calorific value or the heat of formation to the entire mass of the substances expended in the reaction, not only to the fuel alone, as is done in ordinary heat engineering where the consumption of oxidizer (oxygen from the air) is never considered, since the oxygen is taken from the surrounding atmosphere.

Under standard conditions and with standard state of the elements, the calorific value is, obviously,

$$K_G = -\Delta H_T^0.$$

This value depends basically on the type of element participating in combustion. Henceforth we will call it simply the calorific value of standard elements.

As fuels and oxidizers in propellants we should use those which have maximum chemical energy. In the thermal calculations of rocket engines we often use the heat of formation ΔH_T^0 which, as has been shown, is close to the value ΔH_0^0, instead of the chemical energy, as a value to estimate the chemical energy of propellants.

Since there exists in nature only a limited number of chemical elements, in our selection of propellant components we must select those elements which have the greatest supply of chemical energy.

Tsiolkovsky, in one of his first works: "A Rocket into Space," indicated the need for the existence of a definite regularity in the values of the chemical energy of elements in conjunction with the periodic system. Actually, the heats of formation of combustion products for various elements are regularly distributed, forming periods which correspond to the series in the periodic table. More detailed investigations of this problem led to the following results.

Of all the elements, only two, oxygen and fluorine, can be used as oxidizers, giving a great supply of chemical energy, and assuring the necessary combustion intensity. Other oxidizers such as chlorine, bromine, and iodine do not have high chemical energy. The following elements produce the maximum effect as fuels: beryllium, lithium, boron, aluminum, magnesium, silicon, hydrogen, and carbon.

Rocket engines rarely use elements in their standard states as propellants. It is more convenient to use certain complex substances having more favorable physical properties.

However, the formation of substances (propellant components) from standard elements is accompanied, as a rule, by the absorption or liberation of heat, i.e., each propellant component and the fuel in general has its own standard heat of formation.

Considering the heat of formation, the calorific value of a propellant will be

$$K_G = \Delta H^0_{T\ prop} - \Delta H^0_{T\ cp}, \qquad (4.3)$$

where $\Delta H^0_{T\ prop}$ is the standard heat of formation of the propellant components;

$\Delta H^0_{T\ cp}$ is the standard heat of formation of the combustion products.

For example, if the standard heat of formation of a propellant substance is positive ($\Delta H^0_{T\ prop} > 0$), i.e., heat is expended on its formation from standard elements, the calorific value of the propellant also increases by a corresponding amount.

The use of atomic substances in rocket engines is based on just this idea of the use of positive heat of formation.

If the propellant substance has a negative standard heat of formation ($\Delta H^0_{T\ prop} < 0$), its calorific value will be less than that of the standard elements. Therefore, when analyzing propellant components it is necessary to consider their standard heat of formation.

Before examining in greater detail the properties and energy characteristics of various elements, let us give the basic factors to be used in evaluating rocket propellants and their components.

The Basic Requirements of Rocket Propellants

In order to formulate the basic requirements of rocket propellants, let us use the transformed Tsiolkovsky formula (1.14) for the ideal terminal velocity of a single-stage rocket:

$$v_t = w_e \ln \frac{M_t + M_{prop}}{M_t}.$$

Since the terminal velocity developed by a rocket is one of its basic properties, we can consider that the best rocket propellant will be one which assures maximum terminal velocity when used with a given payload weight and initial weight of the rocket. Therefore, the first basic requirement of a rocket propellant is, as we have seen from an analysis of (1.14), that as it burns in the rocket engine it produce a maximum effective exhaust velocity, i.e., maximum specific thrust.

On the other hand, the ratio $(M_t + M_{prop})/M_t$ will be the greater, the smaller the volume of the tanks necessary to carry a certain amount of propellant on the rocket, since the weight of the rocket and, consequently, the terminal mass M_t, depend to a considerable extent on the size of the tanks. In this case we must bear in mind that the propellant tanks take up 50-90% of the space in the rocket. Therefore, a second requirement of rocket propellants is that they have maximum specific weight γ_{prop}.

The specific weight of the propellant, consisting of fuel and oxidizer, is determined by the formula

$$\gamma_{prop} = \frac{1 + \nu}{\frac{1}{\gamma_f} + \frac{\nu}{\gamma_o}}, \tag{4.4}$$

where γ_f is the specific weight of the fuel;

γ_o is the specific weight of the oxidizer;

ν is the amount of oxidizer, in kg, per 1 kg of fuel.

Let us point out that

$$\nu = \alpha\nu_0,$$

where ν_0 is the theoretically required amount of oxidizer, determined from the ordinary material (weight) balance of chemical reactions when carried to completion;

α is the excess oxidizer factor — for LFRE's the factor α is usually somewhat less than unity.

Since for most present-day propellants of LFRE's $\nu > 1$ (ν = 2-5), the specific weight of the oxidizer has much more effect on the value of the specific weight of the propellant than does the specific weight of the fuel.

The need for high specific weight of the propellant is quite obvious and needs no further explanation.

It is much more difficult to explain what quantity of rocket propellant will assure maximum specific thrust.

Since the kinetic energy of a stream of combustion products is obtained due to the chemical energy (calorific value) of the propellant, the first condition for obtaining high specific thrust is maximum chemical energy in the propellant, or maximum calorific value. This is a necessary but, as it appears, far from sufficient condition, although for a long time it was the dominant condition when evaluating propellants.

Another very important condition for attaining high specific thrust is the high efficiency of the basic thermodynamic processes of the conversion of energy, i.e., combustion and expansion.

The efficiency of combustion and expansion depends on the design features of each rocket engine, and on certain properties of the combustion products, which we will call the thermodynamic properties. Thus, one of the conditions for attaining high specific thrust is the good thermodynamic properties of the combustion products.

Studying this problem in more detail, we see that combustion (the formation of combustion products) is also accompanied by the reverse process, dissociation (the decomposition of the products). Dissociation is accompanied by a considerable loss in energy. The degree of dissociation sharply decreases with a decrease in the temperature in the combustion chamber and with a decrease in the number of atoms in the molecules of the combustion products, since molecules of combustion products with a small number of atoms decompose with more difficulty, as a rule, than do molecules with a large number of atoms.

The temperature in the combustion chamber of a rocket engine is determined, in first approximation, by the relationship

$$T = \frac{K_G}{c_p}, \tag{4.5}$$

where c_p is the heat capacity per unit weight of the combustion products at constant pressure. Therefore, with uniform calorific value of the propellant, the temperature in the combustion chamber will be the lower, the greater the heat capacity per unit weight of the combustion products.

For gases with a uniform number of atoms per molecule, the heat capacity c_{pm}, expressed in kcal/g-mole degree, varies within narrow limits. Since 1 kg contains $1000/\mu$ g-mole combustion products (μ is their molecular weight), the heat capacity per unit weight

$$c_p = c_{pm}\frac{1000}{\mu} \text{ kcal/kg degree.} \qquad (4.6)$$

Consequently, the lower the molecular weight of the combustion products, the greater their heat capacity per unit weight. With a uniform calorific value of the propellant, the energy lost to dissociation in the combustion chamber of a rocket engine operating on a propellant whose combustion products have low molecular weight will be less, due to lower temperature.

The molecular weight of combustion products can be described by two other values: the gas constant R of the combustion products, and the so-called gas-formation value V_g.

For any gas, the universal gas constant $\bar{R}$ = 0.848 kgm/g-mole degree or, in thermal units,

$$A\bar{R} = \frac{0.848}{427} = 1.986 \times 10^{-3} \text{ kcal/g-mole degree.}$$

Here A = 1/427 kcal/kgm is the heat equivalent of work.

The gas constant per unit weight

$$R = 848/\mu \text{ kgm/kg degree.}$$

The smaller the value of the molecular weight of the combustion products, the greater their gas constants. Thus, the gas constant for the propellant combustion products of rocket engines should have a maximum value.

Let us further use the equation of state, written for 1 kg of gas:

$$pV = RT,$$

whence

$$V = R(T/p).$$

We see that under specific conditions the specific volume V of the forming combustion products is equivalent to the value of the gas constant R. The value of the specific volume which the combustion products have under normal conditions ($T = 293.16°K$, $p = 1$ kg/cm^2) is called gas formation V_g. It shows how many liters of combustion products are formed from 1 kg of propellant under normal conditions. The greater this value, the larger the gas constant of the combustion products and the lower their molecular weight. Consequently, during propellant combustion there should be maximum gas formation.

The requirements of low molecular weight μ, or a large gas constant R per unit weight of the combustion products, or high gas formation V_g, are completely identical. If they are fulfilled, high heat capacity per unit weight of the combustion products is assured to a certain degree, and therefore they are widely used when calculating the propellants for rocket engines.

Let us show the influence of the molecular weight of the combustion products on the combustion process, examining the combustion of two propellants: oxygen + hydrogen and oxygen + carbon (see Table 4.1). The combustion products of these propellants (non-dissociated) in both cases are triatomic gases, but their molecular weights differ greatly due to the different atomic weights of hydrogen and carbon, and their different molecular structure. The value of c_{pm} is approximately the same for H_2O and CO_2. The heat capacity per unit weight of the combustion products differs by more than a factor of 2, depending on the values of the molecular weights (R or V_g). Therefore, although the propellant oxygen + hydrogen has a calorific value which is almost 1.5 times that of the propellant oxygen + carbon, its combustion temperature (taking dissociation into consideration) is 335° less than for the second propellant. In this regard, the losses to dissociation ΔQ_{dis}, even in absolute magnitude, are less for the first propellant than for the second (865 kcal/kg vs. 935 kcal/kg). The difference in the relative losses to dissociation $\Delta Q_{dis}/K_G$ is particularly noticeable; for the propellant oxygen + hydrogen it is 27%, while for oxygen + carbon it is 44%. As a result, the specific ground thrust under uniform conditions (expansion of the combustion products from a pressure of 100 atm to 1 atm) $P_{sp\ 0(100:1)}$ is considerably greater in the first case than in the second.

Table 4.1.

K_G, kcal/kg	Combustion, not considering dissociation					Combustion, considering dissociation				
	Gas	μ	c_p, kcal/kg degree	R, kcal/kg degree	V_g, liter/kg (NTP)	T, °K	μ	ΔQ_{dis}, kcal/kg	$\frac{\Delta Q_{dis}}{K_G}$	$P_{sp\ 0(100:1)}$ kg-sec/kg
Oxygen + hydrogen										
3210	H_2O	18	0,640	47,1	1330	3660	16,0	865	0,27	376
Oxygen + carbon										
2140	CO_2	44	0,318	19,2	560	3995	33,6	935	0,44	278
Fluorine + hydrogen										
3210	HF	20	0,350	42,4	1230	4980	18,0	950	0,30	411

The influence of the number of atoms per molecule of combustion products on the combustion process can be studied by comparing the combustion parameters of the propellants oxygen + hydrogen and fluorine + hydrogen (see Table 4.1). These two propellants have identical K_G and approximately the same molecular weight, but the combustion product of the second propellant is a diatomic HF molecule, as opposed to the triatomic H_2O molecule. Therefore, because of the greater stability of diatomic molecules to dissociation, despite the considerably higher combustion temperature ($4980^{\circ}K$ vs. $3660^{\circ}K$), the propellant fluorine + hydrogen has only slightly higher absolute and relative losses to dissociation than the propellant oxygen + hydrogen.

As has already been pointed out, the efficiency of the expansion process also depends on the properties of the combustion products. It is determined by the value of the specific heat ratio $k = c_p/c_V$, where c_V is the heat capacity of the combustion products at constant volume. Later it will be shown (see Chapter VI) that the greater the value of k, the more efficient will be the expansion. The value k for combustion products of rocket propellants is a function of the number of atoms i in the molecules of the gaseous combustion products; it increases with decrease in i.

The influence of the efficiency of expansion on the attainable specific thrust is easily seen if we continue our comparison of the propellants oxygen + hydrogen and fluorine + hydrogen. The amount of heat liberated in the combustion chamber for these propellants, as has been shown, is practically identical, while the specific thrust $P_{sp\ 0(100:1)}$ for fluorine + hydrogen is considerably higher than for oxygen + hydrogen: 411 kg-sec/kg vs. 376 kg-sec/kg (see Table 4.1). This is due to the fact of the complete absence, in the combustion products of fluorine + hydrogen, of triatomic gases, and the high efficiency of expansion.

Another important condition for effective expansion is the gaseous state of the combustion products as they expand in the rocket engine nozzle, since the expansion process and its corresponding energy transition during expansion occur only when the combustion products are in the gaseous state.

Thus, for effective expansion, the propellant combustion products must have such a low boiling point that they remain completely, or at least mostly, gaseous as they pass through the chamber nozzle. In this case, not much heat should be expended on the evaporation of the combustion products, since it is lost with the gaseous combustion products which leave the nozzle.

Now let us mention briefly what we mean by the phrase "good thermodynamic properties of the combustion products." This refers to the following properties of the combustion products: low molecular weight, small number of atoms in the molecules, low boiling point, and low heat of evaporation.

We should mention that it is very difficult to evaluate the advantages of a certain rocket propellant, since the specific thrust, as we have stated, cannot be described merely by the qualities of the propellant, but depends, to a greater extent, on the design features of the engine, mainly of the combustion chamber including the nozzle. We have also seen that the specific thrust is also affected by the additional consumption of the components for the auxiliary needs of the rocket feed system.

The influence of the specific weight of the propellant on the flight performance of the rocket is closely connected with the weight characteristics of the design of the rocket itself.

Therefore, strictly speaking, it is impossible to evaluate the advantages of using a certain propellant if we divorce it completely from the design of the rocket, its propulsion unit, and the feed systems. Only an approximate evaluation can be made.

Sometimes, when calculating the joint influence of the calorific value of the propellant and its specific weight on the properties of a rocket propellant, we use their product

$$K_V = K_G \gamma_{prop}.$$

The value K_V is called the heat density. Physically, it is the concentration of chemical energy per unit volume of the tanks. Since the value K_G in itself is only characteristic of the specific thrust, we should consider that the product

$$P_{sp} \gamma_{prop} = P_{sp\ V},$$

is a more correct combined evaluation parameter; this is measured in kg-sec/liter, and in its physical sense is the impulse removed from a unit volume of the tanks. However, this value as well is only partially valid for a final estimate of the propellant quality, since it assumes that P_{sp} and γ_{prop} exert uniform influence on the terminal velocity of the rocket.

When evaluating propellant properties we usually use the values of the specific thrusts under surface conditions, i.e., $P_{sp\ 0}$ kg-sec/kg and $P_{sp\ 0\ V}$ kg-sec/liter.

Thus, when selecting a propellant, we can perform many operations on the projection of several variants of rockets using various propellants, and on the basis of the obtained results, determine the propellant best suited under the given conditions.

An Analysis of the Fuel and Oxidizer Elements as Rocket Propellant Components

Table 4.2 gives the basic properties and the energy characteristics of fuels and oxidizers for LFRE's.

Table 4.2. Properties of Basic Fuels and Oxidizers and Their Combustion Products

Name of element	Symbol or formula	Phase	Molecular weight	Specific weight γ_{prop} kg/liter	Melting point t_{mp} °C	Boiling point t_{bp} °C
Hydrogen	H_2	liquid	2,016	0,07	—257,14	—252,79
Lithium	Li	solid	6,941	0,534	180	1400
Beryllium	Be	solid	9,01	1,85	1280	—
Boron	B	solid	10,82	1,73	2300	—
Carbon	C	solid	12,01	2,25	—	—
Magnesium	Mg	solid	24,32	1,74	650	1120
Aluminum	Al	solid	26,97	2,70	658	(2000)
Silicon	Si	solid	28,06	2,35	1414	(2400)
Oxygen	O_2	liquid	32,00	1,14	—227	—183
Fluorine	F_2	liquid	38,00	1.5*	—218	—188

Fuel element	Oxygen combustion products: Chemical formula	Phase	Molecular weight	$\Delta H^{0}_{298,16}$ kcal/g-mole	K_G kcal/kg	ν_0 kg/kg	γ_{prop} kg/liter	K_V kcal/liter	t_{bp} °C	ΔH_{ev} kcal/kg	Fluorine combustion products: Chemical formula	Phase	Molecular weight	$\Delta H^{0}_{298,16}$ kcal/g-mole	K_G kcal/kg	ν_0 kg/kg	γ_{prop} kg/liter	K_V kcal/liter	t_{bp} °C	ΔH_{ev} kcal/kg
Hydrogen	H_2O	gas	18,016	— 57,798	3210	7,95	0,42	1350	100	—	HF	gas	20,008	— 64,20	3210	0,46	0,46	1475	19	—
Lithium	Li_2O	solid	29,88	—142,400	4760	1,15	0,75	3570	1300	(1160)	LiF	solid	25,94	—146,3	5650	2,74	0,87	4930	1680	(1960)
Beryllium	BeO	solid	25,01	—146,000	5830	1,78	1,52	8850	(3900)	Very high	BeF_2	solid	47,01	—227,0	4830	4,22	1,23	5950	(1300)	(850)
Boron	B_2O_3	solid	69,64	—302,000	4350	2,21	1,28	5570	—	—	BF_3	gas	67,82	—265,4	3810	5,28	1,21	4750	—101	—
Carbon	CO_2	gas	44,01	— 94,052	2140	2,66	1,32	2830	—78	—	CF_4	gas	88,01	—162,5	1850	6,32	1,22	2250	—128	—
Magnesium	MgO	solid	40,32	—143,840	3530	0,66	1,43	5050	(2250)	Very high	MgF_2	solid	62,32	—263,5	4210	1,56	1,31	5550	(2230)	(1110)
Aluminum	Al_2O_3	solid	101,94	—399,500	3900	0,835	1,66	6460	2700	(1137)	AlF_3	solid	83,97	—311,0	3710	2,12	1,41	5220	(1290)	(1100)
Silicon	SiO_2	solid	60,06	—205,300	3350	1,14	1,50	5030	1900	(1310)	SiF_4	gas	104,06	—361,3	3470	2.79	1,33	4600	—95	—

When examining this table, we should not be surprised at the blank spaces and the doubtful data (the numbers in parentheses). Although most of the substances in the table have long been used in technology, their properties and those of their compounds have only recently begun to be studied in detail, in connection with rocketry. Data in the literature on these substances are extremely contradictory and often change. However, the information in the table is sufficient to give a basic evaluation of these elements as LFRE propellant components.

Of the oxygen compounds, BeO has the highest heat of formation per unit weight: 5830 kcal/kg. This is considerably higher than the calorific value of the hydrocarbon propellants now used. B_2O_3 and Li_2O also have high heats of formation. H_2O, MgO, Al_2O_3, and SiO_2 have lower heats of formation. CO_2 has the lowest heat of formation. The volume concentration of energy (thermal density) is maximum for BeO and minimum for H_2O. This latter fact is the essential drawback of liquid hydrogen as a rocket fuel. Of the oxygen compounds, water vapor and CO_2 have the best thermodynamic properties. The diatomic substances BeO and MgO have an extremely high boiling point and heat of evaporation. For BeO, e.g., it is higher than the heat of formation of the solid product. The thermodynamic properties of the other oxides are even worse.

Of the fluorine compounds, LiF has the highest calorific value per unit weight. Other elements also have high energy concentration, per unit weight and unit volume. As in the case of the oxygen propellants, those fluorine propellants which use the most readily available fuel elements – hydrogen and carbon – have the lowest calorific values. In this case, the heat of formation per unit weight of HF is the same as for H_2O, while CF_4 is lower than CO_2. Despite this, it might be advantageous to use fluorine oxidizers (e.g., OF_2) for ordinary hydrocarbon propellants, if we select the propellants such that the hydrogen combines basically with the fluorine, and the carbon combines with the oxygen. The thermodynamic properties of the combustion products of such propellants will be better than for oxygen + hydrocarbon propellants. The thermodynamic properties of the fluorine compounds of all the other elements are better than for the oxygen compounds.

These, in brief, are the basic results of an analysis of the elementary propellants. Propellants whose components are complex substances should be analyzed in like manner. In this case, the standard heats of formation of the propellant components, as well as the types of fuels and oxidizers used in them, influence the calorific value of the propellants.

2. Present-Day Liquid and Solid Propellants for Rocket Engines

The Requirements of Liquid Propellants

In addition to the basic requirements examined above, the use of propellants in LFRE's involves many other extremely diversified requirements, ones which are difficult to fulfill. Fulfillment of these requirements simplifies the creation of a reliably designed rocket engine and assures its successful operation under the most diverse conditions. The cooling of an engine requires that the propellant components absorb as much heat as possible or, so to speak, have maximum heat absorptivity.

The value of the heat absorptivity of the component

$$Q = c(T_{bp} - T_0) \text{ kcal/kg},$$

where c is the heat capacity of the component; T_{bp} is the boiling point of the component at the pressure in the cooling jacket (see below for a description of the cooling system); and T_0 is the initial temperature of the component.

Components having a higher heat capacity and a higher boiling point will have greater heat absorptivity. The heat absorptivity of a component should, more correctly, pertain to 1 kg of the propellant burned in the engine, since both the amount of heat liberated in the engine and the amount of the substance used for cooling are proportional to the total propellant consumption, not to that of an individual component.

The specific heat absorptivity of the fuel

$$Q_f = c_f(T_{bp\ f} - T_0)\frac{1}{1 + \nu} \text{ kcal/kg of propellant;}$$

the specific heat absorptivity of the oxidizer

$$Q_o = c_o(T_{bp\ o} - T_0)\frac{\nu}{1 + \nu} \text{ kcal/kg of propellant.}$$

With uniform physical properties (heat capacity and boiling point) the oxidizer will have greater heat absorptivity than the fuel, since $\nu > 1$. Therefore, if an oxidizer with a high boiling point is used in the LFRE it will, as a rule, be used for cooling. The total heat absorptivity of the coolant, QG (where G is the propellant weight flow per unit time), should be greater than the total amount of heat to be removed from the cooled surface of the engine per unit time. If this condition is not fulfilled by one individual component, the engine must be cooled by both components simultaneously.

The feeding of the propellant into the combustion chamber requires that the components have low viscosity, and also that this viscosity be weakly dependent on temperature. Otherwise, a change in the temperature of the surrounding medium may lead to a change in the ratio ν of the components fed to the chamber, and decrease the usefulness of the propellant.

Mixture formation requires maximum equal volumetric flow rate of the propellant components, since in this case a uniform mixture of the propellant components forms more easily in the chamber.

The ignition and combustion of the propellant in the chamber require low ignition temperature for nonhypergolic propellants, and a minimum ignition delay time for hypergolic propellants. This last requirement is extremely important.

By hypergolic delay time we mean the time it takes for the propellant to ignite from the moment the liquid components of the hypergolic propellant come together. This value is sometimes called the induction period. Naturally, during chemical combustion, the amount of propellant which accumulates in the combustion chamber when the engine is started will (other conditions being equal) be the greater, the longer the hypergolic lag period. As has already been mentioned, the accumulation of the mixture in the chamber can lead to an explosion in the engine.

To successfully start an engine under various meteorological conditions and at various altitudes it is necessary that the hypergolic lag time not increase appreciably with a drop in propellant temperature or a decrease in atmospheric pressure. To assure normal combustion, it is required that the propellants not be susceptible to vibration combustion, in the case of LFRE's, anomalous combustion, in the case of SFRE's (see Chapter V), or detonation.

The use of engines and rockets requires that the propellants be physically and chemically stable (in order that the components be stored for long periods of time without any particular precautionary measures being taken), explosion proof, and have high boiling point and low solidification point. The propellant should be nontoxic and should not exert corrosive action on the structural materials. In addition, a rocket propellant should be inexpensive, be able to be produced in large quantities, and enough raw material must be available for its production.

It goes without saying that as yet there are no propellants which satisfy all the above requirements.

At present, a rather limited number of chemical compounds are used as the propellant components for rocket engines. However, there will undoubtedly be more available in the near future as a result of vast investigations in this field.

The Classification of Liquid Rocket Propellants

Rocket propellants can be classified according to their various features. The first classification is according to the physical state of the propellant. According to this we divide it into liquid and solid. There is an intermediate group of hybrid propellants, one of whose components is liquid, the other — solid. We might note the attempts to use such propellants in LFRE's. For example, there are rocket engines which use jellied gasoline (napalm), carbon rods placed directly in the combustion chamber, and LOX fed from tanks.

Let us first examine the classification of liquid rocket propellants. These propellants must first be divided according to the number of components which must be fed to the combustion chamber. The following types of propellants are known at present: tripropellants and bipropellants with separate feed (three and two systems of pipes, respectively, are required to feed them to the combustion chamber), and monopropellants, fed to the chamber through a single pipeline system. Separate-feed bipropellants are most widely used at present. Sometimes a tripropellant is used, but this complicates the feed system, and is usually a forced measure. For example, sometimes water is fed to the chamber as the third component; this decreases the extremely high combustion temperature.

Bipropellants are divided according to the type of oxidizer used. There are two types of oxidizers: those with high boiling points, and those with low boiling points. The first type of oxidizer has great operational advantages and makes it possible to produce rockets, containing fuel, which can be stored for long periods. The propellants can be classified in even greater detail according to the specific oxidizer and fuel used in a given propellant. Additional classifications can be derived to increase the number of substances used as propellant components in LFRE's.

Monopropellants can be divided into two classes. The first contains propellants which consist of a single chemical substance in which there is a supply of chemical energy. Such a propellant can be called a homogeneous monopropellant. The properties of such propellants are determined entirely by the properties of the given chemical substance. The second class of monopropellants consists of propellants with two (sometimes more) liquid substances, dissolved in one another. In this case, one of the substances, as a rule, has excess oxidizer, and the other has excess fuel. A monopropellant with various properties can be obtained by varying the ratio of these substances.

The disadvantage of monopropellants is that they are, as a rule, explosive when their composition is most advantageous, i.e., when the oxidizer excess factor α is close to unity.* To decrease the danger of explosion, monopropellants are ballasted with water or one of the components, but this decreases the calorific value of the propellant and impedes combustion. Therefore, much effort has been devoted to finding substances (so-called flegmatizers) which, when added in small amounts, will sharply decrease the danger of explosions of the propellant.

The third method for propellant classification is that with which we are already familiar: the classification into nonhypergolic and hypergolic. Figure 4.2 gives the classification of liquid rocket propellants.

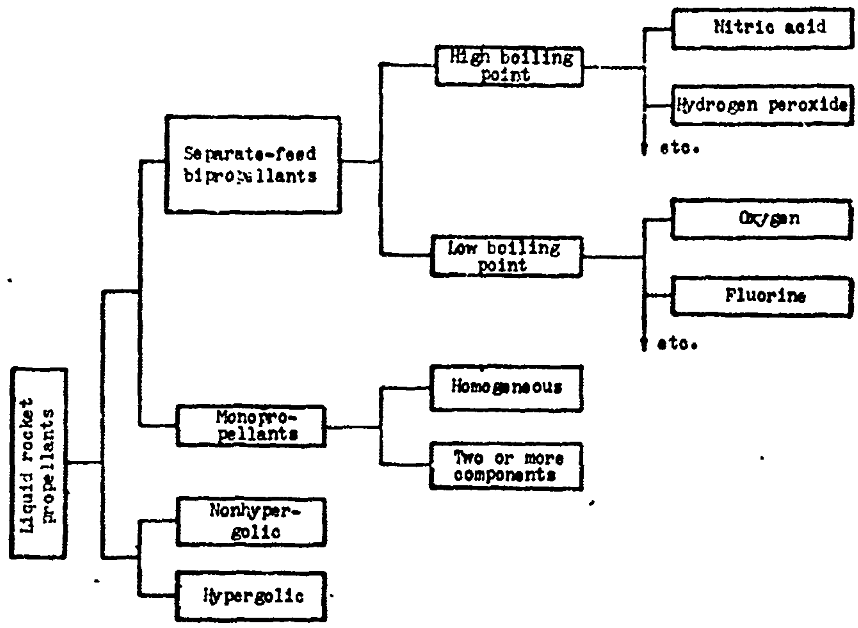

Fig. 4.2. Classification of liquid rocket propellants.

The properties of the propellants are determined mainly by the properties of the oxidizers, and therefore the propellants are also divided into groups according to the type of oxidizer used. Therefore, we will begin the description of propellants with a description of the oxidizer properties.

Tables 4.3, 4.4, and 4.5 give the properties of oxidizers, fuels, and certain other rocket propellants. These tables give the element composition, i.e., the elements (C, H, O, and N) in 1 kg of fuel or oxidizer, the heat of formation, the specific weight, the melting (solidification) point, and the boiling point. For the propellants, the tables give the calorific value, the specific weight γ_{prop}, sample values of the molecular weight μ of the combustion products, the specific ground thrusts $P_{sp\ 0(40:1)}$ and $P_{sp\ 0\ V(40:1)} = \gamma_{prop} P_{sp\ 0(40:1)}$, and also the so-called parameter complex $\beta = S_{cr} p_t / G$ kg-sec/kg (see p.).

These data were obtained from calculations using a chamber pressure of 40 atm and a nozzle exit pressure of 1 atm. The propellant component ratio in the calculations was selected such that a given propellant would produce maximum thrust.

* The composition of a propellant for which $\alpha = 1$ is called stoichiometric.

Table 4.3. Basic Data on Certain Pure Oxidizers for Liquid-Fuel Rocket Engines

Name	Chemical formula	Molecular weight	Element composition, kg/kg					Heat of formation ΔH^0_{293} kcal/g-mole	Specific weight at 15°C kg/liter	Melting point °C	Boiling point*** °C
			O	C	H	N	F				
Nitric acid	HNO_3	63,02	0,762	—	0,016	0,222	—	—41,66	1,52	—41,6	+86
Nitrogen tetroxide	N_2O_4	92,02	0,696	—	—	0,304	—	— 6,80	1,47	—11,2	+21
Tetranitromethane	$C(NO_2)_4$	196,04	0,653	0,061	—	0,286	—	+ 8,80	1,65	+13,8	+126
Liquid oxygen (LOX)	O_2	32,00	1,000	—	—	—	—	0,00****	1,14*	—227	—183
Hydrogen peroxide	H_2O_2	34,02	0,940	—	0,060	—	—	—44,84	1,46	—2	+151
Liquid fluorine	F_2	38,00	—	—	—	—	1,000	0,00****	1,51*	—218	—188
Fluorine monoxide	OF_2	54,00	0,457	—	—	—	0,543	+5,50****	1,53*	—224	—145
Water (in the liquid state)**	H_2O	18,02	0,889	—	0,111	—	—	—68,35	1,00	0	+100

*at the boiling point
**the water is contained in the oxidizer as an inert substance
***at atmospheric pressure
****for the gaseous substance

Table 4.4. Basic Data on Certain Fuels for Liquid-Fuel Rocket Engines

Name	Chemical formula	Molecular weight	Element composition, kg/kg				Heat of formation ΔH^0_{293} kcal/g-mole	Specific weight at 15°C kg/liter	Melting point °C	Boiling point* °C
			C	H	O	N				
Kerosene	—	—	0,858	0 135	0,007	—	—46,0	0,76÷0,84	170(150÷315)**	—50÷—70
Ethanol	C_2H_5OH	46,07	0,522	0,131	0,347	—	—66,36	0,798	78,5	—117,3
Methanol	CH_3OH	32,04	0,375	0,125	0,500	—	—57,02	0,791	64,6	— 91,9
Aniline	$C_6H_5NH_2$	93,06	0,074	0,076	—	0,150	+ 7,08	1,022	184,4	— 6,2
Triethyl amine	$(C_2H_5)_3N$	101,07	0,712	0,149	—	0,139	—42,33	0,728	89,5	—114,8
Xylidine	$(CH_3)_2C_6H_3NH_2$	121,12	0,792	0,092	—	0,116	—46,2	0,98	210	— 54
Furfurol	$C_4H_3OCH_2OH$	98,06	0,613	0,062	0,325	—	—63,1	1,13	171	— 32
Vinyl ethyl ether	$C_2H_3OC_2H_5$	72,07	0,667	0,111	0,222	—	—46,0	0,754	36	— [illegible]
Hydrazine	N_2H_4	32,03	—	0,125	—	0,875	+12,05	1,01	113,5	— 2
Dimethylhydrazine (unsymmetric)	$N_2H_2(CH_3)_2$	60,12	0,400	0,134	—	0,466	+11,28	0,83	63	— 58
Hydrazine hydrate	$N_2H_4 \cdot H_2O$	50,06	—	0,122	0,318	0,560	—63,13	1,03	118,3	— 40
Ammonia	NH_3	17,08	—	0,177	—	0,823	—16,00	0,68***	—33	— 77

*at atmospheric pressure
**start of boiling
***at the boiling point

Table 4.5. Basic Data on Certain Propellants for Liquid-Fuel Rocket Engines

Oxidizer	Fuel	K_G kcal/kg	γ_{prop} kg/liter ($\alpha=1$)	K_V kcal/liter	μ	T^*, °K	$P^*_{sp\,0}$ (40:1) kg-sec/kg	$P^*_{sp\,0\,V}$ (40:1) kg-sec/liter	β kg-sec/kg
98% nitric acid	Kerosene	1460	1,36	1990	26	2980	230	313	156
the same	Tonka 250	1490	1,32	1970	26	3000	235	310	157
the same	Aniline (80%) + furfurol (20%)	1420	1,39	1980	26	3050	225	313	156
Tetranitromethane	Kerosene	1590	1,47	2340	26	3200	245	360	163
Nitrogen tetroxide	the same	1550	1,38	2140	25	3300	240	331	163
80% hydrogen peroxide		190	1,35	730	23	780	90	122	72
80% hydrogen peroxide	Methanol (50%) + hydrazine hydrate (50%)	1020	1,30	1330	20	2600	215	280	149
LOX	Kerosene	2200	1,00	2200	24	3600	275	275	177
the same	93.5% ethanol	2020	0,99	2000	23	3300	255	247	172
the same	Dimethylhydrazine	2200	1,02	2240	22	3350	285	291	168
the same	Ammonia	1650	0,89	1470	19	3000	285	254	167
Liquid fluorine	Hydrazine	2230	1,32	2940	20	4650	315	455	187
the same	Ammonia	2315	1,18	2730	20	4650	315	410	197

*the values for combustion temperature T and specific thrust $P_{sp\,0}$ are preliminary, for average (data-wise) LFRE's with a 40:1 ratio of chamber pressure to nozzle exit pressure.

Nitric Acid and Nitrogen Oxides. Tetranitromethane

Nitric acid, HNO_3, is a product widely used in the national economy. It was produced in great quantities even before the advent of LFRE's.

The first proposal for using nitric acid as an oxidizer for LFRE's was made in 1930. At present, nitric acid propellants are widely used in rocketry.

Pure nitric acid is a colorless liquid. Technical nitric acid always contains a certain amount of water, and also nitrogen oxides, which color it. The color of the acid varies from yellow to reddish brown, depending on the nitrogen oxide content.

The presence of water in nitric acid is undesired, since this decreases the calorific value of the propellant. Therefore, LFRE's use acid with a concentration of 99-96%, i.e., with a water content of not more than 1-4%.

Pure nitric acid (see Table 4.3) contains 76% oxygen and has a quite low negative heat of formation, which makes it a relatively powerful oxidizer. Of all the widely used oxidizers, nitric acid has the highest specific weight, which makes it possible to produce a propellant having high energy concentration per unit volume.

The boiling point (+86°C) and freezing point (-42°C) of nitric acid highly favors its use in LFRE's. A 10% addition of water to nitric acid lowers the freezing point somewhat (to -68.5°C). The further addition of water raises the freezing point.

Nitric acid is an oxidizer with a high boiling point, and has all the advantages of such oxidizers. The boiling point increases with an increase in pressure; at the pressures which occur in the cooling jacket of an LFRE, the boiling point of nitric acid exceeds 200°C.

The heat capacity of nitric acid is about 0.5 kcal/kg degree which, together with the rather high boiling point and high relative content of nitric acid in the propellant ($\nu_0 = 5.47$), makes HNO_3 a high-quality oxidizer with great heat absorptivity.

Nitric acid also has a number of disadvantages. HNO_3 vapors are toxic; when nitric acid gets on the skin it causes severe burns. Therefore, precautionary measures must be taken when working with HNO_3.

Nitric acid has a highly corrosive effect on metals and other structural materials. Aqueous HNO_3 reacts particularly vigorously with metal; therefore, the engine and its parts must be carefully flushed after using HNO_3. Stainless steel, aluminum, and certain plastics are materials which can withstand the effects of HNO_3.

Nitric acid evaporates readily and tends toward spontaneous decomposition with the liberation of gaseous oxygen, which causes inconvenience in storage.

Various additives can be added to HNO_3 to improve its properties as an oxidizer. These can be added to increase the calorific value of the fuel, increase the specific weight of the oxidizer, decrease the corrosive effect on structural materials, increase the activity of the oxidizer with respect to the fuel (particularly with respect to fuels that are hypergolic with HNO_3), and, finally, to decrease the freezing point of the propellant components. Many additives change not one, but several properties of HNO_3, i.e., they have a combined effect.

Nitrogen tetroxide (e.g., the oxidizer RFNA), sulfuric acid, and other substances are used as additives to HNO_3.

Nitrogen tetroxide is an oxygen-rich nitrogen oxide with a considerably lower (compared with HNO_3) negative heat of formation (-74 kcal/kg, vs. -660 kcal/kg for HNO_3). Liquid nitrogen tetroxide is yellow colored, and evaporates readily. Its vapors decompose in air and have a clearly expressed yellow color. It is difficult to use pure N_2O_4 as an oxidizer because of its high freezing point (-11.2°C) and low boiling point (+21°C). Therefore, N_2O_4 is used only as an additive to HNO_3 to increase the calorific value of the propellant. Moreover, the addition of N_2O_4 to HNO_3 gives a solution with a higher specific weight than the specific weights of N_2O_4 and HNO_3 individually. The maximum specific weight of such a solution, containing 40% N_2O_4, is 1.63 kg/liter. The addition of N_2O_4 also increases the activity of the oxidizer and, accordingly, facilitates the starting of the LFRE.

The addition of N_2O_4 to HNO_3, just as the addition of water, changes the freezing point of the mixture. The lowest freezing point (-73°C) is obtained with the addition of 18% N_2O_4. Any further increase in N_2O_4 increases the melting point of the mixture.

Concentrated sulfuric acid is used as an additive, to lower the corrosive properties and improve the starting of the engine, particularly with hypergolic fuels. Mixtures of HNO_3 and H_2SO_4 are called nitrating mixtures. One of the disadvantages of adding H_2SO_4 to HNO_3 is the decrease in calorific value of the propellant.

The addition of a small amount of hydrofluoric acid (liquid HF) to HNO_3 sharply decreases its corrosive properties.

Ferric chloride is added to HNO_3 to lower the freezing point and increase the activity of the acid.

Besides HNO_3 and N_2O_4, tetranitromethane, of the nitrogen compounds of oxygen, can be used as an oxidizer. It has a positive heat of formation. The great value of tetranitromethane as an oxidizer is its high specific weight (1.65 kg/liter), which is higher than for HNO_3. Tetranitromethane does not corrode structural materials. The use of tetranitromethane is limited by its tendency toward explosions. In addition, it is a highly toxic substance, effecting the mucous membranes.

The freezing point of tetranitromethane is relatively high, +13.8°C, although when mixed with N_2O_4 it drops to -26°C; this makes it possible to use this mixture with no difficulty as an oxidizer in LFRE's.

Fuels Used in Propellants Containing Nitric Acid and Nitrogen Oxides

The most widely used fuel component of propellants containing HNO_3 and nitrogen oxides is kerosene. The main physical properties of kerosene are given in Table 4.4, while the properties of the propellant HNO_3 + kerosene are given in Table 4.5.

Kerosene can be used as the fuel for an LFRE because of its many positive qualities. Propellants containing kerosene have a high calorific value. Kerosene remains liquid throughout a broad range of temperature changes. Kerosene can be used to cool the engine: its heat capacity is 0.45 kcal/kg degree, and its boiling point at increased pressures is 250°C. There are no difficulties in transporting and storing kerosene, and it can be easily produced, thanks to the vast development of the petroleum refining industry.

Kerosene can be used as the fuel with all oxidizers which have oxygen compounds of nitrogen as their base. A propellant with HNO_3 and kerosene has a calorific value of 1460 kcal/kg. It increases somewhat for nitrogen oxides (to 1550 kcal/kg) because of the better properties of the oxidizer. The disadvantage of kerosene as a fuel is its relatively low specific weight (0.76-0.84 kg/liter). Kerosene + HNO_3 and other propellants with kerosene and nitrogen oxides are not hypergolic, and require forced ignition.

Numerous attempts to activate kerosene so that it will be hypergolic with HNO_3 have been unsuccessful. Only recently, information has appeared to the effect that a relatively small (about 20%) amount of unsymmetric dimethylhydrazine* $N_2H_2(CH_3)_2$ added to kerosene produces a fuel which is quite reliably hypergolic with HNO_3 and other such oxidizers.

Other hydrocarbons, e.g., alcohols, which produce nonhypergolic propellants in combination with HNO_3, have not been used for LFRE's.

Hypergolic propellants have also been developed using HNO_3 and nitrogen oxides. Such propellant fuels are complex organic compounds, e.g., aniline $C_6H_5NH_2$, furfurol $C_4H_3OCH_2OH$, unsymmetric dimethylhydrazine $N_2H_2(CH_3)_2$, xylidine $(CH_3)_2C_6H_3NH$, and triethyl amine $(C_2H_5)_3N$. Table 4.4 gives their composition and basic physicochemical properties. Of the special properties of these hydrocarbons, let us merely indicate the following: aniline and furfurol have somewhat high specific weights (1.022 kg/liter and 1.13 kg/liter, respectively), and aniline has a low, positive, heat of formation (+76 kcal/kg).

In order to obtain as short a hypergolic delay time as possible, other conditions being satisfactory, we select the best mixtures of fuel consisting of various substances. There are a great many hypergolic propellants with various compositions. Their common disadvantage is their high cost and unavailability of the components.

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* Depending on its molecular structure, dimethylhydrazine has two modifications: symmetric and unsymmetric, having different properties but the same chemical composition.

In practice, in addition to being used as the basic propellant in an engine, hypergolic mixtures are often used as igniters in chemical combustion systems.

The most widely used hypergolic fuels in reaction with HNO_3 (and other compounds of nitrogen and oxygen) are as follows:

1) a mixture of 50% xylidine and 50% triethyl amine, called "tonka 250";

2) a mixture of 80% aniline and 20% furfurol.

Hypergolic propellants containing HNO_3 and nitrogen oxides are considerably more resistant to vibrational combustion than is the propellant HNO_3 + kerosene. Therefore, despite their relatively high cost, they are sometimes used as the basic propellants for rocket engines.

Liquid Oxygen (LOX)

Liquid oxygen is an even more powerful oxidizer than HNO_3, since it contains 100% oxidizer. LOX is a transparent light-blue liquid which boils at $-183^{\circ}C$. Its specific weight is considerably less than that of HNO_3, 1.14 kg/liter at the boiling point. Because of its low boiling point, LOX cannot be used as a coolant. Therefore, oxygen is not suitable for use in rockets which must be stored while fueled. The rocket's tanks are filled with LOX directly before launching.

Much oxygen is lost to evaporation when LOX is stored, transported, and placed in the tanks of rockets.

LOX is relatively harmless. When small amounts get on one's skin, it boils, and the layer of gaseous oxygen that forms protects the skin from frostbite.

Recently, LOX has been used extensively in many branches of engineering; therefore, it can be produced in large amounts. The problems of storing and transporting LOX have also been satisfactorily solved. Therefore, despite the unavoidable losses to evaporation, the cost of the LOX used in rockets is not great.

The Fuels Used in Propellants Containing LOX

Any of the hydrocarbons can be used as fuels in combination with LOX, producing nonhypergolic propellants. The propellant LOX + kerosene has a high calorific value, 2200 kcal/kg. This is generally one of the most powerful propellants in use at the present time. It is used, e.g., in the Atlas and the Thor. Attempts to use kerosene + oxygen as a propellant had been made at the outset of the development of rocketry. This propellant was difficult to use in LFRE's because of its high combustion temperature in the combustion chamber, and also the relatively small amount of kerosene in the propellant (approximately 20%), which made it difficult to use to cool the engine.

Propellants with LOX as the oxidizer and ethanol or methanol as the fuel are widely used at present.

Table 4.4 gives the main features of ethanol and methanol. Ethanol with 93.5% or less concentration is used in technology. It is too complex to obtain a higher concentration. The calorific values of the alcohols are lower than for kerosene, since they have a higher negative heat of formation, although the combustion temperature of alcohol in kerosene is lower. This facilitates the creation of a reliably operating engine. The alcohols have low specific weights (0.8 kg/liter). The boiling point is sufficiently high (considering the pressure in the cooling jacket of the engine), so that the alcohol can be used as a coolant. The heat capacity of the alcohols is somewhat higher than that of kerosene, approximately 0.6 kcal/kg degree. Due to the fact that the alcohol itself contains a considerable amount of oxygen, the relative amount of alcohol in the propellant can be 40-45%. This also makes it possible to use the alcohol to cool the engine.

Ethanol and methanol can mix in any proportion with water. This makes it easy to produce a propellant with varying calorific value, and thus to lower the temperature in the combustion chamber and increase the total heat absorptivity of the fuel to any degree. This was just the method used by the designers of the first LRBM's, using an aqueous alcohol solution of 75% concentration, although this decreased the specific thrust considerably (to 208 kg-sec/kg).

The low solidification temperature of alcohol allows it to be used within a broad range of temperatures of the surrounding medium.

Great amounts of alcohol are produced from the most diverse raw materials. Alcohol has no corrosive effect on structural materials, which makes it possible to use relatively inexpensive materials for the alcohol tanks and lines.

Ethanol can be replaced by methanol which, in reaction with oxygen, gives a somewhat poorer-quality propellant. Methanol mixes with ethanol in any proportion, which makes it possible for it to be used when there is insufficient ethanol, and added to the fuel in a certain percentage. LOX propellants are used almost exclusively in long-range rockets which allow, and even require (because of the great weight), that they be fueled at the launch site.

Hydrogen Peroxide

Pure hydrogen peroxide (i.e., 100% concentration) is not used in engineering since it is an extremely unstable product, capable of spontaneous decomposition, and exploding easily as a result of any (it would appear) insignificant external disturbance: a blow, illumination, the slightest contamination by organic substances, and admixtures of certain metals.

Rocketry utilizes more stable highly concentrated solutions of hydrogen peroxide in water (most often, 80-96% concentration). For increased stability, small amounts of substances (e.g., phosphoric acid) are added to hydrogen peroxide to prevent its spontaneous decomposition. The handling of 80% hydrogen peroxide at present requires only the usual protective measures taken when handling strong oxidizers. Hydrogen peroxide of such a concentration is a transparent light-blue liquid with a freezing point of $-25^{\circ}C$.

Heat is liberated when hydrogen peroxide decomposes into oxygen and water vapor. This liberation of heat is explained by the fact that the standard heat of formation of hydrogen peroxide is 44.84 kcal/g-mole, while that of water is 68.35 kcal/g-mole. Thus, when peroxide decomposes in accordance with the reaction $H_2O_2 = H_2O + \frac{1}{2}O_2$, chemical energy is released; this equals the difference $68.35 - 44.84 = 23.51$ kcal/g-mole, or 690 kcal/kg.

Hydrogen peroxide of 80% concentration can decompose in the presence of catalysts, with heat liberation of 190 kcal/kg and the release of free oxygen, which can be used to oxidize the fuel. Hydrogen peroxide of such a concentration has considerable specific weight (1.35 kg/liter). Stabilized hydrogen peroxide can be used to cool the engine.

Stainless steel and very pure aluminum (no more than 0.5% impurities) can be used as the materials for the tanks and pipelines for engines operating on peroxide. Copper and other heavy metals, which aid in the decomposition of hydrogen peroxide, cannot be used. Certain types of plastic can be used as packing and seals. Concentrated hydrogen peroxide burns the skin and ignites organic materials.

Propellants Using Hydrogen Peroxide

Two types of propellants using hydrogen peroxide have been created. The first type is a separate-feed propellant, in which the oxygen liberated by the decomposition of hydrogen peroxide is used to combust the fuel. An example is the propellant used in the previously described interceptor plane engine (Chapter III). It consisted of 80% concentrated hydrogen peroxide and a mixture of hydrazine hydrate ($N_2H_4 \cdot H_2O$) and methanol. When a special catalyst is added to the fuel the propellant becomes hypergolic. The relatively low calorific value (1020 kcal/kg) and also the low molecular weight of the combustion products determine the low combustion temperature, which facilitates the operation of the engine. However, because of the low calorific value, the engine has low specific thrust (190 kg-sec/kg).

The second separate-feed fuel consists of 90-92% concentrated hydrogen peroxide and kerosene. It gives approximately the same specific thrust as the HNO_3 propellant.

Hydrogen peroxide with water and ethanol can form relatively explosion proof ternary mixtures; these are examples of monopropellants. The calorific value of such safe mixtures is relatively low: 800-900 kcal/kg. Therefore, they can hardly be used as the basic propellant for LFRE's. Such mixtures can be used in steam generators.

The decomposition of concentrated peroxide, as has already been mentioned, is greatly utilized in rocketry to obtain the steam which is the working substance for the turbine in the pump-feed system.

We also know of engines in which the energy of peroxide decomposition serves to create thrust. The specific thrust of such engines is low (90-100 kg-sec/kg).

Two types of catalysts are used to decompose peroxide: liquid (e.g., an aqueous solution of potassium or sodium permanganate), and solid.

The solid catalyst is a porous substance, whose pores contain the catalyst to decompose the peroxide, e.g., potassium permanganate. Other types of solid catalysts are formed by the sintering of various mixtures, also containing the catalyzing substance.

The solid catalyst is placed directly in the reactor, where the hydrogen peroxide decomposes; this eliminates the need for a system to feed the liquid catalyst to the reactor.

Solid Rocket Propellants

Historically, the first propellants used in rocket engines were solid propellants — powders. Powder is a special type of explosive — a projectile explosive. In addition to the requirements placed on all rocket propellants, solid propellants should satisfy a number of additional requirements, including the following:

1. A solid propellant should burn stably at low pressures in the chamber. This

is due to the fact that the combustion chamber of a solid propellant engine is simultaneously the container for all the propellant. At high pressure, the great thickness of the walls leads to great chamber weight and a high value of the relative terminal weight μ_t of the rocket.

2. A solid propellant must have high mechanical stability. This is necessary because the propellant grain undergoes pressure in the combustion chamber, and considerable inertial loads. The destruction of the grain during the combustion process would lead to an increase in the combustion surface, an increase in pressure in the chamber, and an explosion.

3. To assure the required duration of combustion of the solid propellant, the grain should have the appropriate dimensions.

4. The properties of a solid propellant grain should be uniform and identical. Otherwise, there may not be uniformity within the same types of rockets and engines.

Certain other requirements placed on solid propellants and powders will be discussed in Chapter V. devoted to propellant combustion.

Rocket engines of the past first used ordinary artillery powder, black and smokeless, prepared using a so-called volatile solvent (ether and alcohol) which, as the powder solidifies, is removed by evaporation. However, about 20 years ago special rocket powders were created using a nitrocellulose base. These powders are colloidal solutions of two or more organic nitrogen compounds.

Nitrocellulose is the basic substance used in the powder, assuring the required physical properties.

Nitrocellulose is obtained by treating cellulose, i.e., organic fibrous substances (wood cellulose or cellulose obtained from cotton), with concentrated HNO_3. Bales of cellulose are immersed in vats containing HNO_3. With such processing the cellulose is converted into esters containing oxygen, carbon, hydrogen, and nitrogen.

Thus, nitrocellulose is a monopropellant which liberates heat as is burns. The calorific value of this propellant is low, since there is not enough oxygen for complete combustion.

Nitroglycerine is used as the liquid to dissolve the nitrocellulose and assure that it will jell (i.e., produce a solid solution of uniform composition and with uniform physicochemical properties). Nitroglycerine is a solvent which can remain in the powder and does not have to be evaporated out. Such powders are called nonvolatile solvent powders. Nitroglycerine is the second basic substance which makes up the powder.

Nitroglycerine is obtained by treating glycerine with a mixture of concentrated nitric and sulfuric acids. A nitroglycerine molecule [chemical formula $C_3H_5(ONO_2)_3$] contains a considerable amount of oxygen which oxidizes the fuel elements of not only the nitroglycerine but, in part, the nitrocellulose as well. The calorific value of nitroglycerine is 1485 kcal/kg. Nitroglycerine is the basic substance which assures combustion; therefore, its content, by percents, determines the calorific value of the powder.

At normal temperature, nitroglycerine is a heavy (specific weight 1.6 kg/liter) oily liquid. Pure nitroglycerine is colorless, while technical nitroglycerine has a pale yellow color. Its vapors are toxic, causing headaches, while large doses can cause poisoning.

Ordinary artillery powder (smokeless) is prepared by dissolving nitrocellulose in a volatile solvent (a mixture of alcohol and ether) which, after the grains are prepared, is removed by drying. Large grains cannot be made from pyroxilin since they cannot be completely dried. In addition, pyroxilin powders have a relatively low calorific value. This is the reason why modern rocket powder is made using nonvolatile solvents.

Glycerine, the source product for nitroglycerine, requires edible fats in its production. Therefore, in the production of rocket powder we tend to partially replace glycerine with other solvents, e.g., diethyl phthalate, dinitrotoluene, diethyleneglycol dinitrate, and certain others.

In addition to the above-named basic substances, rocket powder contains small amounts of various additives. To make the powder chemically stable it contains certain additives such as diphenyl amine and centralite. These substances, which keep the powder from chemically decomposing, also prevent changes in the physical properties and structure of the powder. A change in structure of the powder during storage is highly undesirable, as it may disrupt the normal combustion process and cause the engine to explode. Certain additives (vaseline, wax) make the powder mass plastic, which is necessary for the preparation of powder grains.

Since rocket powder is a mixture of various substances, the calorific value of the powder is determined by the composition and heat of formation of its components.

The nitrogen, which acts as the bond between the fuel and oxidizer elements in a single molecule, is a ballast in powders, as in the liquid propellants. Therefore, a high nitrogen content leads to a decrease in the calorific value of the powder. The nitrocellulose used in the powders usually contains 12-14% nitrogen.

The calorific value of the powder is also determined by the value of the so-called oxygen balance, i.e., the value ν for powder or its components. For most powders, $\nu < \nu_0$ or, as is said, the powder has a negative oxygen balance, which determines the oxygen deficit ($\alpha < 1$) and incomplete combustion of the fuel elements.

Most rocket powder components, except diphenyl amine, have a negative heat of formation.

Because of the aforementioned factors (high nitrogen content, negative oxygen balance, and negative heat of formation of the components), the powder has a relatively low calorific value, varying between 820 and 1250 kcal/kg depending on the nitroglycerine (or other solvent) content.

The so-called derived force of the powder f_0, in kg-m/kg, is often used, in addition to the calorific value, as a power characteristic of rocket powder.

The powder force is the product of the gas constant R and the temperature of the combustion products T. To determine the derived force of the powder, the combustion temperature is taken in relation to the conditions of powder combustion in the rocket engine at constant pressure.

The derived force of the powder enters directly into expressions for exhaust velocity and therefore, in certain cases, it is more suitable for calculations than the calorific value.

Table 4.6 gives the composition and properties of certain rocket powders.

For ordinary rocket powders, combustion at relatively low pressures cannot be easily guaranteed; moreover, the powders are expensive. Therefore, new solid propellants have recently been developed which cannot be called powders, strictly speaking (projectile explosives).

Inorganic substances, rich in oxygen, are used in these propellants as the oxidizer; these include: perchlorates (e.g., potassium or ammonium perchlorate) and nitrates (e.g., ammonium or potassium nitrate).

These inorganic oxidizers are mixed mechanically with the fuels. Many substances rich in fuel elements can be used as fuels. Those most often used at present include rubber, asphalt, organic resins, polyethylenes, etc. Substances containing metal fuel elements can also be used as solid propellant fuels. Solid propellant fuels are bonding agents which assure first, that the mixture will be uniform, and second, that solid grains with the required mechanical properties will be obtained after cooling (hot mixing for the asphalt-type propellants) or after vulcanizing (for the rubber-base propellants). Table 4.7 gives the composition and properties of some new types of solid propellants.

Table 4.6. Composition and Properties of Certain Rocket Powders

Powder	Composition: Substance	Composition: Content wt. %	Calorific value K_Q, kcal/kg	Combustion temperature at constant pressure, T°K	Derived powder force f_0, kg-m/kg	Specific thrust* $P_{sp\ 0}$, kg-sec/kg	Specific weight γ_{prop}, kg/liter
J. P.	Nitrocellulose (13.25% N_2)	52,2	1230	3170	100 000	230	1,60
	Nitroglycerine	43,0					
	Diethyl phthalate	3,0					
	Diphenyl amine	0,6					
	Potassium nitrate	1,1					
	Aniline black	0,1					
J. P. N	Nitrocellulose (13.25% N_2)	51,50	1230	3170	100 000	230	1,61
	Nitroglycerine	43,00					
	Diethyl phthalate	3,00					
	Centralite	1,00					
	Potassium sulfate	1,25					
	Carbon black	0,20					
	Candelilla wax	0,05					
Slow-burning powder	Nitrocellulose (12.2% N_2)	56,5	880	2330	76 000	(195)	—
	Nitroglycerine	28,0					
	Dinitrotoluene	11,0					
	Centralite	4,4					
	Candelilla wax	0,1					

*the specific thrust was computed from the average data for powder rocket engines

Table 4.7. Certain Properties of New Solid Propellants

Brand	Properties: Substance	Properties: Content, wt. %	Combustion temperature at constant pressure, T°K	Molecular weight of the gaseous products, μ	Derived force f_0, kg-m/kg	Specific thrust* $P_{sp\ 0}$, kg-sec/kg	Specific weight γ_{prop}, kg/liter
ALT-161	Potassium perchlorate $KClO_4$	76	2060	30	58 000	182	1,77
	Asphalt	16,8					
	Petroleum	7,2					
—	Ammonium perchlorate NH_4ClO_4	80	2670	25	90 000	200	1,75
	Rubber	20					
—	Ammonium nitrate NH_4NO_3	80	1720	22	66 000	225	1,55
	Rubber	18					
	Catalyst	2					

*the specific thrust values are approximate

These propellants have the following advantages: high specific weight, the ease of casting the grains, the relatively low cost, and the ability to burn at low pressures.

We can expect that solid propellants will be developed further. Their calorific value can be increased by using new fuel elements and changing the composition such that the oxygen content will assure maximum combustion of the propellant.

Figure 4.3 gives the classification of solid rocket propellants.

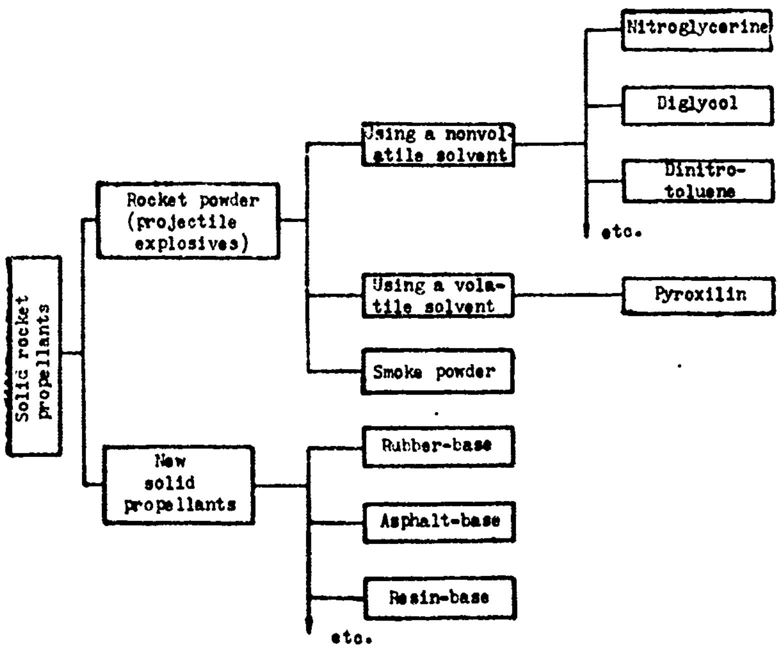

Fig. 4.3. Classification of solid rocket propellants.

The Development of New Propellants with Increased Efficiency

One of the most urgent requirements for the further improvement of rocket vehicles is an overall increase in the specific thrust of the engine. This can be done mainly by developing new propellants.

In the first stages of working with new propellants, beginning with the works of A. F. Tsander and Yu. V. Kondratyuk, particular attention was devoted to the values of the calorific value and the specific weight of the propellant, while there were no particular requirements placed on the thermodynamic properties of the combustion products.

Such an approach to the development of new propellants was dictated mainly by the fact that the calorific values of the ordinary fuel and oxidizer elements used are very low and many metal propellants* have a calorific value which is 2.5-3 times greater than present propellants (see Table 4.2). The specific weights of metal propellants are also high.

The lack, at that time, of the necessary theoretical and experimental data, describing the combustion products of metal propellants, resulted in the fact that these propellants could not be evaluated from the viewpoint of the thermodynamic properties of the combustion products. However, metal propellants, especially those which use

* By metal propellants we will arbitrarily mean all propellants which use fuels of inorganic origin, including boron and silicon.

oxygen as the oxidizer, have essential shortcomings in just this respect. These shortcomings can be demonstrated by using as our example the combustion of an oxygen + aluminum propellant. Because of the low heat capacity per unit weight, the temperature of the combustion products, even taking dissociation into account, reaches 5000°K. In this case, the heat expended on dissociation is 2600 kcal/kg, i.e., 67% of K_G. The high boiling point of the oxides of aluminum results in 21% (by weight) of the combustion products being in the liquid state in the thrust chamber. The great number of atoms in the molecules of the combustion product (Al_2O_3), and also the great amount of the combustion product in the condensed phase, cause the low efficiency of expansion. Therefore, despite the high K_G, the calculated specific thrust $P_{sp\ 0(100:1)}$ is only 230 units in all. This shows that many of the hopes of using metal propellants were unfounded.

Somewhat better results can be obtained by the combustion of metal propellants and fluorine. However, it will be very difficult to introduce the highly toxic and corrosive fluorine oxidizers into rocketry. Solid (metal) fuels can appear as melts, suspensions of finely ground metal in ordinary fuel, or organometallic compounds and metal hydrides, liquid under normal conditions. The physical properties of certain organometallic compounds and metal hydrides are given in Table 4.8 (the values in parentheses are those of the physicochemical constants which have not been reliably verified).

Table 4.8. Properties of Certain Organometallic Compounds and Metal Hydrides

Substance	Chemical formula	Melting point, °C	Boiling point, °C	Heat of formation kcal/g-mole	Specific weight, kg/liter
Pentaborane	B_5H_9	(50)	(60)	(0)	(0.64)
Diboranimine	B_2H_7N	−66	76	(−10)	(0.70)
Diethyl beryllium	$Be(C_2H_5)_2$	12	(200)	(−35)	(0.60)
Trisilane	Si_3H_8	−117	53	(−20)	(0.88)
Trisilane amine	$(SiH_3)_3N$	−106	52	(+10)	0.895

The disadvantages of metal fuels, detected during a more detailed study of them, has made it necessary at present to turn our attention to fuels with better thermodynamic properties of the combustion products. The substantial influence of these properties can be shown by using the propellant oxygen + ammonia as an example (see Table 4.5). This propellant, with a calorific value of only 1650 kcal/kg, can produce a ground thrust of 280-290 kg-sec/kg, approximately equal to the specific thrust of oxygen + kerosene. In this case, the temperature in the combustion chamber is only 3000°K (as opposed to 3600°K for oxygen + kerosene). Such results occur because of the great amount of diatomic gas with low molecular weight in the combustion products.

This is why, at present, there is much discussion of the use of nitrogen hydride fuels such as ammonia, hydrazine, and their derivatives in LFRE's.

Even better results in the use of nitrogen hydride fuels can be expected if fluorine oxidizers are used with them. In this case, with combustion, triatomic gases generally do not form, and the efficiency of combustion and expansion greatly increases. As has been shown, fluorine oxidizers can be advantageous in the combustion of ordinary hydrocarbon fuels.

Table 4.9. Substances with Positive Heats of Formation

Substance	Chemical formula	Heat of formation		Specific weight kg/liter
		kcal/g-mole	kcal/liter	
Atomic hydrogen	H	51,6	51 500	—
Atomic oxygen	O	58,6	3640	—
Ozone	O_3	35,0	730	1,71 at —183°C
Acetylene	C_2H_2	54,85	2 120	0,618 at —81,5°C
Tetranitromethane	$C(NO_2)_4$	8,80	45	1,65 at —15°C
Fluorine monoxide	OF_2	5,50	102	1,53 at —145°C
Hydrazine	N_2H_4	12,05	376	1,01 at —15°C
Dimethylhydrazine (unsymmetric)	$N_2H_2(CH_3)_2$	11,28	187	0,83 at —15°C

Other methods of increasing the efficiency of rocket propellants are considered in the literature on rocketry problems.

The calorific value of chemical propellants can be increased by using, as the components, substances with a high positive standard heat of formation ($\Delta H_T^0 > 0$).

Fuel or oxidizer elements in the atomic, not the standard molecular, state are examples of substances with positive heats of formation. In this case, tremendous amounts of energy are expended on the formation of the atomic elements (see Table 4.9). For example, 1 kg of atomic hydrogen accumulates 51,500 kcal of heat. However, as of now, there are no known cases where pure atomic hydrogen or oxygen has been produced or stored for any length of time.

Other examples are substances such as ozone, which requires 730 kcal/kg for its formation from molecular oxygen, and acetylene, which expends 2120 kcal/kg in forming from carbon and hydrogen.

These substances have the disadvantage that they are only slightly stable and tend to explode during storage and combustion, which makes it difficult to use them in engines. The only substance of this type, with a very low heat of formation, which is used in rocketry, is tetranitromethane $C(NO_2)_4$.

It will be considerably easier to conduct further experiments in finding rocket propellants with increased efficiency, since we now have more or less accurate tables of the thermodynamic properties of the combustion products. This will make it possible to select the most effective combinations of propellant components by calculation methods, without having to resort to experiments which are costly and difficult to carry out. However, as the calculations show, chemical propellants have a clearly perceptible specific thrust limit of the order of 380-400 kg-sec/kg.

3. The Energy of Nuclear Reactions and Its Use in Rocket Engines*

The Atomic Nucleus and Mass Defect

A further and very essential increase in specific thrust of a rocket engine can be achieved through the use of the energy of nuclear reactions, or so-called atomic energy.

Nuclear reactions, unlike chemical reactions, occur so as to change the structure of atoms and their nuclei.

If we schematize, to the maximum, the structure of an atom, we can say that its nucleus consists of two types of heavy particles having relatively large mass: protons and neutrons. The proton is a positively charged particle whose mass is practically equal to that of a hydrogen atom. The neutron, as its name implies, in general has no charge, and its mass is also close to that of the hydrogen atom. The total number of protons and neutrons contained in the nucleus is called the mass number of the nucleus.

In the formation of the nucleus of an atom from free protons and neutrons, just as in the case of the formation of molecules from atoms, energy is released. This energy release is due to the fact that the nuclei of elements are a stable system bound by intranuclear forces, the origin of which during nucleus formation should be accompanied by a decrease in the potential energy of the system.

The energy of nucleus formation is most conveniently calculated using Einstein's mass-energy equivalence principle. According to this principle, the energy and mass are connected by the relation

$$E = mc^2, \tag{4.7}$$

where E is the energy;

m is the mass;

c is the speed of light, equal to approximately 3×10^8 m/sec.

The energy equivalent to a mass of 1 kg-sec^2/m is 9×10^{16} kg-m. On conversion

* G. S. Sutherland. Recent Advances in Space Propulsion. ARS Journal, Vol. 29, No. 10, October 1959, pp. 698-705, and the bibliography to Sutherland's article.

to heat units per 1 kg of weight this gives

$$\frac{9 \times 10^{16}}{9.81 \times 427} = 2.15 \times 10^{13} \text{ kcal/kg.}$$

From the mass-energy equivalence principle it follows that the interaction of the elementary particles of a nucleus, forming a stable atom (with negative potential energy), is accompanied by a decrease in their mass compared with the mass of these same particles separated by a distance such as to exclude interaction between them. This mass decrease Δm in nuclear reactions is called mass defect and can be determined experimentally. From relation (4.7) the decrease in energy of the system, i.e., the value of the energy thus released, is

$$E = \Delta mc^2.$$

Nuclear Reactions

The formation energy and, consequently, the mass defects of different nuclei are different. Therefore, there are possible, in principle, nuclear reactions that lead to the formation of nuclei with greater mass defect than in the parent nucleus. Here the newly formed nuclei will be more stable. The energy released during the formation of the new nucleus is equivalent to the difference in mass defects of the newly-formed and the parent nuclei.

Nuclear reactions occur in various ways depending on the elements that participate. The nuclei with greatest mass defect are those of the atoms of elements in the middle of the periodic table. The nuclei of the atoms of these elements are the most stable.

Nuclear energy can be released in two ways. The first is decomposition of the nuclei of heavy elements (at the end of the periodic table) with the formation of lighter ones, which is accompanied by a high mass defect. The second is by synthesis of relatively heavy nuclei from the nuclei of light elements, which is also accompanied by a high mass defect. In the first case we are dealing with fission reactions of the nuclei of heavy elements; in the second — with so-called thermonuclear reactions.

The most easily accomplished fission reactions are those of the atomic nuclei of elements having large mass numbers. The higher the mass number of the heavy-element nucleus, the less stable the nucleus and the greater its tendency to decay. It is these heavy nuclei of elements that have the property of radioactivity, which is characterized by radiations occurring during nuclear fission.

However, to obtain a high constant energy yield, during fission reactions it is necessary to artificially excite the nuclei. The magnitude of the excitation energy in itself should be considerable; in addition, the exciting particle carrying the necessary energy should penetrate the nucleus. Charged particles are not very suited for intense nuclear excitation, since they lose much of their energy on overcoming the forces of the electric field of the nucleus itself. The particles most suited to nuclear excitation are those having no neutron charge.

To maintain the continuing fission reaction it is necessary to have an external emission source of neutrons having the necessary energy reserve, or the nuclear reaction itself should be the neutron source for excitation of ever-newer nuclei. The practical use of the energy of such nuclei became possible as a result of the realization of a nuclear fission chain reaction.

The energy released during fission of atomic nuclei can be liberated in explosive reactions (atomic bomb) as well in controlled chain reactions carried out in so-called atomic reactors.

As for thermonuclear reactions, there are more serious difficulties involved in their technical realization. Conditions for the occurrence of thermonuclear reactions require colossal temperatures, commensurate with intrastellar temperatures.

At the present time, of practical interest from the energy standpoint are thermonuclear explosions (hydrogen bomb). The high-temperature source for the excitation of such reactions is the explosive nuclear chain reaction (atomic explosion). Controlled thermonuclear reactions with continuous energy release are in the stage of preliminary scientific and research development.

Each nuclear reaction has its characteristic energy effect. The measure of this is the mass defect which is expressed in percents δ of the initial mass of the active substance. The value of δ is very low; it is less than the relative defect during the formation of nuclei of elements, since it is defined as the difference in the mass defects of nuclei produced as a result of the reaction and the original nuclei. For example, for the fission of uranium nuclei U_{92}^{235}* the value of δ is 0.000731.

Thermonuclear reactions are characterized by a higher mass defect. For the reaction of the formation of helium from lithium and hydrogen $Li_3^7 + H_1^1 \rightarrow 2He_2^4$ the value $\delta = 0.00232$; for the conversion of hydrogen into helium $4H_1^1 \rightarrow He_2^4$, $\delta = 0.00715$.

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* In the notation for nuclear reactions we use conventional designations that show the structure of the nucleus. The superscript indicates the mass number, and the subscript indicates the number of protons in the nucleus, i.e., the number of the element in the periodic table.

However, even with such insignificant mass defects, in view of the enormous value of the energy per unit mass the energy yield K_G will be very high. In accordance with (4.7) the value $K_G = 2.15 \times 10^{13}\ \delta$ kcal/kg. For the above-indicated reactions we will have:

Reaction	K_G, kcal/kg
Uranium fission	$1{,}57 \cdot 10^{10}$
$Li_3^7 + H_1^1 \rightarrow 2He_2^4$	$5{,}00 \cdot 10^{10}$
$4H_1^1 \rightarrow He_2^4$	$1{,}54 \cdot 10^{11}$

For nuclear power engineering of the future, including the future of rocketry, of particular interest is the thermonuclear reaction $H_1^2 + H_1^2 \rightarrow He_2^4$. In this reaction the parent substance is the heavy hydrogen isotope H_1^2, i.e., hydrogen containing in its nucleus, in addition to a proton, one neutron and therefore having an atomic weight which is twice that of ordinary hydrogen. This isotope of hydrogen is called heavy hydrogen, or deuterium. The mass defect of this reaction is quite large — it reaches 0.0059, i.e., $K_G = 1.265 \times 10^{11}$ kcal/kg. At the same time, deuterium is quite widespread in nature. One kilogram of ordinary water contains 0.037 g of deuterium with an energy content of 460×10^4 kcal. Thus, 1 kg of water, according to the energy content of the deuterium in it, is equivalent to 450 kg of petroleum. At present, deuterium is extracted from ordinary hydrogen quite inexpensively. The cost of producing 1 kg of deuterium is only 1% of the cost of an amount of coal having an equivalent energy content.

Nuclear Rocket Engines

The use of the energy of nuclear reactions in rocket engines is possible in various ways. In the so-called nuclear rocket engines (NRE's) the energy of nuclear reactions, released in a nuclear reactor, is absorbed by some working substance in the form of the energy of thermal motion of particles of the working substance. The subsequent conversion of this energy into the kinetic energy of the discarded particles and the creation of thrust is accomplished, as in ordinary rocket engines, in a supersonic nozzle.

As the working substance of an NRE we should use a substance with good thermodynamic properties, i.e., as was shown above, a substance with a small number of atoms per molecule and as low a molecular weight as possible. The best such substance (of those that can be used) is hydrogen H_2; next come ammonia NH_3, water H_2O, and hydrazine N_2H_4. The weight heat capacity of these substances is also relatively high.

Table 4.10 gives data on the energy yield K_G, in kcal/kg, accumulated in 1 kg of working substance consisting of various substances at temperatures of 4000 and 6000°K. The same table gives the amount of active substance G_a (U_{92}^{235} or Pu_{94}^{239}) necessary for heating 1 kg of working substance to the appropriate temperatures.

Table 4.10.

Working substance	Maximum heating temperature 4000°K								Maximum heating temperature 6000°K							
	Pressure ratio 100:1				Pressure ratio 50:1				Pressure ratio 100:1				Pressure ratio 50:1			
	K_G, kcal/kg	G_a, kg/kg	$P_{sp\,0}$ kg-sec/kg	$P_{sp\,0\,V}$ kg-sec/liter	K_G, kcal/kg	G_a, kg/kg	$P_{sp\,0}$ kg-sec/kg	$P_{sp\,0\,V}$ kg-sec/liter	K_G, kcal/kg	G_a, kg/kg	$P_{sp\,0}$ kg-sec/kg	$P_{sp\,0\,V}$ kg-sec/liter	K_G, kcal/kg	G_a, kg/kg	$P_{sp\,0}$ kg-sec/kg	$P_{sp\,0\,V}$ kg-sec/liter
Hydrogen	20 662	$1{,}31\cdot10^{-6}$	1095	77	22 365	$1{,}42\cdot10^{-6}$	1063	75	60 931	$3{,}87\cdot10^{-6}$	1550	109	67 802	$4{,}32\cdot10^{-6}$	1544	108
Oxygen	1 430	$9{,}10\cdot10^{-8}$	277	316	1 551	$9{,}90\cdot10^{-8}$	266	303	4 497	$2{,}87\cdot10^{-7}$	399	455	4 914	$3{,}12\cdot10^{-7}$	381	433
Nitrogen	1 233	$7{,}86\cdot10^{-8}$	267	216	1 241	$7{,}90\cdot10^{-8}$	256	207	2 620	$1{,}67\cdot10^{-7}$	372	301	2 925	$1{,}86\cdot10^{-7}$	361	287
Water	4 459	$2{,}84\cdot10^{-7}$	409	409	4 989	$3{,}18\cdot10^{-7}$	395	395	14 040	$8{,}92\cdot10^{-7}$	683	683	15 356	$9{,}80\cdot10^{-7}$	663	663
Ammonia	5 474	$3{,}48\cdot10^{-7}$	513	350	5 850	$3{,}72\cdot10^{-7}$	517	352	14 763	$9{,}50\cdot10^{-7}$	779	530	16 379	$1{,}04\cdot10^{-6}$	759	517
Hydrazine	3 175	$2{,}03\cdot10^{-7}$	459	463	3 350	$2{,}13\cdot10^{-7}$	457	462	10 300	$6{,}56\cdot10^{-7}$	685	692	11 566	$7{,}35\cdot10^{-7}$	688	696

REMARKS: 1. These tables are compiled using identical nozzle-exit pressure, equal to 1 kg/cm².
2. The heat expended on heating the working substance K_G kcal/kg is specified taking into account the heat of evaporation of the liquid working substance.
3. The specific thrusts $P_{sp\,0\,V}$ kg-sec/liter are calculated for the specific weight of the working substance at the normal boiling point.

As these data indicate, the consumption of active substance – the nuclear energy carrier – is very low, so that the thermodynamic properties of the substance participating in the expansion process, and the specific thrust developed by the nuclear fuel in rocket engines, are determined entirely by the type of working substance, the maximum temperature of its heating, and the pressure drop in the rocket engine.

The values of the specific thrust achieved for various working substances at different temperatures and pressure drops are shown in Table 4.10. The specific thrusts are calculated taking into account dissociation during heating of the working substance, recombination of the combustion products during flow through the nozzle, and expansion from maximum pressure to pressure at the nozzle edge, equal to 1 kg/cm² during static tests of the engine. The data on the value of the specific thrust attest to the fact that at temperatures restricted to 4000°K, high values of specific thrust can be attained when hydrogen is used as the working substance. However, the specific weight of liquid hydrogen at the boiling point is 0.07 kg/liter, which greatly impairs the weight qualities of the rocket. Higher specific thrusts than those developed using chemical fuels can be obtained with working substances having satisfactory specific weights and containing a relatively large amount of hydrogen, viz., liquid ammonia, hydrazine, and water.

Naturally, an increase of the heating temperature of the working substance to 6000°K, i.e., to a temperature considerably exceeding the present design capabilities of rocket engines, leads to a sharp increase of specific thrust.

The task of creating an NRE is considerably more difficult than the problem of producing stationary atomic power plants or other types of engines (in particular, ramjets) in which the use of nuclear fuel is advantageous even without the significant increase in temperatures of the working substance which is standard for these engines.

The primary problem that arises in the creation of a rocket engine is the organization of heat exchange during which the energy of the nuclear fuel will be imparted to the working substance.

Another problem pertaining to the use of nuclear energy is that for a nuclear fission reaction there must be a certain minimum (critical) mass. This requirement is explained by the fact that the dimensions of the atomic nucleus are very small (the cross section of the nucleus is approximately equal to 10^{-24} cm^2); to assure a sufficiently high probability of the collision of a neutron with the nucleus there must be a considerable path length which the neutron must traverse in the active material.

The value of the critical mass and the corresponding critical volume depends on the form of the active material and the conditions under which nuclear reactions take place.

Thus, the reserve of nuclear fuel in a rocket engine is in no way determined by the meager amounts shown in Table 4.10 and calculated on the basis of the energy balance. This reserve is determined at least by the value of the critical mass. It is also to be expected that the dimensions and weight of the nuclear reactor must additionally increase in order to obtain the required heat-transfer surface.

In NRE's it is possible, in principle, to use three types of reactors: reactors with solid, gaseous, or liquid (molten) nuclear fuel.

The simplest type of NRE with a solid nuclear fuel reactor is shown in Fig. 4.4. The liquid working substance (passive mass) is located in tank 1. By pump 2 it is fed through interjacket cooling space 4 to the engine chamber. The flow of the

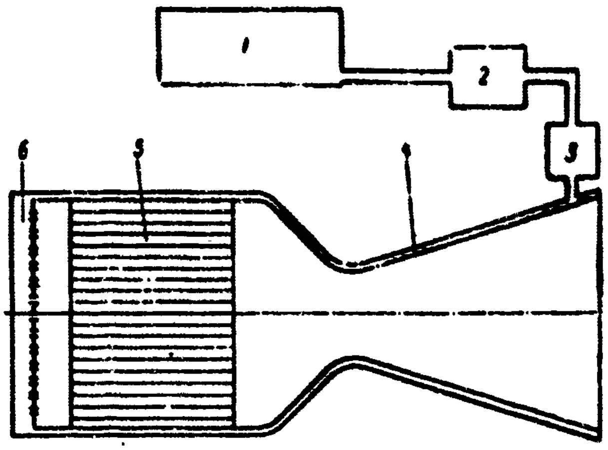


Fig. 4.4. Diagram of an NRE with a packet of solid nuclear fuel. 1) tank with liquid cooling substance; 2) pump; 3) device to control feed of working substance; 4) cooling loop; 5) packet of solid nuclear fuel; 6) head with injectors.

working substance is metered through control device 3.

As it passes through head 6 of the chamber the working substance comes into contact with packet 5 of solid fuel elements.

The fuel elements of such an NRE consist of fissionable uranium in a can made of a material having a high melting temperature. The maximum heating temperature of the working substance is limited by the melting temperature of the can materials. The highest-melting materials are graphite (sublimation point near 4000°), tantalum carbide (melting point near 3900°C), and tungsten (melting point near 3400°C). But even when using such materials, the heating temperature of the working substance, taking into account the temperature drop during heat transfer and the need for a certain reserve, is limited to, e.g., 2800°K when using tungsten. At these heating temperatures the NRE can give a higher specific thrust than LRE's using chemical fuels only when hydrogen is used as the working substance.

According to data in the foreign press, in several years there will be an NRE using hydrogen and having a specific thrust of about 800 kg-sec/kg and a weight (reactor with shielding, feed and nozzle system) of the order of 35-50 kg per one ton of thrust.

Higher working-substance heating temperature and, consequently, higher specific thrusts can be achieved by using in the NRE a reactor using gaseous nuclear fuel.

In this case, to the nuclear reactor are fed, from separate compartments, working substance and nuclear fuel which are mixed in the reactor. When the critical stage is reached in the reactor, nuclear reactions begin to occur. The heat of the nuclear reactions is transmitted directly to the working substance. Therefore, the magnitude of the developing temperatures is not restricted to the melting temperature of the fuel-element materials, as can occur in solid-fuel nuclear reactors, and can attain very high values limited by the conditions of cooling of the reactor walls which will heat mainly due to the extremely intense thermal radiation. In this case the nuclear fuel in the reactor is in the gaseous state. Numerous difficulties will arise in realizing such a logically possible NRE according to data in the cited source.

The basic difficulties are: large dimensions of the reactor and the need for restricting the carry-over of nuclear fuel by the flow of working substance into the nozzle and beyond the engine. This latter is conceivable, e.g., by creating in the reactor core a strong magnetic field to restrict the ionized heavy atoms of nuclear fuel, or by creating a field of centrifugal forces that quite well restrict the heavy atoms of nuclear fuel and do not prevent light atoms of the working substance from reaching the core. As for the critical dimensions of a gaseous-fuel nuclear reactor, previous calculations yielded dimensions which are completely impermissible in an NRE, viz.: a spherical reactor should have had a radius of 120 m with a pressure in the reactor of 100 kg/cm^2. Later calculations, taking into account the use of a neutron reflector in the form of a layer of heavy water, give a critical diameter of 1 m for a spherical reactor, with a pressure in the reactor of 70 kg/cm^2 and a temperature of 17,000°K. The amount of U^{235} necessary for the operation of such a reactor is about 2 kg according to these calculations.

An engine with a reactor using liquid (molten) nuclear fuel is a possible intermediate design for an NRE.

An analysis design of a solid-fuel NRE shows that the problem of using nuclear energy in rocket engines consists mainly in solving the problems of heat transfer, and also in creating small and relatively light reactors capable of operating at high temperatures. Successful solution of these problems could lead to the creation of combination rocket engines with preheating of the chemical fuel components in the nuclear reactor. A typical design of such an engine is shown in Fig. 4.5.

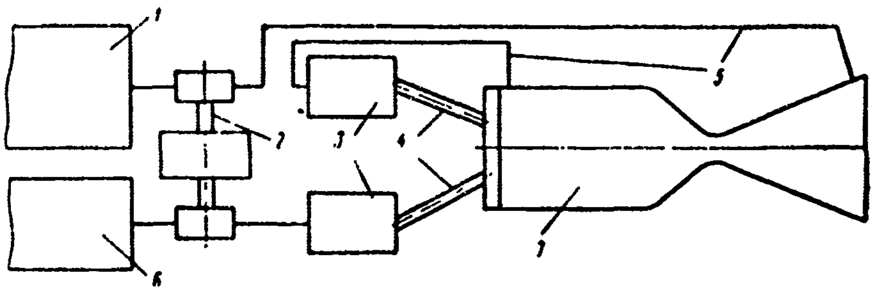


Fig. 4.5. Design of a liquid chemical fuel rocket engine with heating of the components in a nuclear reactor. 1) tank with oxidizer; 2) pump; 3) nuclear reactors for heating and vaporizing the chemical fuel components; 4) pipes to feed vapors of the components into the chamber; 5) pipes for circulation of liquid oxidizer in the chamber cooling system; 6) fuel tank; 7) engine chamber.

In these engines the chemical fuel components, located in tanks 1 and 6, are heated in nuclear reactors 3 to as high a temperature as possible before being sent to the combustion chamber. In this case the fuel components absorb much heat, not only as a result of heating and vaporization, but also as a result of the decomposition of the complex components (e.g., HNO_3, C_2H_5OH, kerosene, etc.) into simpler compounds (CH_4, C_2H_2, COOH, etc.). In such mixed types of engines it is also possible to use other fuels, e.g., water- and metal-base fuels. The vaporized and heated components enter engine chamber 7 where their combustion and expansion occur. As a result of preheating of the components and their subsequent combustion, there develops a specific thrust which is higher than that which would be produced from unheated components. The engine chamber is cooled by one of the components before it enters the nuclear reactor.

Since as a result of preheating of the components there is a considerable rise in temperature in the chamber, the combustion products of the chemical fuels used in such a nuclear rocket engine can, when the fuel components are correctly selected, have good thermodynamic properties.

To limit the maximum temperature in the engine chamber, the fuel can be fed not to the head of the combustion chamber but along the chamber nozzle in such a way that due to chemical reactions the thermodynamic temperature will nowhere exceed a given value. In particular, by selection of the law of component feed along the nozzle we can achieve constancy of the thermodynamic temperature of the gases as they move in the nozzle, i.e., so-called isothermal expansion in the nozzle.

Ion and Photon Rocket Engines

At present, in the press, there is intensified discussion of the possibilities of creating rockets having engines of new, as yet uncreated, types, viz., with plasma, ion, and photon rocket engines. Such engines will permit the obtaining of prospectively high specific thrusts. The idea of creating these engines can become feasible only after study and assimilation of nuclear reactions. Let us examine some proposed designs of these engines and their power possibilities.

In a plasma rocket engine the working substance is heated, with the aid of a powerful electric arc having high current density, to a very high temperature (greater than 10,000°K). At this temperature there occurs practically complete ionization of all particles appearing in the operating region of the arc, i.e., the creation of so-called plasma, for which such an engine is named. The working substance is expanded and thrust is created in the nozzle. According to the foreign press we can expect from such engines a specific thrust of 1000-1200 kg-sec/kg with 50% efficiency. The creation of such an engine involves a great many difficulties associated with high temperature and the erosion of the electrodes, cooling, etc. In connection with the fact that in a plasma rocket engine electrical energy is used to create thrust, this engine belongs to the group of so-called electrical rocket engines.

The ion engine is based on the fact that the acceleration of ejected particles, necessary to create thrust, occurs by the action on the particles of an electrostatic or electromagnetic field. For this, the accelerated particles should have an electric charge, i.e., they should be ionized. From this we get the name "ion engine." In principle, the ion engine should consist of three basic parts (Fig. 4.6): charged particle generator 2, charged particle accelerator 4 with winding 5, and power plant 3 to generate the electric power necessary to create and accelerate the ionized particles. As can be seen from the description, an ion rocket engine can also be classified as an electrical rocket engine.

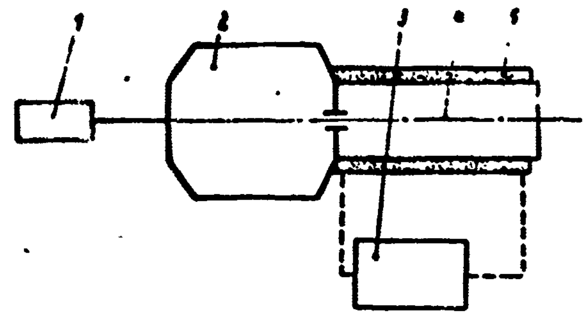


Fig. 4.6. Diagram of an ion rocket engine. 1) tank with working substance; 2) charged particle-ion generator; 3) atomic power plant; 4) charged particle accelerator; 5) electrical winding to create an accelerating electromagnetic field.

The charged particles necessary for the operation of the ion engine can be produced in various ways, the best known of which is the ionization of atoms of certain substances during their heating; in this case, the substances to be ionized should have low ionization energy. An example of such a substance is cesium, for which the ionization energy is only 4 ev in all (approximately 85 kcal/g-mole or 650 kcal/kg). Ionization of cesium (the combining of an electron with a cesium atom) is accomplished during heating of its vapors on the surface of an incandescent tungsten grid. According to the foreign press, at a grid temperature of about 1500°K there is almost complete ionization of cesium atoms at an ion current density of up to 100 ma/cm^2.

Charged particles produced in some way or other enter accelerator 4 where they are accelerated by an electromagnetic field created by winding 5 or by a system of flat electrodes that accelerate and focus the ion flux (not shown in Fig. 4.6).

During the outflow of negative cesium ions, there is formed in the accelerator and generator a positive space charge that prevents acceleration of the ions in the accelerator. It should be neutralized by the intense isolation of electrons by means of some special device.

A change in the momentum of the accelerated particles leads to the occurrence of a reactive force applied to the winding or to the flat electrodes of the accelerator; this is transmitted to the engine housing.

The kinetic energy of a charged particle passing a potential difference of U volts is ieU ev, where i is the number of electron charges of the ion. In thermal units, the kinetic energy of such particles is 23.07iU kcal/g-mole.

Let us make some elementary calculations which will allow us to evaluate certain basic properties of an ion rocket engine. Calculations will start from the fact that in an ion generator all particles are singly ionized ($i = 1$). In this case the kinetic energy of a mole of ionized gas having velocity w_a at the outlet is

$$\frac{mw_a^2}{2} = 23{,}07\,U \text{ kcal/g-mole.}$$

Considering further that there are ionized, as a rule, atomic gases, the atomic weight of which we designate by A, and that the mass m of one mole of ions is $\frac{A \times 10^{-3}}{9.81}$ kg-sec^2/m, let us calculate the ion escape velocity:

$$w_a^2 = 2 \cdot 23{,}07 \cdot 427\,U \frac{1}{m} = 2 \cdot 23{,}07 \cdot 427 \frac{U}{A} 9{,}81 \cdot 10^3 = 1{,}93 \cdot 10^8 \frac{U}{A}$$

or

$$w_a = 13{,}9 \cdot 10^3 \sqrt{\frac{U}{A}} \text{ m/sec} = 13{,}9 \sqrt{\frac{U}{A}} \text{ km/sec.}$$

As is evident from the obtained expression, the ion escape velocity is proportional to $\sqrt{U/A}$.

The specific thrust of an ion rocket engine is, as usual,

$$P_{sp} = \frac{w_a}{g_0} = 1{,}415 \cdot 10^3 \sqrt{\frac{U}{A}} \text{ kg-sec/kg.}$$

To obtain higher specific thrusts it is necessary either to increase the voltage of the electrical field created in the accelerator or to decrease the atomic weight of the ions used.

The total thrust of an ion rocket engine will be

$$P = P_{sp}G,$$

where G is the ion flow per second.

Let us now determine the power consumption necessary to impart a velocity w_a to 1 kg of ions. This is:

$$L = \frac{w_a^2}{2g_0} = \frac{\left(13{,}9 \cdot 10^3 \sqrt{\frac{U}{A}}\right)^2}{2g_0} \approx \frac{U}{A} 10^7 \text{ kg-m/kg.}$$

With ion flow G kg/sec the required power N is

$$N = \frac{GL}{102} \text{ kw,}$$

while with an ion flow of 1 kg/sec

$$N' \approx 10^5 \frac{U}{A} \text{ kw/kg/sec.}$$

The total power expended on the operation of an ion engine should be greater since first, a certain amount of energy must be expended on ionizing the working substance, and second, part of the work of the electromagnetic field that accelerates the ions will be expended on an increase in the velocity of random thermal motion of the ions, which is useless for the creation of thrust.

If we designate by ζ the coefficient that accounts for the additional power expenditure due to the above-enumerated reasons, the total power expended in accelerating 1 kg of ions in 1 sec will be

$$N' = \zeta 10^6 \frac{U}{A} \text{ kw/kg/sec.}$$

Now let us determine the power necessary for an ion engine to create thrust P in kilograms. With a specific thrust P_{sp} the expenditure of working substance is

$$G = \frac{P}{P_{sp}} = \frac{P}{1{,}415 \cdot 10^3 \sqrt{\frac{U}{A}}}$$

while the expended power

$$N = GN' = \zeta 10^6 \frac{U}{A} \frac{P}{1{,}415 \cdot 10^3 \sqrt{\frac{U}{A}}} = 70P\sqrt{\frac{U}{A}}\,\zeta \text{ kw.}$$

The power used to create 1 kg of thrust will be

$$N'' = 70\sqrt{\frac{U}{A}}\,\zeta \text{ kw/kg,}$$

while if we do not take into account the energy expended on ion generation, and the above-indicated losses (i.e., if we consider $\zeta = 1$), then

$$N'' = 70\sqrt{\frac{U}{A}} \text{ kw/kg.}$$

The power spent on creating 1 kg of thrust is directly proportional to $\sqrt{U/A}$ and, consequently, proportional to the specific thrust of the engine. Thus, the lower expenditure of working substance which will occur with an increase in specific thrust requires increased power of the power plant of an ion rocket engine.

To evaluate the basic properties of ion engines, let us make certain elementary calculations using the above-derived formulas without taking into account the energy expended on ionization and the losses on increasing the energy of thermal motion of the particles. The results of these calculations are given in Table 4.11.

The derived data attest to the fact that the type of working substance, particularly its atomic weight, influences principally only the voltage of the required current. The energy consumption (and, consequently, the power of the power plant) and the mass flow of the working substance are direct functions of the specific thrust of the engine. With a 10-fold increase in specific thrust there is a 10-fold decrease in the consumption of the working substance, and a 10-fold increase in the expended power.

Table 4.11.

Ion engine parameters	Working substance – singly ionized hydrogen atoms ($i = 1$, $A = 1$)		Working substance – singly ionized cesium atoms ($i = 1$, $A = 133$)	
	$P_{sp} = 1000$ kg-sec/kg	$P_{sp} = 10{,}000$ kg-sec/kg	$P_{sp} = 1000$ kg-sec/kg	$P_{sp} = 10{,}000$ kg-sec/kg
Necessary electrical voltage of the field U, volts	0,5	50	66,5	6 650
Energy N' expended on acceleration of 1 kg of ions per second, kw	5 000	500 000	5 000	500 000
Energy N'' which must be expended to produce thrust of 1 kg, kw	50	500	50	500
Expenditure of working substance G necessary to create a thrust of 100 kg, kg/sec	0,1	0,01	0,1	0,01
Necessary energy of power plant N, used to accelerate ions in engine with a thrust of 100 kg, kw	5 000	50 000	5 000	50 000

Consequently, the basic properties of an ion rocket engine are such that it is most suitable for the creation of low absolute thrusts (kg units) with very high specific thrusts or with very high exhaust velocities.

Such engines are desirable only for controlled space flights of the future which must be accomplished at very high speeds. Only under such flight conditions will rocket engines with high outflow velocity operate with sufficiently high external efficiency (see Section 3, Chapter I).

The source of electrical energy for generation and acceleration of ions can be an atomic or solar power plant. The production of an electric current from nuclear or solar energy can be realized through a closed thermodynamic loop, as a result of which a turbine activates some type of electric generator, or by the direct conversion of thermal energy into electrical energy using thermoelectric batteries. According to data in the cited article, at considerable energies (of the order of tens of kilowatts and higher), devices with nuclear reactors and turboelectric generators will have the least specific weight.

Figure 4.7 shows the diagram of the photon engine. It is depicted extremely simplified, since we cannot discuss here the ways of embodying this design in an actual model. In principle, however, the photon engine consists of chamber 1 in which any working substance is heated to a very high temperature – of the order of tens and hundreds of millions (10^7-10^8) degrees Kelvin – due to the occurrence of a thermonuclear reaction.

Such a thermonuclear reaction can occur as a result of the heating of the working substance using a powerful highly-concentrated electrical discharge or as a result of other nuclear processes occurring with a high mass defect and, consequently, with high energy release. Naturally, the chamber walls 1 by some means, e.g., due to winding 2 that creates the electromagnetic field which concentrates the zone of the nuclear reaction in the center of the chamber, should be shielded from the effects of high temperature.

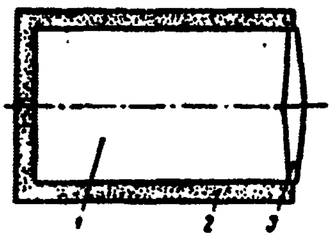


Fig. 4.7. Schematic of a photon engine.
1) chamber in which thermonuclear reaction occurs; 2) chamber wall shielding; 3) transparent chamber wall.

In chamber 1 one of the walls, e.g., wall 3, is transparent, capable of passing light. The flux of light electromagnetic waves or, in other words, the photon flux moving at the speed of light (i.e., with a velocity of $\sim 3 \times 10^5$ km/sec) creates a reactive force and, consequently, engine thrust.

The reaction force caused by light radiation, i.e., by the ejection of photons, is the reason for the occurrence of light pressure which was detected theoretically and experimentally by Academician P. N. Lebedev as early as 1900.

This reaction force can be calculated as follows.

If the momentum of the photon passing through the transparent chamber wall is designated by L, the total momentum carried off by all photons N radiated by the body per unit time will be LN.

The number of radiated photons N can be expressed in terms of the total energy of the luminous radiation Φ and the energy of a single photon ε. Obviously,

$$N = \Phi/\varepsilon.$$

Then the total momentum carried off by the photons per unit time will be

$$LN = \Phi(L/\varepsilon).$$

Since the reactive force P is equal to the per-second momentum of the ejected mass, then

$$P = LN = \Phi(L/\varepsilon).$$

If photons are radiated uniformly from the entire surface, it is convenient to express the thrust in terms of the magnitude of the radiation pressure p_{rad}:

$$p_{rad} = \frac{P}{S} = \frac{\Phi}{S}\frac{L}{\varepsilon}.$$

The expressions for thrust of a photon engine and radiation pressure contain the ratio of the momentum of a photon to its energy L/ε, which can be found from the following. The increase in kinetic energy of a moving body is associated with the increase in the momentum by the following relationship

$$d(mw^2/2) = wd(mw).$$

Applying this expression to a flying photon, and considering that the speed of the photon is constant and equal to the speed of light $\underline{c}$, we obtain the differential equation

$$d\varepsilon = c\ dL,$$

from which, after integration, we find

$$\varepsilon = cL + C,$$

where C is the integration constant. With kinetic energy $\varepsilon = 0$, and the momentum $L = 0$. From this it follows that C also equals 0. Then,

$$\varepsilon = cL \text{ and } L/\varepsilon = 1/c.$$

Using the obtained relationship, we obtain the radiation pressure:

$$p_{rad} = (\Phi/S)(L/\varepsilon) = I/c,$$

where $I = \Phi/S$ is the specific radiation flux, i.e., the flux radiated by a body per unit time from a unit of surface. If we assume that incandescent gases — products of the thermonuclear reaction — are an absolutely black body, to calculate the specific radiation flux $I = I_0$ we can use the Stefan-Boltzmann equation

$$I_0 = 5.67 \times 10^{-5} T^4 \text{ erg/sec-cm}^2.$$

The radiation pressure of an absolutely black body $p_{rad\ 0}$ is

$$p_{rad\ 0} = \frac{5.67 \times 10^{-5} T^4}{2.9979 \times 10^{10}} \text{ dyne/cm}^2 = 1.927 \times 10^{-21} T^4 \text{ kg/cm}^2.$$

As can be seen from this expression, the radiation pressure $p_{rad\ 0}$ is negligibly small at low temperatures, but can attain very high values at the very high temperatures which are actually attainable at present because of the mastering of thermonuclear reactions. Thus, if in the zone of a thermonuclear reaction there is achieved a temperature of one million degrees, the radiation pressure of an absolutely black body is about 2000 kg/cm^2. This makes it possible to attain a thrust equal to 2000 kg per 1 cm^2 area of radiation of the photon engine.

If the properties of the products of the thermonuclear reaction are actually analogous to those of an absolutely black body, the radiation pressure can actually guarantee the required thrust for a photon rocket engine. However, many authors consider that gas heated to high temperatures has low density and does not contain complex (polyatomic) molecules, because of which it cannot possess high emissivity.

These concepts have been partially confirmed by calculations and experiments through the realization of controllable thermonuclear reactors; it is expected that when hydrogen atoms are heated in them to a temperature of 50×10^6 °K their radiation intensity will correspond to that of an absolutely black body heated only to 5000°K. The light pressure with such radiation intensity is only 1.2×10^{-6} kg/cm^2, i.e., a very small magnitude.

Thus, the question of the possibility of creating a photon rocket engine must still be considered unresolved. Just as problematic are the methods for utilizing a photon engine, since it can be used effectively on rocket vehicles only at flight speeds close to the speed of light.

Note: All data on propellants and their components given in Chapter IV are taken from foreign sources.

CHAPTER V

PROCESSES IN A ROCKET ENGINE COMBUSTION CHAMBER

1. Burning in a Liquid-Fuel Rocket Engine

Preparatory Processes and Burning of Fuel in the Chamber of a Liquid-Fuel Rocket Engine

In the combustion chamber of a rocket engine chemical reactions of burning occur, as a result of which a significant part of the fuel's chemical energy passes into heat and is expended on heating of combustion products.

There exist many proofs that fuel components, before reacting, have to be evaporated, and that burning, thus, occurs in a gas phase. Concurrently, there are examples of reactions occurring in liquid phase, as, for instance, in the case of burning of spontaneously inflammable fuels and especially at the moment of their ignition. It is possible to assume, however, that also in this case a large part of the fuel reacts only after evaporation.

In any case, a chemical reaction is possible only during contact of fuel and oxidizer molecules in a proportion necessary for burning. Therefore, in order to carry out burning of liquid fuels, it is necessary in the beginning to form as uniform a mixture of fuel and oxidizer vapors as possible.

The process of creating such a mixture is called carburetion.

Physical completeness of burning of fuel in the chamber is determined by the fractions of reacting fuel and oxidizer molecules; it depends on the quality of preliminary mixing of fuel in the engine and on the time which is alloted to the burning process. Besides the time necessary for complete burning will be less, the better the preparatory process of carburetion.

Since the time of burning in the engine with the given parameters is determined by dimensions (volume) of the combustion chamber, then improvement of carburetion allows a decrease in its dimensions. To decrease the dimensions of the combustion chamber is profitable, since as a result the weight of the engine decreases, and the problem of its cooling is simplified.

For appraisal of the quality of the carburetion process two criteria are used. The first criterion assumes that complete burning of fuel is possible only when in every point of the chamber the necessary relationship between the fuel components exists. When there is a shortage of oxidizer in any section of the chamber, the fuel will not be used completely, but with a shortage of fuel a certain quantity of the oxidizer is unused.

Therefore, the carburetion process should occur in such a way that in every point of the chamber the necessary relationship between fuel and oxidizer is established, or, so to speak, the distribution of the relationship of components should be the most uniform. As a limit, in ideal carburetion at every point there should be that relationship between oxidizer and fuel which is selected for the engine as a whole. This requirement becomes especially important because products of combustions, forming in different points of the combustion chamber, then flow through chamber nozzle in the form of separate streams. These streams mix very weakly. Thus, if in formation of the mixture of fuel and oxidizer an equal distribution of the relationship of fuel components is not attained, then the irregularity appearing is significantly maintained in the expiration process, leading to physical incompleteness of burning and loss of chemical energy.

The second criterion assumes that in any elementary volume of the chamber the burning process should occur with equal intensity, for which through every elementary cross sectional area of the chamber an identical quantity of fuel or its combustion products should pass in a unit of time.

The quantity of fuel passing per second through an elementary cross sectional area is called the flow rate pressure, is designated by r and is expressed in g/cm^2 sec or in kg/m^2 sec.

Consequently, it is possible to say that the carburetion process should ensure as equal a distribution of flow rate pressure on the chamber cross section as possible. As a limit, in ideal carburetion in every point of the chamber cross section the flow rate pressure should be equal to the average flow rate pressure $r = G/S_{c.c.}$, where G — weight of fuel per second flow rate; $S_{c.c.}$ — cross section of the combustion chamber.

The jet stream of gas is of importance in the engine chamber. Due to weak mixing of gas streams, the time during which fuel and combustion products pass through the combustion chamber, for streams with different flow rate pressures, is different. A stream with low flow rate pressure, having a low speed, is in the combustion chamber for a long time, even longer than is necessary for completion of the burning process. But then, the stream with a large flow rate pressure passes through the chamber very fast, and the time which it is in the combustion chamber turns out to be insufficient for completion of chemical reactions. This leads to additional losses of chemical energy of the fuel.

The cap of the combustion chamber is the organ of carburetion.

In the cap of the chamber atomizers are disposed, injecting fuel and oxidizer in fine atomized form into the chamber. Mixing of fuel components can start still in the liquid phase by means of merging of drops and mutual dissolution of fuel and oxidizer, but the basic share of the mixture will be formed after evaporation of the drops and mixing of components vapors.

Evaporation and mixing of components is connected with the phenomenon of transfer of particles from one point of the chamber to another, i.e., with diffusion and convection flows in the combustion chamber. Besides this, for evaporation and subsequent heating of vapors to the temperature at which the chemical reactions of burning can start and continue, passage of heat from the hotter regions in the chamber is required. Processes of particle transfer and heat transfer occur simultaneously and are intimately connected.

Law of mechanical and thermal motion, which the process of carburetion obeys, are extraordinarily complicated. Therefore, below, considering carburetion, we will be limited only to certain qualitative conclusions.

It is absolutely clear that it is easier to satisfy the requirements for the carburetion process, the smaller the drops of atomized fuel, the more uniform the drops are distributed in the atomizer jet and the more atomizers with small flow rate are located on the cap. The dimensions of the drops and uniformity of their distribution are determined mainly by the type of atomizers.

In liquid-fuel rocket engines two types of atomizers are used,

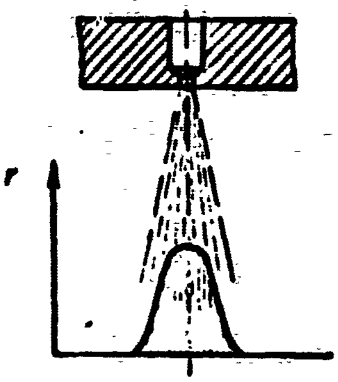


Fig. 5.1. Jet burner and the law of flow rate pressure distribution in a section of the jet of the component.

jet and centrifugal.

Jet atomizers (Fig. 5.1) constitute simple cylindrical holes of small diameter, ensuring the flow of thin streams of liquid. The streams break up into drops due to friction of liquid against gas in the chamber.

A narrow, long jet of the component is characteristic for jet burners; disintegration of the stream and formation of drops starts quite far from the atomizer. Flow rate pressure distribution in the jet of the component turns out to be very nonuniform: the greatest flow rate pressure is in the axis of the jet, but in the periphery it decreases rapidly (see Fig. 5.1). Since the jet for jet atomizers is narrow, mixing of different components starts far from the cap, and flow rate pressure distribution of components in a cross section of the combustion chamber is not sufficiently uniform after displacement of fuel.

For removal of this deficiency, jet burners are installed sometimes with the iuea that two or more streams, ensuing from different atomizers, cross in one point in the chamber (Fig. 5.2). Also, it is often done so that at the point of crossing the stream of fuel and oxidizer meet. As a result of the streams hitting each other, their faster disintegration into drops occurs, and mixing of fuel components is improved.

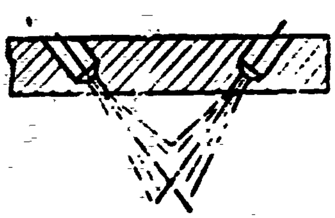


Fig. 5.2. Jet atomizers with crossing axes.

Centrifugal atomizers ensure the best atomization (Fig. 5.3). In these atomizers the fuel component passing through the atomizer channel, obtains rotary motion, which is maintained also outside of

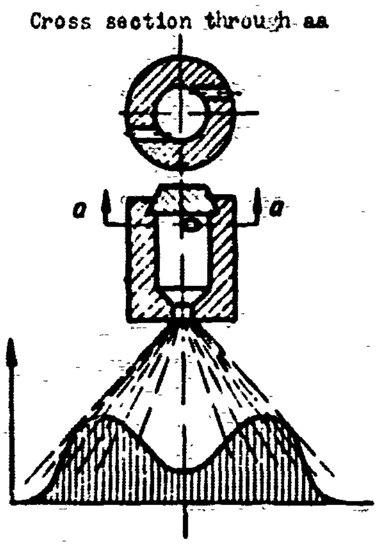


Fig. 5.3. Centrifugal atomizer and the law of flow rate pressure distribution in a section of the jet of the component.

the channel. Under the action of centrifugal forces the outgoing stream is stretched into a film, which rapidly breaks up into drops.

The jet of centrifugal atomizers is wide and short, and drops are distributed in it more evenly, although the greatest flow rate pressure of the component is obtained on a periphery of a certain radius with its center on axis of the atomizer.

Rotation of the liquid in the channel of a centrifugal atomizer results either owing to tangential entrance into the atomizer (Fig. 5.4), or due to motion along a spiral channel, formed by a threaded insert (screw) and the wall of the atomizer.

Sometimes both centrifugal and jet atomizers are used simultaneously in the chamber cap.

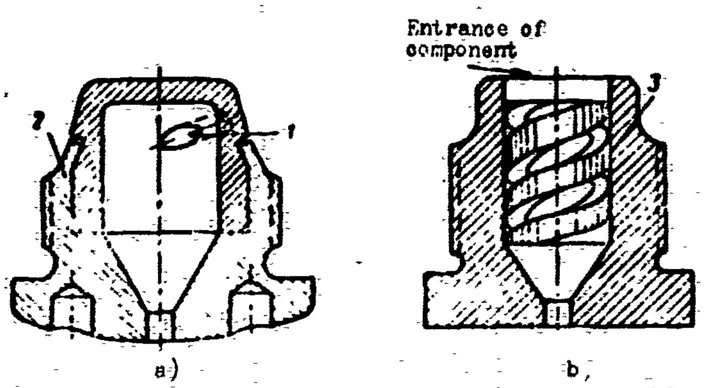


Fig. 5.4. Two types of centrifugal atomizers. a) with tangential entrance; b) screw. 1 — entering tangential hole; 2 — expansion, holding cap of atomizer; 3 — cut insert (auger).

The flow rate of components through the atomizers is determined by their passage section and pressure drop on the atomizers as a result the flow rate is proportional to the quadratic root of the pressure drop. The greater the pressure drop, the higher the quality of atomization of fuel. On the other hand, an increase of drops leads to a growth in the necessary feed pressure that is connected, especially in displacive feeding, with an increase in weight of the engine's feeding system.

As was already mentioned above, obtaining of a uniform mixture requires a great number of atomizers with a low flow rate through every atomizer.

And indeed, completed engines have a large number of small atomizers. Thus, for instance, in the engine of the V-2 rocket, atomization of fuel in a quantity of approximately 3 kg/sec for one prechamber is carried out through 68 atomizers (of which 44 are centrifugal and 24 are jet), so that for one atomizer on the average the flow rate is 42 g/sec.

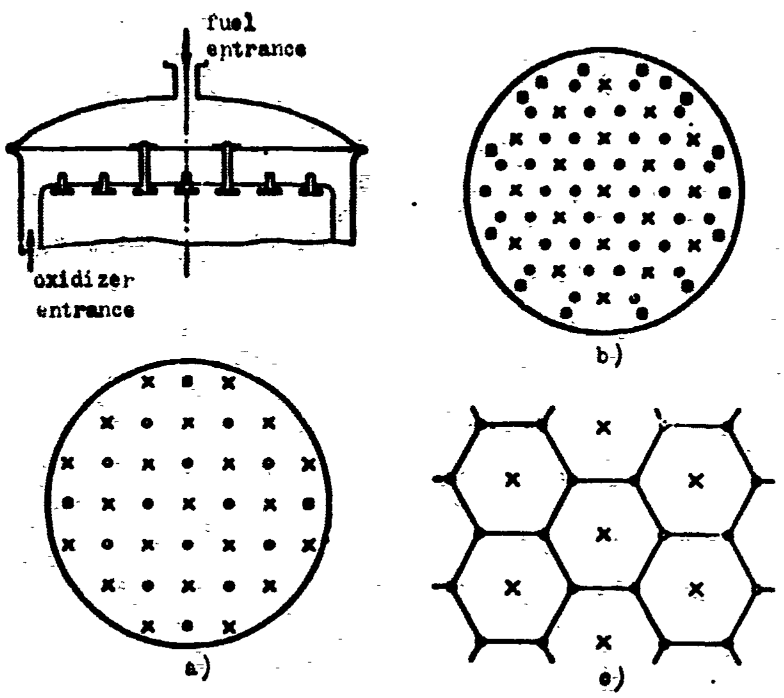


Fig. 5.5. Diagram of flat cap and types of atomizer distribution on it. a) checkered distribution of atomizers; b) honeycomb distribution of atomizers; c) honeycomb distribution scale. × – basic fuel atomizers; • – oxidizer atomizers; ■ – peripheral fuel atomizers for protection of chamber walls.

In order to obtain the most possible equal distribution of components per section of combustion chamber atomizers for fuel and oxidizer are disposed on the cap in a definite order. It is very simple to dispose in the atomizers on the flat cap of the engine in a definite order (Fig. 5.5). On such caps a checkered or honeycomb distribution of atomizers is frequently applied. The latter method of location is convenient because it allows an increase in the number of oxidizer atomizers as compared to the number of fuel atomizers. And although the flow rate of oxidizer is usually 2-4 times greater than the flow rate of fuel, dimensions of oxidizer atomizers and flow rate through them are obtained in this case approximately the same as for fuel atomizers.

The principle of equal distribution of fuel and oxidizer may be disturbed on the periphery of the cap. Here frequently only single fuel atomizers are installed for formation of a boundary gas stream of lowered temperature, protecting the wall from direct influence of the nucleus of flow, in which the temperature is higher. For decrease of unproductive flow rate of fuel through peripheral atomizers, they are made so that the flow rate of the component through each of them as compared to the flow rate of the corresponding component through the basic atomizer is 30-50%.

If the cap is not flat, but of spherical form, the atomizers are arranged in special atomizer chambers (see Fig. 5.10).

For evaporation and subsequent heating of vapors it is necessary to heat the atomized fuel. This heating occurs as a result of vortex motion of gas at the cap, radiation from volumes of gas having high temperature, radiation from hot chamber walls and, finally, after the start of the burning reaction by means of direct transmission of heat liberated during the reaction. The greatest value in feeding of heat to fuel drops in process of evaporation of vortex motion of gas near the cap.

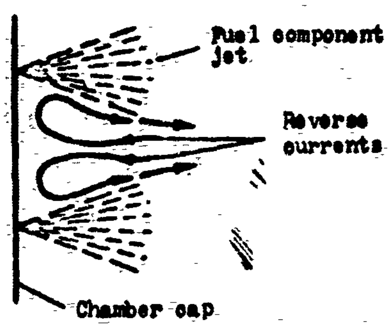


Fig. 5.6. Diagram of the appearance of the vortex motion and reverse currents for the cap of combustion chamber.

Vortex motion of gas near the cap is accompanied by reverse flows in the space between the jets of fuel components ejected by the atomizers (Fig. 5.6). These reverse currents carry heat necessary for evaporation of fuel and promote mixing of fuel and oxidizer.

In the mixture of fuel and oxidizer vapors chemical reactions start due to heating, as a result of which heat is evolved, sufficient for continuation of the reaction in the total volume of the combustion chamber.

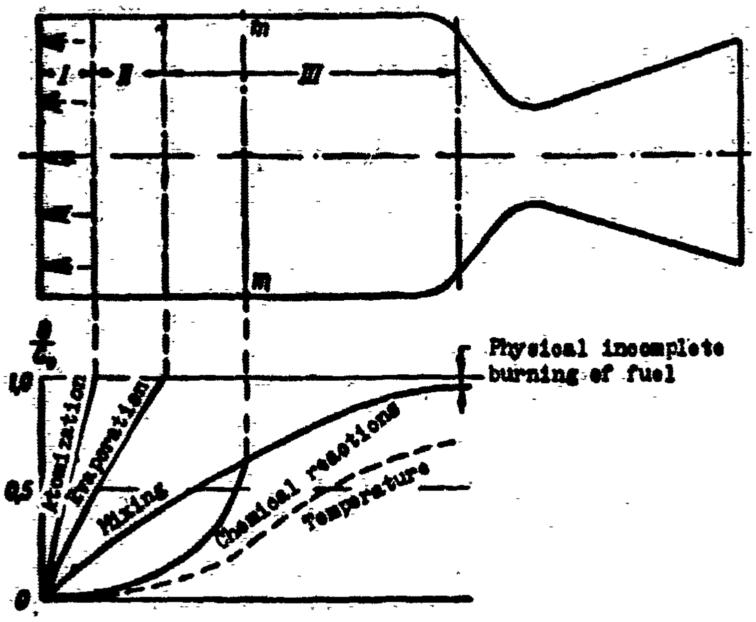


Fig. 5.7. Combustion chamber of liquid-fuel rocket engine and its separation into different zones. I — atomizing zone; II — evaporation zone; III — zone of mixing and chemical reactions; mm — section of transition of burning from the kinetic region into the diffusion region; G/G_0 — relative quantity of atomized or evaporated, or mixed, or reacted fuel.

In accordance with such a picture carburetion processes and burning in a liquid-fuel rocket engine, the combustion chamber may be conditionally divided into several characteristic zones (Fig. 5.7).

In the first zone, directly adjoining to the internal surface of the cap, disintegration of jets of fuel components into drops occurs. Therefore, this zone may be called the <u>atomizing zone</u>. Other processes — evaporation and mixing — occur in it with a relatively small intensity.

As a result of further advance of fuel along the chamber evaporation of components becomes more intense, and their mixing starts. In this zone chemical reactions already also start, but their rate is small due to low temperature. Consequently, the second zone may be conditionally called the <u>zone of evaporation and mixing.</u>

Finally, as a result of accumulation of fuel vapors and increase in their temperature, in the following, third, zone chemical reactions begin to take place intensely. In the first part of this zone the temperature of the gas is still relatively low, and correspondingly the rate of chemical reaction is slow. Therefore, all the evaporated and mixed fuel does not burn here right away, but steadily in conformity with the rate or, so to speak, with the kinetics of the chemical reactions. This region of the combustion chamber is called the region of kinetic burning.

Growth of temperature leads to a very sharp growth of speed of chemical reactions, where starting from a certain temperature all fuel, which turns out to be mixed, practically burns instantaneously. Now the burning rate will be almost wholly dependent on the speed of mixing of the components. Since the speed of mixing is determined by the diffusion rate,* then this region is called the region of diffusion burning. The process of burning in a liquid-fuel rocket engine occurs chiefly in the diffusion region, so that the time necessary for combustion is determined by the speed of mixing. Thus, the third zone of chamber of burning is the zone of mixing and chemical reactions.

Time of Presence of Fuel in Combustion Chamber

The dimensions of the combustion chamber have to be such that mixing and chemical reactions succeed to be completed before entrance into the nozzle of the chamber. This will ensure the fullest transformation of chemical energy into heat and will decrease physical incomplete burning of fuel.

The necessary dimensions of the chamber are determined by the conditional magnitude of time of presence in the chamber τ.

If the flow rate of the fuel, temperature of burning products in the combustion chamber before entrance into the nozzle, and pressure in that same section are respectively equal to G, T_c and p_c, then the general volume of gases passing through the chamber in one second is

$$V_{\text{sec}}=G\frac{RT_{\kappa}}{p_{\kappa}}. \quad [\kappa = c]$$

*So-called turbulent diffusion is implied. In distinction from molecular diffusion, in this case there occurs a random shift not of separate molecules, but of small volumes of gas.

This volume of gas will be in the chamber during period of time

$$\tau=\frac{V_{к}}{V}=\frac{V_{к}p_{к}}{GRT_{к}}, \qquad (5.1)$$

where V_c — volume of combustion chamber.

Magnitude τ is called the time of presence.

Magnitude τ only indirectly reflects the real time of stay of fuel and its burning products in the chamber. The fact is that the volume of fuel as a result of its burning in the chamber increases from an insignificantly small magnitude of liquid fuel volume to a value V, and the time of stay is calculated namely by this greatest volume. The average volume of fuel in the combustion chamber will be less than V. Thus, the real time of stay of fuel in the chamber is more than τ but corresponds to it with definite conformity.

Time of stay τ, necessary for sufficiently complete combustion of fuel, is determined experimentally and through the study of operated engines. In existing engines it is 0.003-0.008 sec. With an increase of pressure the time of stay in the chamber is increased; consequently, a chamber with that flow rate of fuel at a large pressure may be of smaller dimensions.

The time of stay in a chamber with a given cap construction is the basic factor determining physical fullness of fuel combustion, i.e., that share of fuel which succeeds to enter into chemical reaction in the chamber.

Construction of Combustion Chambers and Chamber Caps

As can be seen from expression (5.1), time of stay τ does not depend on the shape of the combustion chamber, so that for a given volume the chamber can have any shape. However, selection of the combustion chamber shape cannot be totally arbitrary.

First of all the form of the combustion chamber should correspond to the form and construction of the cap. Besides it is necessary to try to make the cap and chamber such that carburetion in the combustion chamber is everywhere approximately identical and that there are no zones where fuel is insufficient.

The relationship of length and diameter of the chamber also affects the process of carburetion and burning. In a long chamber with a small cross section at its cap, it is difficult to install the necessary number of burners. In a short chamber the carburetion zone occupies a significant part of the chamber volume, and the length of the mixing and burning zones becomes very small. The usual relation of cross sectional area of the chamber to area of the smallest, so-called critical, section of the nozzle is within the limits 3-10.

However, there also exist chambers with a very small cross section equal to the critical section of the nozzle (a chamber of such form is called a semithermal nozzle). The merit of a semithermal nozzle is the decrease of diameter and weight of the combustion chamber. Attempts of use of chambers of such type in liquid-fuel rocket engines are known.

In the contemporary engines combustion chambers of two geometric forms, cylindrical and spherical (or close to spherical) are applied most frequently.

The advantage of a spherical chamber is that this chamber of identical volume has the least surface area as compared to a chamber of any other form. A small surface area determines the chamber's low weight and the small quantity of heat which is transmitted in the cooling system.

A spherical chamber is also profitable because of its durability indices. At identical durability the wall of a spherical chamber is half as thick as the wall of a cylindrical chamber. Therefore, in that case when the thickness of the chamber walls is determined not by technological or practical considerations, but by durability (as in engines of large dimensions and with great pressure in the chamber p_c), the spherical form should be prefered.

The deficiencies of the spherical chamber are the complexity of its manufacture and difficulties connected with achievement of good joint function of the cap and chamber. In this respect the cylindrical chamber has great advantages.

Really, considering Fig. 5.8, on which a chamber is represented of a liquid-fuel rocket engine, utilized as an accelerator, we see that a flat cap, fixed on this engine, allows even distribution of fuel through the cross section of the chamber, not leaving any zones into which fuel does not enter and where combustion would not occur — so-called "dark" zones.

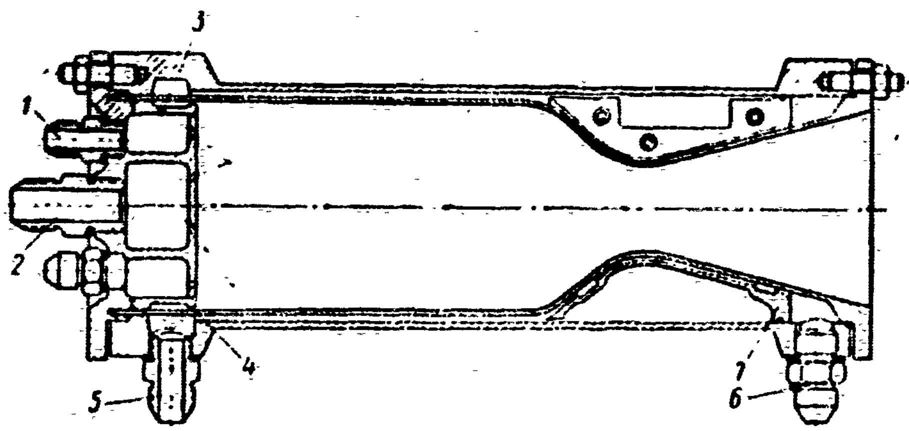


Fig. 5.8. Cylindrical chamber of a liquid-fuel rocket engine — maneuver accelerator. 1 — fuel feed; 2 — oxidizer feed; 3 — passage for supply of fuel for internal cooling; 4 — gap for outlet of fuel for internal cooling of chamber; 5 — input of oxidizer for cooling; 6 — outlet of oxidizer from cooling jacket; 7 — split insert, forming a narrow channel through which oxidizer passes, cooling the nozzle of the chamber.

Such a flat cap, fixed on a combustion chamber of pear-shaped form (Fig. 5.9), no longer allows complete use of the chamber volume for combustion. Part of the chamber, being beyond the borders of the cylindrical volume with a diameter equal to that of the cap, is practically not used for burning of fuel. At the same time, the dimension of the flat cap is obtained such that it is not possible to dispose a large number of centrifugal burners on the cap. For this engine it was necessary to be limited to application in less perfected jet burners.

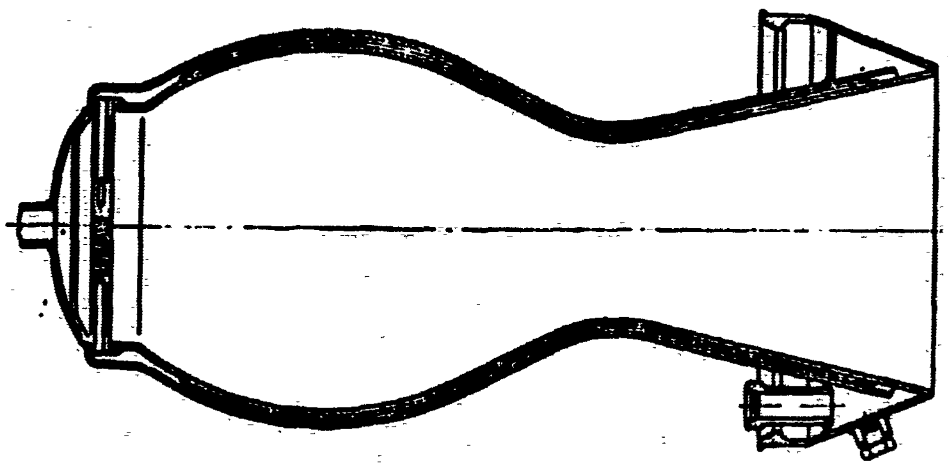


Fig. 5.9. Pear-shaped combustion chamber with flat cap.

In the same way the volume of the combustion chamber in the V-2 rocket engine (Fig. 5.10) is used incompletely, on the spherical cap of which 18 prechambers are distributed. The fuel mixture, prepared in the prechambers, ensues as a powerful jet into the chamber. Although these jets, colliding with each other, are mixed thoroughly, a significant part of the chamber volume between the jets ensuing from the prechambers for is not used the burning process.

The concept on the arrangement of a flat cap with jet burners gives Fig. 5.11.

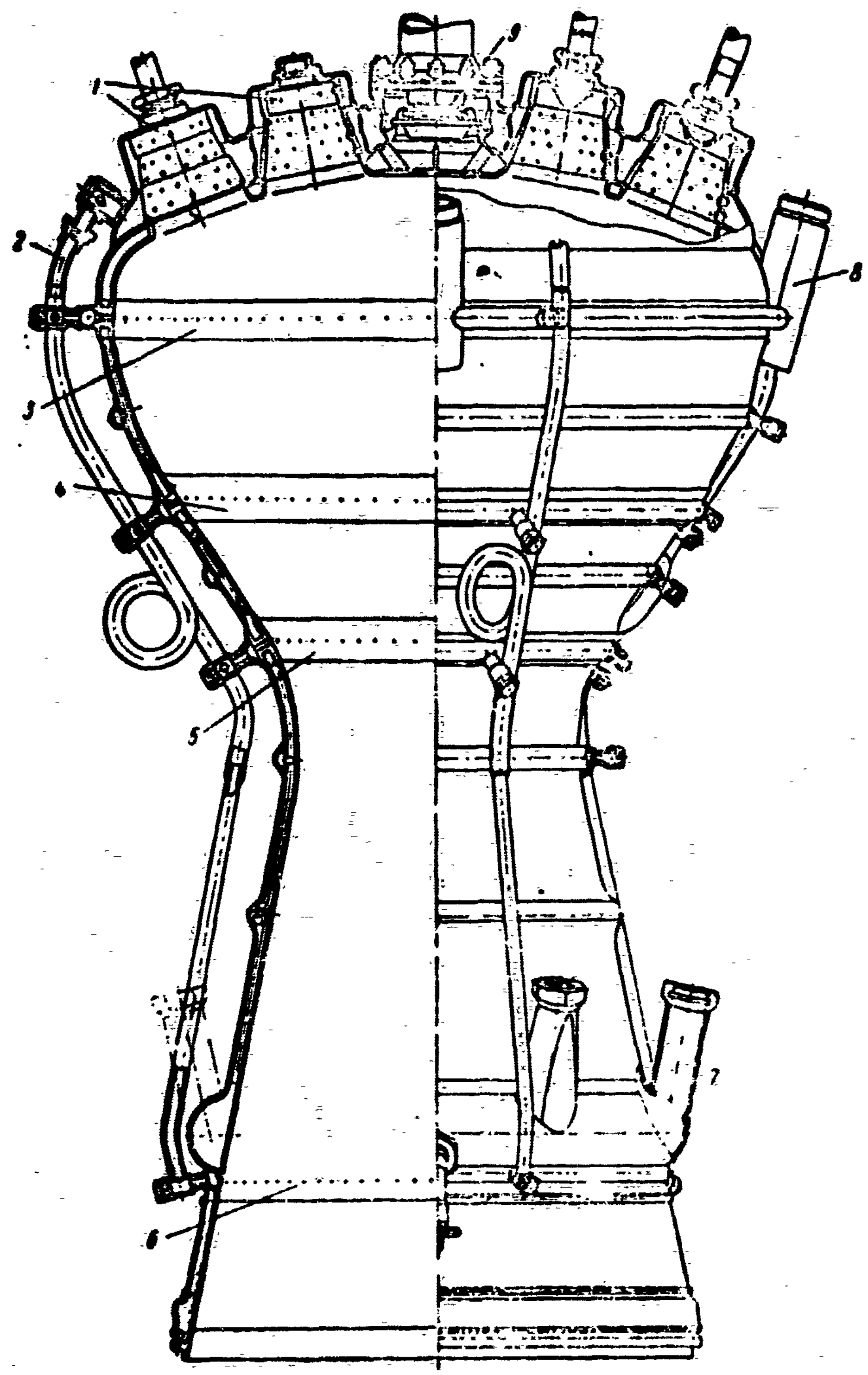


Fig. 5.10. Combustion chamber of long-range rocket V-2 engine. 1 – prechambers; 2 – pipe for inlet of fuel for internal cooling; 3, 4, 5 and 6 – belt of holes for introduction of fuel onto the internal surface of the chamber wall; 7 – pipe for inlet of fuel into the cooling cavity; 8 – brackets for bracing engine to frame; 9 – main alcohol valve.

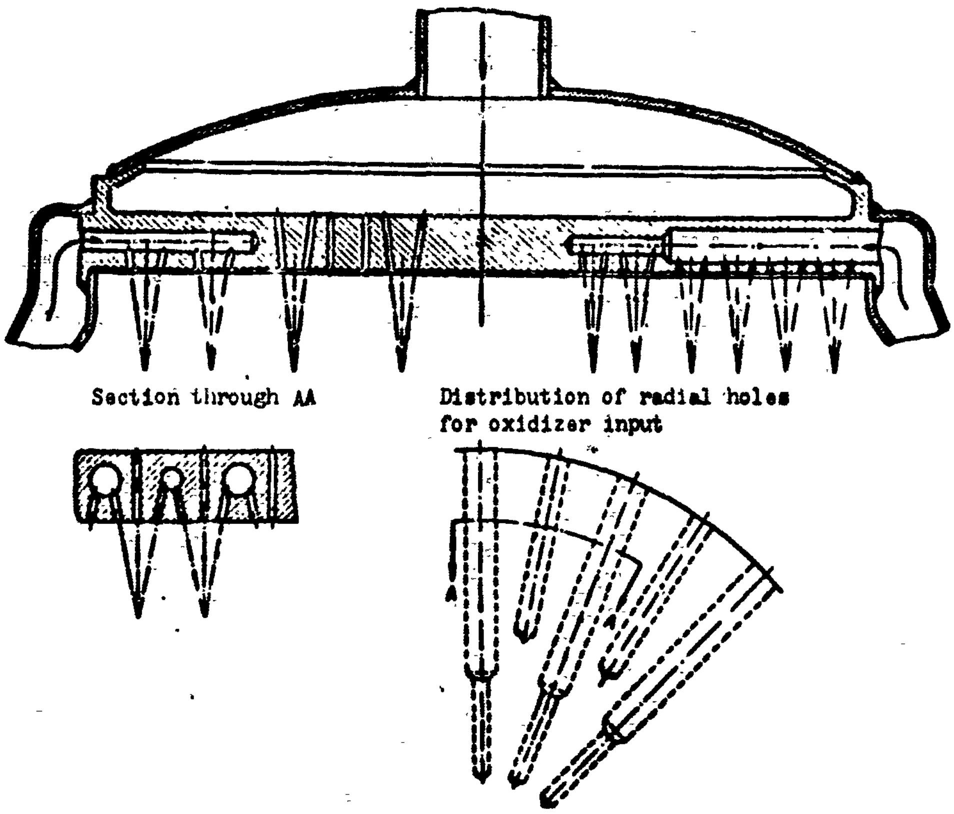


Fig. 5.11. Flat cap of combustion chamber with jet atomizers.

The openings of the fuel and oxidizer atomizers are drilled slanted toward the engine axis and at such an angle to each other to ensure carburetion in the liquid phase. Fuel is brought in from the upper cavity of the cap, and for input of oxidizer it is necessary to drill long radial holes. Manufacture of such a cap turns out to be very complicated.

It is also possible to place centrifugal atomizers on a flat head. It is necessary to consider that they can give more equal distribution of the mixture through the cross section of the chamber. Furthermore, centrifugal atomizers ensure crossing of atomizing fuel and oxidizer jets with a normal location of atomizers, when their axis are parallel to the axis of the chamber; thus, installation of centrifugal atomizers does not require drilling of slanted holes. Centrifugal atomizers however, occupy more space; therefore, the cap should have large dimensions. In the cylindrical chamber this is easily possible to execute without an increase of chamber volume, by increasing its diameter and decreasing length.

In a spherical combustion chamber by conditions of assembly it is very difficult to place burners directly on its bottom. Furthermore, because of small surface area of the bottom, it is difficult to dispose the necessary number of small burners on it. Therefore, prechambers are used on engines with spherical chambers. Arrangement of the prechamber on the engine of the V-2 rocket is shown in Fig. 5.12. In every such prechamber the oxidizer — liquid oxygen — will be atomized by one atomizer cap with a large number of jet atomizers. Fuel — alcohol — is brought in through atomizers located on the lateral conical surface of the prechamber. It is clear that such a method of mixing does not ensure (at least by simple means) equal distribution of components in the cross section of the prechamber.

In order to improve carburetion, in the described prechamber a complicated system of coordinated atomizers is applied. Oxygen is injected through holes placed on concentric circles and inclined toward the axis of the prechamber at different angles, so that the stream of oxidizer could more evenly fill the prechamber volume.

Belts of fuel atomizers are located in correspondence with belts of oxidizer holes. Centrifugal atomizers are in upper belt 5. The small range of the jet of these atomizers allows protection of the prechamber wall from a direct hit on it by oxygen. The two following belts 6 consist of jet atomizers, which owing to great range carry fuel into the center of the prechamber volume. In the lower belts 7 again centrifugal atomizers are placed. All enumerated measures allow improvement of the quality of carburetion, but on the whole it is

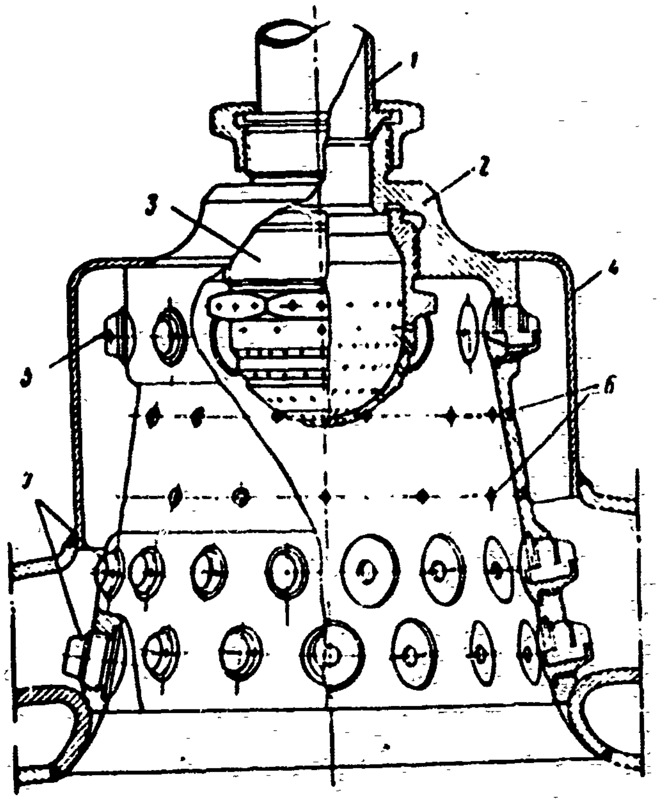


Fig. 5.12. Prechamber of an engine having a spherical cap (long-range rocket V-2). 1 — oxidizer input (liquid oxygen); 2 — body of prechamber; 3 — atomizer cap of oxidizer with jet atomizers; 4 — perchamber housing, forming a cavity for feeding fuel to atomizers; 5 — upper belt of centrifugal atomizers; 6 — belt of jet atomizers; 7 — belt of centrifugal atomizers.

worse than in an engine with a flat cap. As a result of this the time of stay τ, necessary for completion of burning of fuel in the chamber, is increased and, as result there is an increase of the relative volume and weight of the combustion chamber

Ignition of Fuel in a Liquid-Fuel Rocket Engine

The established process of burning in the engine was described above. Initial burning of liquid fuel in the combustion chamber — ignition — in certain cases presents special requirement for the engine.

As we already know, fuels can be spontaneously inflammable and unspontaneously inflammable. Conditions of their ignition in the combustion chamber are different. Anergolic propellants are ignited by means of their injection into an ignition jet which fills the combustion chamber. The ignition jet is created by a special device and becomes sufficiently powerful to ignite the basic components in those quantities, in which they are fed during start-up. Ignition of anergolic propellants does not present special requirements for cap construction.

Spontaneously inflammable fuels start to react and evolve heat during their contact still in liquid form. Therefore, for their reliable ignition it is expedient to ensure good contact of components in the liquid phase.

Spontaneously inflammable fuels have a certain period of delay of self-ignition. At start-up of the engine in this period an accumulation of liquid unburned fuel occurs in the chamber. Afterburning of accumulated fuel leads to a sharp increase ("throwing") in pressure, which may be dangerous for the engine.

In order to decrease accumulation of fuel in the chamber, at the initial stage of start-up the flow rate of fuel should be artifically decreased in accordance with the delay time of self-ignition.

Namely because of this in a system of supply of engines by spontaneously inflammable components or in engines with chemical ignition devices are applied ensuring gradual build-up of fuel feed in the start-up period. These devices are constructed either in the form of throttle flaps, relatively slowly opened during start-up (see description of zenith rocket engine), or in the form of turning valves and multistage valves, gradually allowing access of fuel to the atomizers.

In big liquid-fuel rocket engines with pump feed, smooth build-up of fuel feed is carried out due to inertia of the turbopump unit, which has a given number of turns for a certain interval of time.

Engines used in aircraft have to ensure several startings during one flight of the aircraft. In such engines a very complicated ignition system, connected usually with the supply system, is created.

Starting of engines of separate stages in multistage rockets in flight should be carried out in conditions of low temperatures and high vacuum. Ignition of fuel in such engines is hampered and requires very thorough selection of the source for creation of the ignition jet.

Vibration Burning

In standing starts using liquid-propellant rocket engines it was noticed that so-called vibration burning can appear in the combustion chamber. Periodically, with a frequency up to 200 cps, a change in gas pressure occurs in the chamber (Fig. 5.13). The amplitude of pressure oscillations can attain very great size, which leads to disturbance of the correct functioning of many units not only of the engine, but also of the whole rocket. In certain cases as a result of oscillations even destruction of the engine is possible.

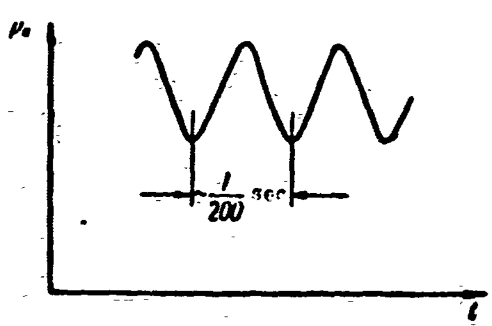


Fig. 5.13. Character of pressure change in a combustion chamber during vibration burning.

Pressure change in the chamber leads to a corresponding change of engine thrust. Periodic changes of thrust harmfully affect durability of rocket junctions and the work of steering equipment. Therefore, appearance of vibration burning in the engine should be eliminated, or in extreme case it is necessary to limit the vibration to a very small amplitude.

Theoretical analysis of conditions of engine function during vibration burning is very difficult to conduct. As is established, appearance of vibration burning is caused by the fact that from the moment of entering of lqiuid fuel into the chamber to formation of gasiform burning products a certain time passes (time of transformation).

Very schematically, the appearance of vibration burning in the chamber of a liquid-fuel rocket engine can be presented by the following form.

We assume that the pressure in the supply system, for instance in the engine reservoirs during displacive supply, remains constant. Let us assume further that by some accidental cause the pressure in the combustion chamber fell as compared to nominal. Then the flow rate of fuel through the atomizers will be increased, since the pressure drop on the atomizers will increase. Increased flow rate of fuel through the atomizers will take place until after the time of order τ in which the first portion of the fuel with increased flow rate will not be turned into gas and will not start to ensue from the engine. From this moment the pressure in the chamber of the liquid-fuel rocket engine increases proportionally to the increased second expenditure of combustion products of fuel and will be greater than the nominal. In connection with this, the pressure drop on the atomizers and the flow rate of fuel through them will decrease. Upon the expiration of the time τ, the flow rate of gasiform products will also decrease, in connection with which the pressure in the chamber will fall, and consequently, the conditions for repetition of the preceding cycle of oscillations will arise.

Change of flow rate, evoked by oscillations of pressure in the chamber, depends on nominal pressure drop on the atomizers. The greater this drop, the less will be the relative change of pressure drop on the atomizers. The less will be the change of flow rate. Thus, an increase in pressure drop on the atomizers counteracts the appearance of pressure oscillation and vibration burning.

In exactly the same way the oscillatory conditions in the chamber depends on the volume of the burning chamber. The greater the chamber volume the greater the part of unnecessary flow rate of fuel transforms into the gas reserve in the chanber. Thereby in a chamber of large dimencion, pressure oscillation is dampened.

Pressure oscillations in the chamber may also cause oscillation of the fuel column in feeder tubes which under the determined relationships of frequencies can lead to an increase in amplitude of pressure during vibration burning. If during vibration burning such strong pressure oscillations occur that feeding of fuel into the chamber in some moment in general will stop, then during subsequent restoration of fuel feed, explosion of the engine is totally possible.

Besides increase of pressure drop on the atomizers and increase of the combustion chamber volume, measures preventing appearance of vibration burning are an increase of the burning rate that leads to decrease of transformation time for fuel into gasiform products, and also selection of such geometric dimensions of the chamber and supply systems, which do not allow development of oscillations in them.

Burning rate depends basically on the properties of the fuel. Therefore, different fuels have different inclination to vibration burning. In selection of fuels for liquid propellant rocket engines, at present much attention is allotted to finding fuels stable with respect to appearance of vibration burning.

Besides vibrations of low frequency (up to 200 cps), oscillations of significantly higher frequencies having smaller amplitude also take place in the engine chamber.

As is proposed, the cause of appearance of high frequency vibrations in many respects is similar to causes of onset of oscillations of low frequency. However, they are connected not with the burning process in the chamber on the whole, but with processes of burning and transformation of liquid fuel into heated gases in separate small volumes of the chamber.

2. COMBUSTION OF SOLID ROCKET FUELS

Rate of Combustion of Solid Fuel

Solid rocket fuel represents a homogeneous mass which is impenetrable to gases and each small volume of which contains the necessary mixture of fuel and oxidizing compnents. The burning of the fuel takes place at the surface, and the flame front penetrates inside the body, or, as it is said, the arc of the body only as the higher layers are burnt up.

It has been established theoretically and experimentally that the combustion of fuel is preceded by thermal decomposition of the material on its solid surface. The intensity of the decomposition reaction is basically determined by the rate at with which heat is applied from the zone where the further combustion of the gaseous products of decomposition formed at the surface of the burning fuel takes place. The heat is supplied to the surface through the heat conductivity and through radiation, since the flow of the gaseous products is always directed away from the burning surface and thus no appreciable convective heating of the grain is possible.

The basic characteristic of the combustion of ~~fuel~~ powder is taken to be the rate of combustion U_p, i.e., the thickness of the layer of fuel burning per unit of time. Usually this quantity is expressed in cm/sec.

Since the fuel forms a homogeneous mass, it is natural to expect that the fuel will burn evenly over the entire burning surface. Experiments in which the combustion of a fuel grain was suddenly stopped has confirmed this assumption.

Fig. 5.14, a shows a view of a fuel grain before combustion. This grain was ignited over the entire surface, and then after a certain interval of time the combustion was extinguished (see Fig. 5.14, b). From the photographs of the grain it is apparent that the burning takes place fairly evenly over the entire surface.

The weight of ~~fuel~~ powder burnt per unit time, and consequently the amount of combustion products formed, is

$$G_n = F_n U_n \gamma_{n'} \qquad (5.2)$$

where F_p is the burning surface of the ~~fuel~~ powder grains;

U_p is the rate of combustion of the ~~fuel~~ powder;

γ_p is the specific gravity of the ~~fuel~~ powder.

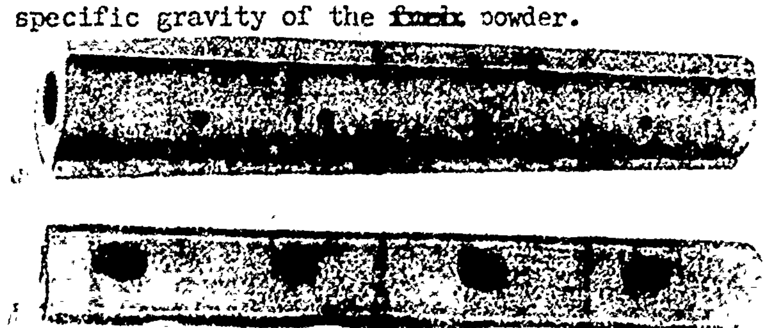


Fig. 5.14 View of a powder grain before combustion (a) and after 70% combustion (b).

Since the specific gravity of the fuel is a constant quantity ($\gamma_f \approx 1.6 \div 1.7$ kg/l) the amount of gases formed depends on the size of the burning surface and on the rate of combustion.

The rate of combustion of powder (or of another solid rocket fuel) is determined primarily by composition. However, the rate of combustion is also affected strongly by the pressure at which the combustion takes place and by the initial temperature of the powder grains. The increase of the pressure facilitates a supply of heat to the powder grain and accelerates the reactions taking place on its surface.

At pressures up to 200 kg/cm^2, which are characteristic of the combustion chambers of powder engines and of powder pressure accumulators, the dependence of the rate of combustion of the powder on the pressure p can be expressed approximately by the empirical formulae

$$U_n = \alpha + \beta p \tag{5.3}$$

or

$$U_n = b p^n, \tag{5.4}$$

where α, β, b and n are constants found experimentally.

For nitroglycerine powders, the value of the exponent n in formula (5.4) varies within the limits of 0.6 - 0.8. The coefficients ' and b do not depend only on the composition of the powder, but also to a considerable extent on the initial temperature of the powder (before combustion). The effect of the initial temperature of the powder on the rate of combustion is natural. The increase of the temperature facilitates the ~~taking place of the~~ decomposition reactions of the powder on the surface of the charge and thus increases the rate of combustion. At the same time, because of the relatively large rate of combustion and the small heat conductivity of the powder, the entire powder grain is not heated through during the process of combustion. Consequently, the temperature of the powder remains almost constant and equal to its initial temperature (only a very thin layer is heated, which is practically already a part of the reaction).

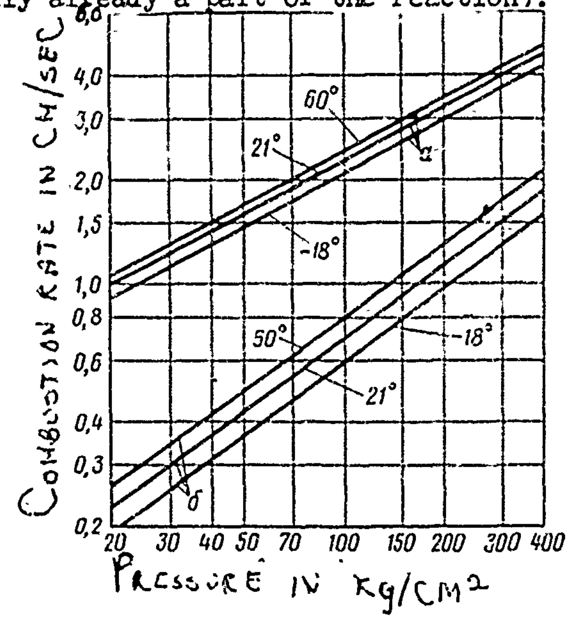


Fig. 5.15. Dependence of the rate of combustion of powder on the pressure and initial temperature of the charge.

a - rapidly burning powder, b - slowly burning powder

The dependence of the rate of combustion of the powder charge on its initial temperature is quite appreciable and causes a considerable variation in the rate of formation of combustion products within the temperature interva from -50 to +50°C, which is characteristic of the different seasons and climatic conditions.

Typical relations of the rate of combustion of powders to the pressure and temperature are shown in logarithmic coordinates in Fig. 5.15.

The upper group of lines belongs to powder having a large rate of combustions (2 cm/sec at a pressure of 80 kg/cm^2 ~~at~~ and an initial temperature of the charge of $t=21°C$). The rate of combustion of this powder depends to a relatively small extent on the pressure ($n=0.52$) and the initial temperature.

The lower group of lines refers to a slowly burning powder (rate of combustion 0.6 cm/sec at the same pressure and temperature as above). However, the dependence of the rate of combustion of this powder on the temperature and especially ~~under~~ on the pressure is stronger ($n=0.71$).

Solid rocket fuels on a perchlorate base, which are now widely used in the rocket technology of foreign armies, follow qualitatively the same combustion laws as ordinary powders. It should be noted that the rate of their combustion depends to a considerably smaller extent on the initial temperature of the charge than in ordinary powders. For certain fuels, the pressure also has a smaller effect on the rate of combustion. Moreover, the new solid fuels are able to burn stabily at relatively low pressures.

These properties of the new solid fuels are their great advantage, since the amounts of product of combustion formed per unit of time in the burning of fuel will depend to a smaller extent on climatic and meteorological conditions.

Consequently the thrust of the engine, as well as the pressure in the combustion chamber of a rocket engine (very important), will fluctuate in a smaller range. The reduced range variation of the pressure makes it possible to decrease the safety factor of the combustion chamber. If it is considered that perchlorate solid fuels can burn at a considerably lower pressure than ordinary nitroglycerine powders, it will now be clear that in this case the combustion chamber of the rocket engine can be made much lighter.

Shape of Rocket Powder Charges

As follows from expression (5.2), the amount of gas formed per unit of time as constant pressure is determined by the size of the burning surface of the charge F_p. During the process of combustion of the powder grain, the quantity F_p generally does not remain constant and can diminish as well as increase. If the surface F_p decreases during the process of combustion, the amount of gas formed per unit time will also decrease. This type of burning of the charge is called degressive. If the burning surfaceincreases with time, the amount of gas formed per unit of time also increases. In this case, the burning of the charges were progressive.

By making powder grains in various shapes, it is possible to regulate within certain limits the formation of the gases in time.

Ordinarily, in the design of powder rockets, an attempt is made to obtain constant thrust of the engine over the trajectory. For this it is obviously necessary to produce a constant amount of gases per unit time, i.e., to have a burning surface of the powder grains of constant size. To satisfy this condition, grains are made in a special shape. An example is provided by the hollow cylindrical grain (called a tubular grain) shown in Fig. 5.16. In this grain, the burning of the outer cylindrical surface results in a decrease of the burning surface, while the burning of the inner surface results in an equivalent increase of the burning surface. If the grain is very long, the effect of the burning of the ends on the common surface will be very small, and the combustion will take place over an almost constant surface (or, more accurately, over a very slowly decreasing surface). The burning of a grain of this type will be slightly degressive.

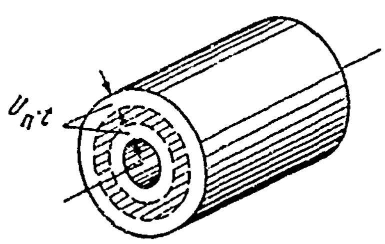


Fig. 5.16. The burning of a tubular grain.
The dotted line shows the burning surface after t/sec. Shading indicate the surface burning from the end.

We note that, in some cases, for instance to obtain a large velocity of the rocket in launching it from short guides, it is necessary to achieve rapid burning of the powder (in 0.1 - 0.3 sec). In such cases, the surface of the grain is increased and their thickness (the thickness of the arc) is decreased. A multi-grained charge is made (Fig. 5.17,a). Other more complicated shapes of grains are used to obtain the required change of the burning surface (see Fig. 5.17, b and c). Especially interesting is a grain ~~here~~ having the shape shown in Fig. 5.17, d. Such a grain burns only over the internal surface of its geometrically shaped channel (star-shaped in cross section). Therefore, the volume of the powder, which as we know has a small heat conductivity, protects the walls of the combustion chamber of the rocket engine from overheating. Grains of this shape can find use in large solid-fuel rocket engines which are designed to operate for a relatively period of time.

In some cases, it is necessary to obtain a solid-fuel charge or powder charge which can burn for an extremely long period of time (several tens of seconds). This is necessary, for instance, in large rocket engines or in powder accumulators of pressure. Such charges can be made out of armored grains. In these grains, a part of the surface of the powder is covered with a plastic form (for instance, acetyl cellulose) which does not burn and which keeps the part of the grain's surface covered by it from catching fire. Fig. 5.18 shows a grain which is armored on all sides, except for one end.

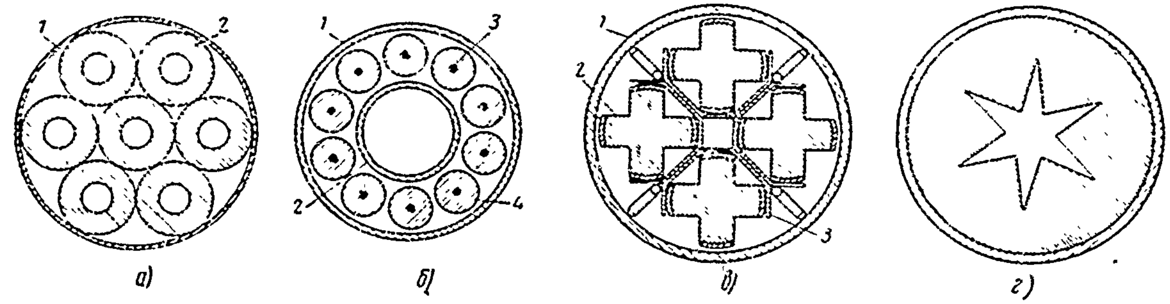


Fig. 5.17 Shapes of powder grains and arrangement of the powder charge

a — seven-grain charge of the thin-arc type.
1— wall of the rocket chamber, 2 — powder grain.
b — charge of tubular grains fixed by means of spindles. 1 — wall of the rocket chamber, 2 — central glass for an explosive charge, 3 — spindle holding the grain, 4 — powder grain.
c — charge of star-shaped grains.
- wall of the rocket chamber, 2 — star-shaped powder grain, 3 — steel partitions.
d— charge which has a channel of geometric design and which burns from inside.

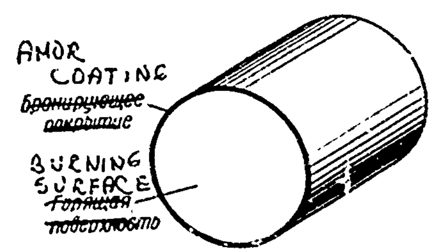


Fig. 5.¹⁸ . Armored powder grain burning from one end.

The burning surface of an armored grain of this type remains strictly constant. This is very convenient in producing a stable regime of combustion and in obtaining a constant rate of formation of combustion products.

The Burning of a Powder Charge in the Combustion Chamber of a Powder Engine.

The relations discussed above are characteristic of the burning of a powder charge when there is no flow of gases along the burning surface of the grain. However, when a grain burns in the combustion chamber of a powder engine, the gases which are formed flow on the surface of the grain as they travel towards the nozzle. Experiments have shown that the rates of combustion of the powder depends on the velocity of the flow of gases flowing on the surface; the greater the velocity of the flow, ~~the~~ the bigger the rate of combustion is. This is explained by the increase of the heat supplied to the powdered from the burning gases.

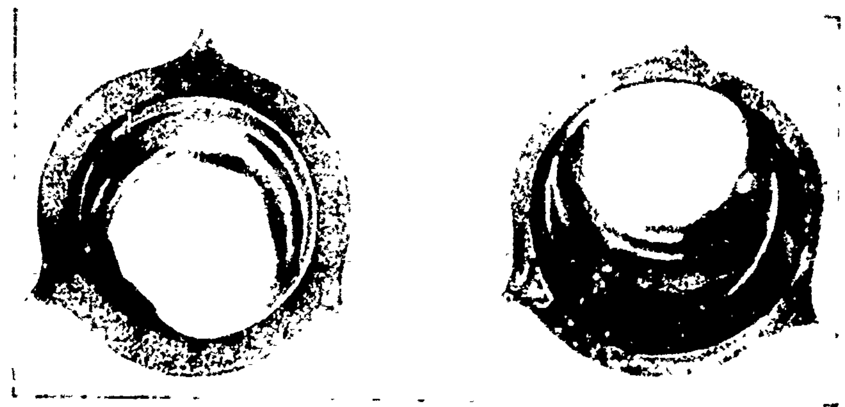


Fig. 5.19. A partially burned powder grain. Lest - view from the ~~side~~ ~~of the~~ bottom of the chamber, right - view from ~~side of~~ nozzle.

In powder engines, the combustion chamber is filled with fuel to the highest degree possible. The space between the burning grains is small, and the velocity of the gases flowing on the surface of the grains is fairly large. The velocity of the gases increases towards the outlet. For this reason, powder grains burn more rapidly on the side of the nozzle. Fig. 5.19 shows a photograph of a grain which was partially burned in an engine. It is clearly seen that the grain burned more on the side of the nozzle.

At small pressures in the combustion chamber, it is possible that interrupted burning, or so-called anomalous burning, will arise. The separate burning consists of the powder charge periodically extinguishing and lighting up again. The pressure at which anomalous burning arises depends on the composition of the powder as well as on the temperature of the charge. A decrease in the initial temperature of the charge favors the arising of anomalous burning.

With nitroglycerine rocket powders, it is possible to lower the pressure at which anomalous burning begins to 20 - 40 kg/cm by adding special substances to the powder. Solid fuels on a perchlorate base have a smaller tendency towards anomalous burning and burn stably at low pressures without any admixtures.

Ignition of a Rocket Powder Charge

The ignition of a rocket powder charge is carried out by means of an igniter. A diagram of the construction of the igniter is given in Fig. 5.20.

To light the igniter, an electric current is fed through the wires, 3, of the firing device (electric fuse), 4, to the filament b. This filament becomes incandescent and easily lights the ignition mixture c. From this ignition mixture, the charge 2, is ignited.

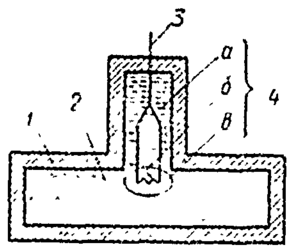


Fig. 5.20. The construction of the igniter.

1- body of igniter; 2- charge of igniter; 3- wires supplying current to electric fuse; 4- electric fuse; a- sealing wax, b- filament, c- easily ignited mixture.

The body of the igniter, 1, can be made of plastic or metal. As the ignition charge burns, the body of the igniter is destroyed, and the hot gases formed in the burning of the igniter blow on the surface of the main powder charge and ignite it. In the process of ignition, first of all the temperature of the surface of the charges is raised to the ignition temperature, and second the pressure in the combustion chamber is raised to a pressure providing for normal burning of the charge.

The time it takes for the powder charge to be ignited must be as short as possible. For this, it is necessary that there be an intensive transfer of heat from the combustion products of the igniter to the powder charge.

Under the conditions of ignition, where the velocity of the flow of the gas in the chamber is small, radiation is important in the transfer of heat. However, the radiative power of gases is smallx. To increase their radiative power, the charge of the igniter is made in such a way that its products of combustion will contain a considerable amount of solid particles, which radiate heat intensively. Therefore, x the charge of the igniter is made either of black powder or from a mixture of magnesium powder (or aluminum powder) and potassium perchlorate ($KClO_4$), in the burning of which a fairly large number of solid particles is formed.

More rapid ignition (5 - 10 msec) is provided by igniters made of $KClO_4$ and Mg (or Al). However, they are more dangerous to handle. Moreover, the metal powder entering into the composition may oxidize if they are kept for a long time, and this will cause failure of the igniter. The time it takes to ingite a rocket charge with black powder is longer (25 - 30 msec), but it is more reliable when stored.

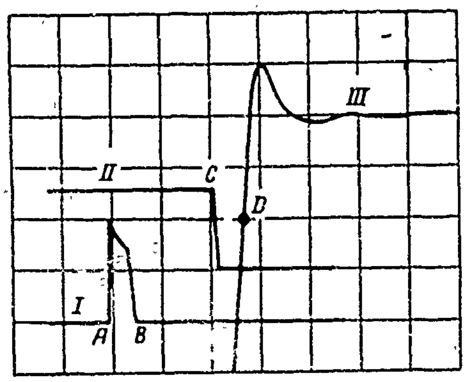
The development of the ignition process in time is shown in Fig. 5.21.

In the combustion chamber of a powder engine the igniter is located at the ends of the charge. More reliable ignition of a powder charge is provided by the igniter located on the bottom of the chamber. In this case, the igniting gases flow about the entire charge and heat it before flowing out of the nozzle.

3. PRODUCTS OF COMBUSTION OF THE FUELS OF ROCKET ENGINES AND THEIR PROPERTIES

Parameters of the State of the Gas Mixture.

The process of combustion which takes place in the combustion chamber of the rocket engine represents a group of complicated chemical reactions and of preliminary processes which are necessary for these chemical reactions.



Each division represents 10 miliseconds

Fig. 5.21. The process of ignition in time.

I- Curve of strength of current in the circuit of the filament.

A- Supply of current to filament, B-Burning of Filament.

II- Curve of the internal stresses in the body of igniter, C- Beginning of destruction of the body.

III- Curve of the increase of pressure in the combustion chamber,

D- The pressure sufficient for the beginning of combustion of charge.

The basic results of the combustion process is the conversion of a liquid or solid fuel into products of combustion heated to a high temperature. Since a fuel always contains several components, the products of combustion represent a mixture of different chemical compounds, most of which are gaseous. The properties of the products of combustion of the fuel have an extremely important influence on the course of the processes of combustion and expansion. As we have seen, in the selection of the fuel it is necessary to take careful consideration of the properties of the products of combustion. Otherwise, it is impossible to evaluate the qualities of the fuel correctly. Still greater attention must be devoted to the thermodynamic properties of the materials which can be used as carriers of energy in the nuclear rocket engines of the future. Therefore, we must familiarize ourselves with the basic thermodynamic properties of gaseous substances and with the quantity defining their state.

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The state of the gas is characterized by the following parameters: the absolute pressure p; the absolute temperature T; the density ρ (or else the specific gravity γ, or the specific volume V); and the gas constant R.

For an ideal gas (or a mixture of ideal gases), the parameters p, ρ, and T are connected by the well known equation of ~~space~~ state (the Clapeyron equation):

$$\frac{p}{\rho} = gRT. \qquad (5.15)$$

The gas density is connected with the specific volume by the equation

$$g\rho = \frac{1}{V}. \tag{5.6}$$

Accordingly the equation of space can be written in the form

$$pV = RT. \tag{5.7}$$

The value of the gas constant for a mixture of gases is determined by the composition of the mixture. To compute R, it is possible to use the equation

$$R = \frac{\overline{R}}{\mu_\Sigma}, \tag{5.8}$$

where $\overline{R}$ is the universal gas constant per one kg-mole of any gas or any gas mixture;

$\overline{R} = 848$ kilogram meters/kg-mole deg, or in thermal units Kcal/kg-mole deg;
$A\overline{R} = 1{,}986$ KILOGRAM METERS /Kg -mole deg.

μ_Σ is the apparent molecular weight of the mixture.

The apparent molecular weight of the mixture is

$$\mu_\Sigma = \sum \mu_i r_i, \tag{5.9}$$

where μ_i is the molecular weight of the i-th gas which forms a part of the mixture;

r_i is the volumetric proportion of the gas with molecular weight μ_i.

It is simplest to express the volumetric proportions of the gases in a mixture in terms of partial pressures p_i.

As known, the partial pressure is the term for the pressure which the gas would have if it occupied the entire volume in which the gas mixture is located.

The total pressure of the gas mixture p_Σ is equal to the sum of the partial pressures:

$$p_\Sigma = \sum p_i.$$

The volumetric proportion with which we are concerned is

$$r_i = \frac{p_i}{p_\Sigma}. \tag{5.10}$$

From this equation, we obtain in the place of (5.9)

$$\mu_\Sigma = \frac{1}{p_\Sigma} \sum \mu_i p_i. \tag{5.11}$$

For a gas mixture of constant composition, the value of μ_Σ and R is constant. If the composition of the mixture changes, the apparent molecular weight and the gas constant of the mixture will also change.

Internal Energy and the Specific Heat of the Gas

The thermodynamic processes in rocket engines are accompanied by a change in the energy state of the gas mixtures and by the conversion of energy from one form to another. Therefore, it is necessary that we acquaint ourselves with the energy characteristics of gases.

One of the energy characteristics of the gas is its <u>internal energy</u>. The internal energy of the gas means the energy of the thermal motion of the particles (molecules and atoms) of which the gas is formed. This concept corresponds to the understanding of heat as a form of motion of matter.

The energy of the thermal motion is determined by the temperature of the gas and by the structure of the actual molecule.

As known from physics, the value of the internal energy depends on the number of degrees of freedom of the molecule and represents the sum of the energies of the motion of molecules in each of these directions.

It has been established that the internal energy of a given gas is determined exclusively by temperature. Consequently, the internal energy U can be represented in the form of some function

$$U=\varphi(T). \tag{5.12}$$

Thus, the internal energy is a function of the state of the gas.

The derivative

$$\frac{\partial U}{\partial T}=c_V \tag{5.13}$$

represents the rate of increase of the internal energy as the temperature increases. The quantity c_V is for the <u>constant</u> <u>volume</u> <u>specific</u> <u>heat</u>. Expressed in terms of [illegible] for a unit mass (for instance a mole), the dimensions of this quantity are Kcal/mole deg.

By using the definition of c_V and equation (5.13), we can write the internal energy in the form

$$U=\int_0^T c_V\,dT. \tag{5.14}$$

At first sight it may seem that the internal energy of the gas depends on the conditions (such as constant volume or constant pressure) under which heat is supplied to the gas. Indeed, the term constant volume specific heat means that, when the gas is heated under the condition V = const , the heat is used only in increasing the internal energy of the gas, and not for any other purpose.

Heat Content of a Gas. Adiabatic Exponent.

The second function which characterizes the energy state of a gas is the heat content, or enthalpy H.

The heat content differs from the internal energy inasmuch as it represents the internal energy added to the product pV or, in thermal units, ApV. This product is a measure of the potential energy possessed by one kilogram of gas occupying a volume V at a pressure p. Therefore, the heat content is a measure of the sum of the internal energy and the potential energy of the pressure of the gas. Thus, for instance, the total energy of a compressed spring is equal to the internal energy of the material of the spring, which is heated to a given temperature, and of the work expended in compressing it. This total energy of a compressed spring is analgous to the heat content of a compressed gas.

The heat content is the most important energy characteristic of the gas, since in the vast majority of cases both the internal energy and the potential energy change when the state of the gas changes in different technical processes. Thus, the total change in the energy of the gas which takes place in different gas processes is always determined by the magnitude of the change of the heat content

From the definition of the heat content,

$$H=U+ApV, \tag{5.15}$$

or according to (5.7) and (5.14)

$$H=\int_0^T c_V\, dT+ART. \tag{5.16}$$

It is easily seen that a change in the heat content of the gas ΔH corresponds to an expenditure of heat in heating a gas at constant pressure. If the temperature of the gas is increased by ΔT and its specific volume by ΔV at a constant pressure p, the expenditure ΔQ will be

$$\Delta Q=\Delta U+p\Delta VA=c_p\Delta T,$$

where $p\Delta V$ is the work exerted in the expansion of the gas at a constant pressure p;

c_p is the constant pressure specific heat of the gas.

From the equation of state $pV=RT$, at a constant pressure p

$$\Delta V=\frac{R}{p}\Delta T;$$

Then

$$\Delta Q=c_p\Delta T=\Delta U+AR\Delta T$$

and

$$c_p=\frac{\Delta Q}{\Delta T}=c_V+AR,$$

On the other hand, on the basis of (5.16)

$$\left(\frac{\partial H}{\partial T}\right)_{p=\mathrm{const}}=c_V+AR=c_p. \tag{5.17}$$

Thus, the heat content can be represented in the form

$$H=\int_0^T c_p\, dT.$$

The heat content, like the internal energy, is a function of the parameters of the state of the gas. The change of the heat content in the course of some process does not depend on the type of process and is determined only by the initia and final state of the gas.

The heat content of liquid and solid substances (for instance, fuel components) is almost exactly equal to their internal energy, since their potential energy of compression is negligibly small as a consequence of the smallness of the specific volumes.

We note that for all gases and at all temperatures, the specific heat c_p is greater than the specific heat c_V by the amount AR.

In thermodynamics, as in gas dynamics, the ratio of the constant pressure specific heat to the constant volume specific heat is very important. This quantity is called the adiabatic exponent and is denoted by k:

$$k = \frac{c_p}{c_V} = 1 + \frac{AR}{c_V}. \qquad (5.18)$$

From equation (5.18), it follows that quantity k, depends on the constant volume specific heat of the gas, i.e., on the molecular structure and temperature of the gas. For the technical gases which appear in the products of combustion, the quantity k varies within broad limits depending on the temperature. For a gas of constant chemical composition, the quantity k decreases as the temperature increases. For diatomic gases, for instance, the value of k changes from 1.4 at low temperatures to 1.28 at very high temperatures.

The maximum value $k = 1.67$ is attained by monatomic gases at low temperatures, while the minimum values $k = 1.15$ is attained by triatomic gases at very high temperatures.

The constant pressure specific heat can be expressed in terms of the adiabatic exponent k as follows:

$$c_p = \frac{k}{k-1} AR. \tag{5.19}$$

The internal energy and heat content of the gas, or correspondingly the specific heats c_p and c_v, are computed for the products of combustion of rocket fuels at high temperatures on the basis of experimental data by the method of statistical thermodynamics. Within narrow intervals of temperature, the change of the specific heat with temperature can be represented by functions which are linear or which vary according to some power of the temperature. However, such functions can not be used over the entire range of variation of the temperatures of the products of combustion of liquid-fuel rockets engines.

Chemical Energy and Total Heat Content. Basic Equation of Combustion.

In rocket engines, chemical energy is converted to other forms of energy - internal energy of thermal motion, potential energy of pressure, and finally kinetic energy of the gas jet.

For any substance (including a gas) two energy characteristics which include the supply of chemical energy are distinguished: the total internal energy U_t, and the total heat content I.

The total internal energy is called the sum of the internal energy of thermal motion and the chemical energy $(U_p = U + \Delta H_0^0)$. The total heat content is equal to the sum of the heat content (sometimes called the physical heat content) and the chemical energy, i.e., $I = H + \Delta H_0^0$.

The difference between the total heat content of the fuel and the total heat content of the products of combustion measured at the same temperature T_0, which is equal to the temperature of the fuel before combustion, determines the amount of heat which is given off in combustion of the fuel. This amount of heat is obviously equal to the calorific value of the fuel.

$$K_O = (I_{\text{fuel}} - I_{\text{c.p.}})_{T_0}.$$

Since the calorific value of a fuel is defined at a normal temperature, it is not necessary to consider the difference between the physical heat content of the fuel and the products of combustion because of their small changes at low temperatures. Thus, the calorific value of a fuel is determined basically by the difference between the chemical energies of the fuel and the products of combustion, or, as we saw in Chapter IV, by the difference between their standard heats of formation [see formula (4.3)]

Thus

$$H_{\text{c.p.}} - H_{\text{fuel}} = \Delta H^0_{0\,\text{fuel}} - \Delta H^0_{0\,\text{c.p.}} = K_O.$$

The total heat content is used extensively in computing the combustion and expansion of fuel in rocket engines.

If it is assumed, as is usually done, that in the standard states molecular gases and carbon in the form of β-graphite have a chemical energy equal to zero, then, for example, a chemical energy equal to -94.05 Kcal/mole, or -2140 Kcal/kg is obtained for carbon dioxide. Water vapor also has a negative chemical energy, equal to -57.79 Kcal/mole or -3210 Kcal/kg. The formation of these gases results in a conversion of chemical energy into heat, which may be expended in increasing the physical heat content of the gases.

Some gases which appear in the products of combustion, for instance atomic gases, have a positive chemical energy. This means that, when they are formed (from standard elements), chemical energy is absorbed rather than given off.

By using the concept of total heat content, we can write very easily the basic equation of combustion. For this, it is necessary to apply the law of conservation of energy to the process of combustion.. Let us assume first of all that the combustion is not accompanied with any losses of energy. Then the total heat content I_{cp} of the combustion products at the temperature T which they will have as a result of the process of combustion must be equal to the total heat content I_{fuel} of the fuel entering the combustion chamber:

$$I_{cp} = I_{fuel} \qquad (5.20)$$

In the process of combustion, different energy losses can occur, as through the drawing off of heat by the walls of the chamber, or as a result of incomplete combustion in poor atomization of the fuel. These losses can be taken into account by introducing the coefficient of the completeness of the combustion η . In this case the equation of combustion can be written in the form

$$I_{c.p} = \eta_{cr} I_{fuel}. \tag{5.21}$$

To determine the temperature of combustion from equation (5.20) or (5.21), it is necessary to know the composition of the products of combustion, since both the chemical energy and the heat content depend on the composition of the gas mixture (inasmuch as the specific heat of different gases are different).

In the combustion chambers of liquid-fuel rocket engines, the processes of dissociation have an important influence on the composition of the products of combustion and on the completeness of the transformation of chemical energy.

4. THERMAL DISSOCIATION AND COMPOSITION OF THE PRODUCTS OF COMBUSTION.

Thermal Dissociation and Constants of Equilibrium.

In describing the processes taking place in the combustion chambers of rocket engines, (especially liquid-fuel engines), we devoted special attention to the problems of obtaining complete combustion of the fuel and, consequently, complete conversion of its chemical energy into thermal energy. However, the degree of conversion of chemical energy into thermal energy is determined not only by the construction of the chamber and the heat, but also by the characteristics of physical and chemical properties at high temperatures.

As we already mentioned, combustion processes taking place at high temperatures are characterized by an extremely intensive thermal dissociation, that is, the processes of formation of chemical compounds in combustion under these conditions is partly accompanied by their decomposition.

When reactions take place in a reverse direction, there is also a reverse transformation of energy. As a consequence of dissociation, there is a loss of physical heat content and the extent of use of the chemical energy is diminished.

For instance, at a high temperature, the reaction of oxidization of carbon monoxide is necessarily accompanied by the reverse reaction of decomposition of the carbon dioxide:

$$CO + \frac{1}{2} O_2 \rightleftarrows CO_2.$$

From the kinetic point of view, the occurrence of the reverse reaction is explained by the fact that in a gas mixture those collisions - collisions of CO_2 molecules with one another, or with CO and O_2 molecules. If the force of collision is great enough, it will be accompanied by the decomposition of the CO_2 molecules into its components. Here the source of energy necessary to split the CO_2 molecule is the energy of thermal motion.

As the combustion reaction takes place, the number of initial molecules, i.e., CO and O_2 , gradually decreases, and consequently the speed of this reaction also decreases. On the contrary, as the content of the products of the combustion reaction (in our case CO_2) increases, the rate of the dissociation reaction will increase, since the number of collisions in which CO_2 molecules participate increases. As a result, there is a time at which the rate of the reactions of combustion and dissociation become equal to one another - the state of chemical equilibrium begins.

In this state, the average chemical composition of the gas does not change.

The quantitative relation which is established between the dissociated gas and the undissociated gas in the conditions of chemical equilibrium is defined by the so-called equilibrium constant.

For each chemical reaction, the equilibrium constant establishes a relation between the concentrations or partial pressures of the individual components of the gas mixture, which is in chemical equilibrium.

In computations for liquid-fuel rocket engines, equilibrium constants K_p are used which are expressed in terms of partial pressures. Each partial pressure appears in the expression for the constants with the same power as that with which it effects the rate of the reaction*. For instance, the equilibrium constant of a reaction

$$CO_2 \rightleftarrows CO + \frac{1}{2} O_2$$

is expressed in the following way:

$$K_p = \frac{p_{CO} p_{O_2}^{\frac{1}{2}}}{p_{CO_2}}, \qquad 5.22)$$

where p_{CO}, p_{O_2}, and p_{CO_2} are the partial pressures of the gases forming the given mixture.

From this expression to the equilibrium constant, it is seen that the

* As known from the law of mass action for chemical reactions, this power is given by the corresponding coefficients of the reaction.

stronger the dissociation, the larger the quality K_p will be (since the pressures p_{CO} and p_{O_2} are larger, and the pressure p_{CO_2} is smaller).

From the equilibrium constants, it is possible to find the composition of the gas mixture and to determine the change of its composition under varying external conditions.

The value of the equilibrium constants K_p for a given reaction depends solely on the temperature. This relationship is extremely complicated, and for the reactions between the products of combustion of rocket engines it is impossible to give an analytic expression connecting the equilibrium constant to the temperature.

At the present time, the equilibrium constants are computed by the methods of statistival thermodynamics. For this, it is necessary to know the molecular constants, and most important, the chemical energies of the corresponding substances. Tables of values of the equilibrium constants compiled over the required range of temperature are used extensively in computations forprocesses in rocket engines.

Effect of Temperature and Pressure on the Composition Of Products of Combustion.

The values of the equilibrium constants of dissociation reactions increase strongly with increasing temperature, and in the products of combustion the content of dissociation products increases correspondingly. This is also clear from the

k

the kinetic point of view. If the temperature T of the gas mixture is increased, the number of molecules having a large store of energy will increase, and this will result in an increase in the rate of the dissociation reaction of the products of combustion and in the destruction of the equilibrium established at the previous temperature. At the new higher temperature of the gas, an equilibrium state will again be established. This state will be characterized by equal rates of the initial reaction (combustion) and of the reverse reaction (dissociation), but with a higher content of dissociation products in the gas mixture. Thus, the temperature of the gas mixture influences the composition of this mixture. Indeed, it influences the composition in a way such that, as the temperature of the mixture increases, there is a higher content of gas whose formation requires expenditure of heat.

The equilibrium constant for ideal gases does not depend on the pressure, but this does not mean that the composition of the gas mixture always remains constant when the pressure changes.

Many dissociation ractions are accompanied by a change in the volume of the gas mixture. Thus, for instance, in the dissociation reaction of carbon dioxide, the number of moles is increased by 1/2 mole for each mole of completely dissociated carbon dioxide. In the constant-pressure case, this leads to an increase of the volume of the gas mixture.

For dissociation reactions in which the number of moles changes, the composition of the gas mixture will depend on its pressure. At the same time, the equilibrium constant does not change, while the partial pressures of the gas components are redistributed. If the number of moles increases in the dissociation reaction, the increase of pressure will suppress the dissociation reaction and increase the products of total combustion in the gas mixture. In other words, an increase of pressure reduces the degree of dissociation of gases if it is accompanied by an increase in the number of moles. For dissociation reactions in which the number of moles does not change, the composition of the products does not depend on the pressure. Since the number of moles increases in most dissociation reactions of the combustion products of rocket fuels, an increase of the pressure in the combustion chamber reduces somewhat (not very greatly) the degree of dissociation.

Composition and Temperature of Products of Combustion in Rocket Engines. Completeness of Liberation of Chemical Energy.

Ordinary rocket-engine fuels contain four elements: carbon; hydrogen; oxygen; and nitrogen.

If combustion were not accompanied by dissociation, the products of combustion would consist of carbon dioxide CO_2, water vapor H_2O, and molecular nitrogen N_2*. However, at temperatures which are still fairly low, - about 2800° abs. - a considerable dissociation of carbondioxide in water vapor is observed (Fig. 5.22). As a result, carbon monoxide CO, the hydroxyl group OH, and molecular oxygen O_2 and hydrogen H_2 are formed. At a still higher temperature, the content of nitric oxide NO, as well as the atomic gases hydrogen, H, oxygen O and Nitrogen N, becomes significant in the products of combustion.

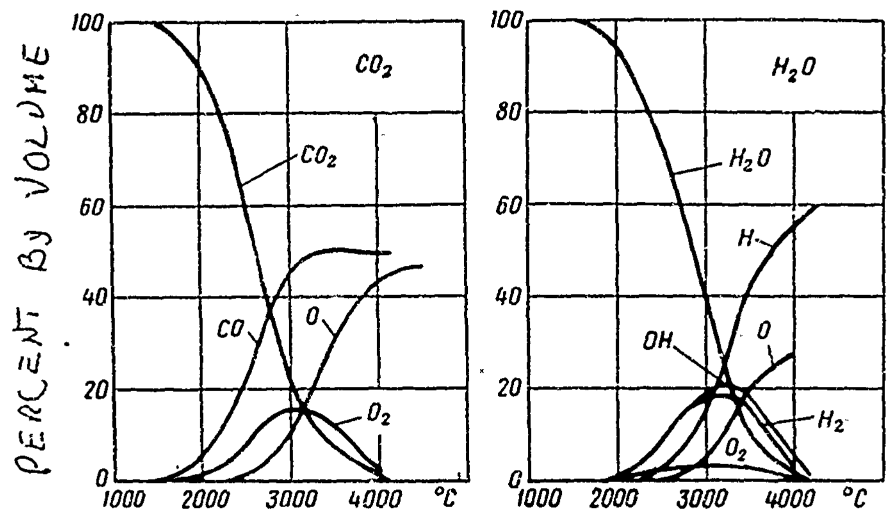

Fig. 5.22. The dissociation of vapor H_2O and carbon dioxide CO_2 depending on the temperature.

The composition of the products of combustion in the chamber is determined from the equilibrium constants in the corresponding dissociation reactions and the pressure in the chamber p_c. In addition, the gas composition depends on the relative content of the different elements in the fuel.

Since the composition of the products of combustion depends on the

* For a stoiciometric relation of the components.

temperature, the solution of equations (5.20) or (5.20) requires a fairly large amount of computations. Usually, it is necessary to assume the value of the temperature, then find the composition of the product of combustion (i.e., the partial pressures of the gases) from the equilibrium constant and the composition of the fuel, and finally verify the basic equation of combustion

$$I_{c.p} = \eta_{cr} I_{fuel}$$

The true temperature and the corresponding comppsition of the products of combustion is thus determined by the method of substitution.

The results of the computations shows that, because of the sharp increase of the degree of dissociation of the products of combustion at temperatures above 3000° abs, an increase of the supply of chemical energy in the fuel does not lead to a proportional increase in the temperature of the chamber. Thus, as an example we can compare the combustion temperatures of two fuels: kerosene + nitric acid, and kerosene + oxygen. The calorific value of the second fuel is 2200 Kcal/kg, which is approximately 50% greater than the calorific value of the first fuel (1460 Kcal/kg). However, the combustion temperature of the fuel kerosene + oxygen (3600° abs) is only 20% higher than the combustion temperature of kerosene with nitric acid (2980° abs). This is a direct consequence of the intensive dissociation of the products of combustion and of the decrease of the extent of liberation of chemical energy.

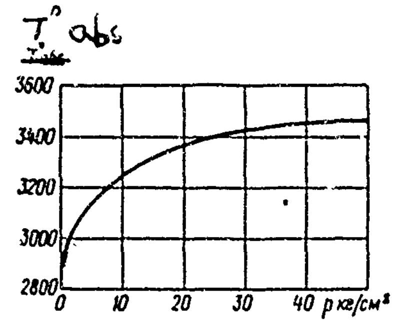

Fig. 5.23. The relation of the temperatures of products of combustion of the fuel oxygen + kerosene to the pressure at $\alpha = 0.7$

Now let us consider the relation of the combustion temperature to the pressure.

This relation is well illustrated by the graph in Fig. 5.23, which was drawn for the products of combustion of the fuel kerosene + liquid oxygen for $\alpha = 0.7$ The dissociation of the products of the complete combustion of ~~its~~ this fuel - these products forming a mixture of carbondioxide CO_2 and water vapor H_2O - takes place with an increase in the number of moles $(CO_2 \rightarrow CO + \frac{1}{2}O_2;\ H_2O \rightarrow H_2 + \frac{1}{2}O_2)$. Consequently, an increase in the pressure must reduce the dissociation of the product of combustion, increase the amount of chemical energy converted to physical heat content of the product of combustion, and finally lead to an increase in the combustion temperature.

As can be seen from the graph (see Fig. 5.23), at first, at small absolute values, an increase in the pressure leads to a rapid increase of the temperature. Then, as the pressure increases further, the temperature increases ever more slowly. This takes place because the raising of the temperature leads in turn to an increase of the dissociation of the products of combustion.

Such a relation of the combustion temperature to the pressure is characteristic of all rocket-engine fuels. Consequently, there is an effort to increase the pressure in the chamber for the purpose of improving the use of the chemical energy of the fuel and thus increasing the specific thrust. 23.

The temperature and composition of the products of combustion is effected by the ratio of the components of the fuel, as well as by the pressure.

If dissociation is ignored, the maximum combustion temperature and the maximum amount of heat liberated in combustion would be obtained for a theoretical ratio γ_0 of the fuel and the oxidizer, i.e., for $\alpha = 1$. However, the phenomenon of dissociation reduces the combustion temperature and the different stabilities of the combustion products with respect to dissociation results in the fact that the maximum temperature and the maximum liberation of heat correspond to values of for presently existing liquid propellent. Thus, they correspond to propellants with a defective oxidizer and an excess of fuel.

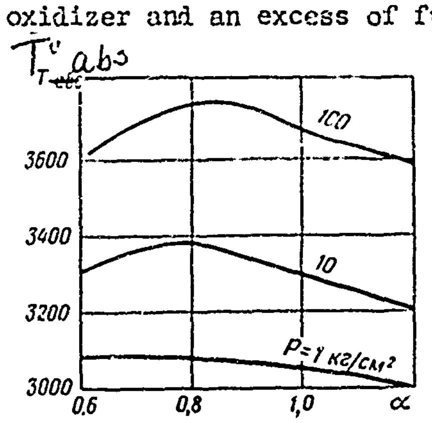

Fig. 5.24. The relation of the combustion temperature of the fuel oxygen + kerosene to the coefficient of the excess of oxidizer α and the pressure p.

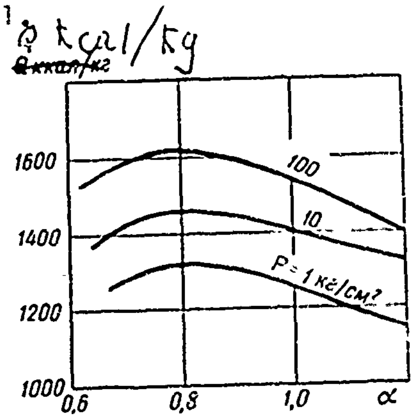

Fig. 5.25. Relation of the heat liberated in the combustion of the fuel oxygen + kerosene to the coefficient of the excess of oxidizer α and to the pressure p.

Fig. 5.24 and 5.25 show the relations of the combustion temperature and the amount of liberated heat Q to the coefficient α of the excess of oxidizer and to the pressure for the case of the fuel oxygen + kerosene. The shift of the maxima of the temperature and of the liberated heat to the side of $\alpha < 1$ is explained by the increase in the relative content of carbon monoxide, which is stable with respect to dissociation, when there is a shortage of oxidizer in the products of combustion. As shown by the curves in Fig. 5.25 the loss of heat through dissociation in the combustion chamber is fairly large: at a pressure of 10 kg/cm^2, it amounts to more than 30% of the calorific value of the propellant which is equal to 2200 Kcal/kg.

We have examined the effect of dissociation for the case of a fuel having a large calorific value. Since the intensity of dissociation diminishes as the temperature decreases, it follows that in fuels with a low calorific value (oxygen + alcohol or nitric acid + kerosene) the influence of dissociation is reduced, although it does remain fairly important. For instance, the loss of heat in the combustion chamber through dissociation amounts to 18 - 25% of the calorific value for the fuel oxygen + alcohol, and to 12 - 18% of the calorific value for nitric acid fuels.

We note that the carrying out of the computations which make it possible to evaluate the quality of the combustion of fuel in rocket engines is possible only by solving a fairly cumbersome system of equations by the method of successive approximation. Consequently, it is necessary to perform many computations of this type to find the best conditions for the burning of a given fuel (α and p_{ch}). At the present time, electronic computers are used for these computations.

It is also necessary to perform computations of the combustion before the subsequent analysis of the exhaust of the products of combustion through the nozzle. As we shall see below, the equations for the exhaust laws contain quantities determined by the composition of the products of combustion; the gas constant R, and the adiabatic exponent k.

As shown by formulae (5.8) and (5.9) the gas constant R depends on the composition of the gas and is determined by the combustion temperature and the pressure at which the combustion takes place.

The adiabatic exponent k changes as a result of the joint influence of the temperature and the composition of the combustion products on the specific heat.

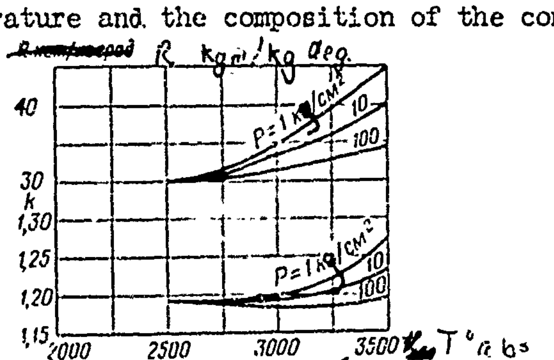

Fig. 5.26. The relation of the gas constant of the products of combustion R and the adiabatic exponent k to the temperature T and the pressure p.

The effect of the temperature and the pressure of the products of combustion on their gas constant R and their adiabatic exponent k is illustrated by the graph in Fig. 5.26. As this graph shows, the gas constant of the products of combustion increases with increasing temperature. This is a consequente of the reduction of the content of monotomic gases in the products of combustion, which is in turn a consequence of dissociation. For the same reason, the adiabatic exponent of the products of combustion increases with increasing temperature and decreasing pressure.

Special Thermodynamic Properties of Gas Mixtures at High Temperatures.

Any thermodynamic process which takes place in a gas mixture heated to a high temperature is accompanied by the reactions of dissociation or recombination, which lead to a change in the chemical composition of the gas mixture. Thermodynamic processes of this type are frequently encountered in practice. In particular, they occur in combustion chambers and especially in the nozzles of rocket engines in rapid flight in the atmosphere, in air flows about streamlined rudders, and so on.

A change in the chemical composition of the gas mixture effects first of all the gas constant R, which does not remain constant, although the parameters of the gas mixture for any state (i.e., for any chemical composition) do satisfy the equation of state as an ideal gas.

$$pV = RT \quad \text{OR} \quad \frac{p}{\rho} = gRT$$

The change of the chemical composition also effects the specific heat of the gas.

We shall call specific heat which take into account the chemical energy of the gas total specific heat and denote them by a dash. Thus, by definition, the total constant volume specific heat c'_V is equal to

$$c'_V = \left(\frac{\partial U_n}{\partial T}\right)_{V=\text{const}} = \left(\frac{\partial U}{\partial T}\right)_{V=\text{const}} + \left(\frac{\partial \Delta H_0^0}{\partial T}\right)_{V=\text{const}} \approx c_V + \left(\frac{\partial \Delta H_0^0}{\partial T}\right)_{V=\text{const}}. \qquad (5.23)$$

Likewise, the total constant pressure specific heat c'_p can be written

$$c'_p = \left(\frac{\partial I}{\partial T}\right)_{p=\text{const}} = \left(\frac{\partial H}{\partial T}\right)_{p=\text{const}} + \left(\frac{\partial \Delta H_0^0}{\partial T}\right)_{p=\text{const}} \approx c_p + \left(\frac{\partial \Delta H_0^0}{\partial T}\right)_{p=\text{con}} \qquad (5.24)$$

Here the quantities $\left(\frac{\partial U}{\partial T}\right)$ and $\left(\frac{\partial H}{\partial T}\right)_{p=\text{CONST}}$ are approximately equal to c_V and c_p respectively.

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As we showed above, when the temperature increases, the content of produtts of dissociation which have a large supply of chemical energy relative to the initial substances is increased in the gas mixture. Therefore, the quantities $\left(\frac{\partial \Delta H_0^0}{\partial T}\right)_{V=\text{CONST}}$ and $\left(\frac{\partial \Delta H_0^0}{\partial T}\right)_{p=\text{CONST}}$ are always positive, and consequently $c'_V > c_V$ and $c'_p > c_p$.

In the range of temperatures which occur in the combustion chamber and nozzle of the rocket engines, the quantities $\partial \Delta H_0^0 / \partial T$ can be of the same order as c_V and c_p, and can even exceed them.

The numerical values of the total specific heat are determined by the nature of the processes of heating or cooling of the reaction mixture of gases.

Let us consider the case where, in the process of heating or cooling, the gas mixture always satisfies the conditions of chemical equilibrium. This is possible if the dissociation reaction or recombination reaction, which are necessary for chemical reactions to be constantly maintained in the gas mixture, is able to take place in the time in which the gas heats up or cools down. Thus a process of heating or cooling of a gas is called a limit equilibrium process.

Theoretical studies and special experiments have shown that the state of the products of combustion in the nozzles of rocket engines changes according to a limit equilibrium process, despite the short time of heating up and cooling down. Therefore, from now on, we shall take the total specific heat c'_p and c'_v to be the specific heat contributed under the condition of equilibrium heating or cooling.

In the case of a limit equilibrium process, the quantitative composition of the gas mixture and, consequently, the total heat content or total internal energy of the gas mixture are completely determined by the chemical structure of the molecules of a gas mixture, as well as by the value of the temperature and pressure.

We note that under these conditions the quantities $\left(\frac{\partial \Delta H^0_s}{\partial T}\right)_{p=\mathrm{CONST}}$ and $\left(\frac{\partial \Delta H^0_s}{\partial T}\right)_{v=\mathrm{CONST}}$ will not be equal, since when a gas is heated in a constant volume its pressure will increase. As we know, an increase in pressure obstructs the dissociation reaction which accompany the increase in the number of moles. Consequently, a gas heated at constant volume will not change its composition as rapidly as a gas heated at constant pressure.

Fig. 5.27 shows a graph of the change of the total heat content of the product of dissociation of water vapor with temperature. In constructing this function, the composition of the products of combustion for different temperatures is determined in accordance with the graph of Fig. 5.22. For comparison, a graph of the change of the physical heat content of undissociated water vapor is given.

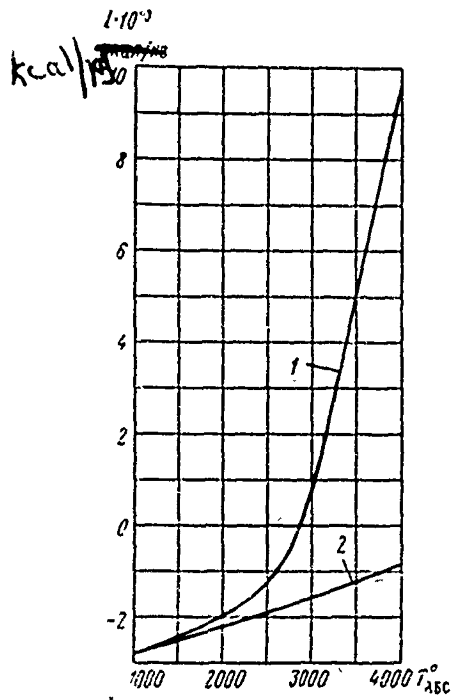


Fig. 5.27. Change of the total heat content of water vapor which is dissociating in equilibrium, and of water vapor which does not change its chemical composition as the temperature T changes.

1- total heat content of water vapor which is dissociating in equilibrium,

2- total heat content of water vapor which does not change its chemical composition.

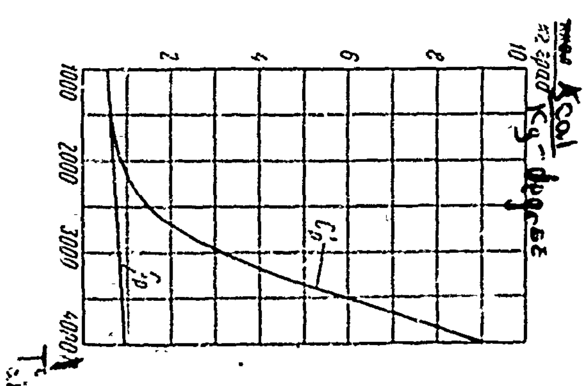


Fig. 5 28. Change in the specific heat c_p' of water vapor which is dissociating and c_p of water vapor which is not dissociating in relation to the temperature T.

Fig. 5.28 shows the relation of the specific heats c'_p and c_p of water vapor to temperature. It is seen that $C'_p > C_p$. At high temperatures the total specific heat is many times greater than the specific heat of a gas of constant chemical composition.

By means of the concept of the total specific heat, it is possible to use the same methods to describe thermodynamic processes in gas mixtures as those used to describe processes in gases of constant composition. In particular, the very important adiabatic process which takes places in gases of variable composition can be described by the equation

$$\left(\frac{p}{\rho}\right)^{k'} = \text{const},$$

where the quantity $k' = C'_p / C'_v$ represents the ratio of the total specific heat and can be called the adiabatic exponent of the gas of variable composition.

To compute k', it is necessary to know the values of the total specific heats. They can be found by performing the corresponding thermodynamic computations. We must note that the equation $C_p - C_v = AR$ which is suitable for gases of constant composition, no longer holds for thermodynamic processes in gases of variable composition.

In computations of the process of expansion in rocket engines which use ordinary fuels and whose products of combustion do not have too high a temperature, (not above 3000° abs.), it is possible to use an average index for the process of expansion n_{is}, called the isotropic index. The quantity n_{is} accounts for (although not quite accurately) the variability of the composition of the products of combustion.

C H A P T E R VI

FLOW OF COMBUSTION PRODUCTS THROUGH THE NOZZLE OF A ROCKET ENGINE

1. Basic Laws of Motion of Gas Flow

Parameters of Gas Flow

Combustions products formed in the combustion chamber of the engine proceed into the nozzle, where their expansion and acceleration occurs. In the process of motion through the nozzle the parameters of gas flow are changed. To parameters of flow of gases in the nozzle, besides the above-indicated magnitudes characterizing the state of the gas: p, V (or ρ) and T, also pertain speed w and area of cross section of flow S.

During motion through the nozzle a decrease of temperature and pressure of the gas occurs. A decrease in temperature leads to decrease of the degree of dissociation of strongly dissociated gas in the chamber — recombination of atoms and radicals into molecules.

Decrease of pressure prevents this process insignificantly. It is clear that recombination leads to additional liberation of heat and promotes fuller conversion of chemical energy of the fuel into kinetic energy of gas flow ensuing from the nozzle.

At the moment we will not consider the phenomenon of recombination and will consider flow of gas of constant composition through the nozzle. As a fundamental equation for determination of parameters of flow we use the equation of the state of the gas

$$\frac{p}{\rho}=gRT \qquad (6.1)$$

or in another way

$$pV=RT.$$

The second equation, into which parameters of gas flow enter, is the equation of thermodynamic process. We will stop on this equation somewhat more specifically.

The state of the gas during thermodynamic processes can be changed in the most diverse way, for instance, under constant volume, under constant pressure or under constant temperature. Depending upon the above, the connection between parameters of the state of the gas turns out to be different.

The most general form of the equation of thermodynamic process is

$$\frac{p}{\rho^n}=\mathrm{const}$$

or

$$pV^n=\mathrm{const}. \qquad (6.2)$$

In the following we will consider only such processes whose index n, remains constant for the whole process.

Taking different values of index n, it is possible to describe basic thermodynamic processes occurring in gases. Thus, for instance, taking $n = 0$, we set $p = \mathrm{const}$; consequently, equation (6.2) in this case will express the isobaric process equation. Such a process has place, for instance, in the combustion chamber of a rocket engine. This process in pV-coordinates on Fig. 6.1 is depicted by line I (isobar).

At n = 1 we get pV = const, or, taking into account the equation of state, we get T = const, i.e., the isothermal process equation (line II — isotherm — on Fig. 6.1).

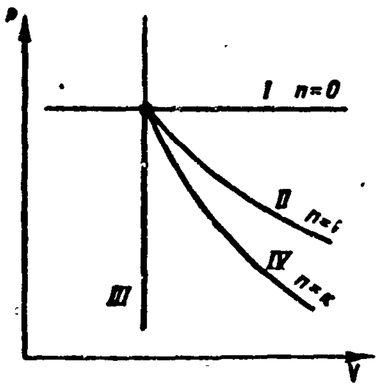


Fig. 6.1. Graph of basic thermodynamic processes in gas.

At $n \rightarrow \infty$ equation (6.2) will be the equation of isochoric process V = const.

Line III (isochore) on Fig. 6.1, differentiates two characteristic regions, the region of expansion processes (increase of specific Volume V) and the region of compression processes (decrease of specific Volume V).

From all possible processes in gases we will be interested mostly in the expansion process of a gas of constant composition, occurring in conditions of absence of heat exchange between gas and environment. One would think that such conditions for the motion of a gas, especially in the nozzle of a rocket engine, are not able to exist, since the gas has a high temperature and touches the intensely cooled walls of the nozzle. However, in reality it is not so. The time of contact of the gas with the walls of nozzle is very small. Due to great speed of flow this time is, even in an engine having significant dimensions, of a magnitude of 0.001 sec. Furthermore, a large part of the gas passes far away from the walls and only in small degree transmits heat to them.

By the shown causes the quantity of heat given by the gas to the walls, is insignificantly small as compared to its general reserve in the gas, and the expansion process of the gas in the rocket engine nozzle is possible to consider occurring without heat exchange with the walls. Such a process of expansion is called adiabatic.

The adiabatic process, occurring in gas, is characterized by the fact that the simplest form of the law of conservation of energy appears in it.

During adibatic expansion a decrease in temperature and pressure of the gas occurs. In connection with this decrease of temperature, internal energy which the gas possesses decreases. It is possible to show that a decrease of pressure during adiabatic expansion of gas evokes also a decrease of potential energy of gas pressure, not considering growth of specific volume. According to the law of energy conservation the difference of energies which the gas possesses in the beginning and at the end of the process, wholly passes into the work of expansion of gas.

Work of gas expansion is used in various machines differently: in a piston engine, for instance, it is turned into work of shifting the piston; in a reactive engine – into kinetic energy of gas flow. The quantity of work obtained during adiabatic expansion of gas is easy to calculate having determined the change of internal and potential energies of the gas in the expansion process. This we will do below during conclusion of the energy equation.

Index n for adiabatic process, occurring in a gas of constant composition is equal to the index of adiabat:

$$n=k=\frac{c_p}{c_V}.$$

As can be seen from Fig. 6.1, the expansion curve with index $n = k$ (line IV) passes between the isotherm and isochore somewhat steeper than the isotherm, since $k > 1$.

Established and Transient Flow of Gas

For determination of the connection between parameters of moving gas it is insufficient to use only thermodynamic relationships; it is necessary to consider also regularities characterizing gas flows.

In the first place we will separate from all possible flows established, or stationary, flow. As established flow of gas it is understood that in every given point of space the gas parameters (speed, pressure, temperature, density) do not change in time. If this constancy of parameters by time is not observed, this motion is called transient.

In many technical problems connected with motion of gas, flow of gases can be considered established, and if not completely, then in any case on the average. This significantly simplifies solution of many practical problems.

Namely as such we will consider motion of gas in the nozzle of a rocket engine. As a result one should exclude from consideration the period of starting and stopping of the engine, and also the moment of transition of the engine from one set of conditions to another, when the flow rate of fuel is changed in time, and consequently, the flow rate of gas and parameters of gas flow. In these cases we have an example of transient flow of gas.

Distribution of Speeds by Cross Section of Flow. One-Dimensional Flow

Gases, as liquid, possess viscosity. Characteristic of viscosity is the coefficient of viscosity μ, determined by the relationship

$$\tau=\mu\frac{\Delta w}{\Delta y},$$

where τ – tangent force, referred to a unit of area which appears between parallel moving layers of gas or liquid located at a distance Δy from each other;

Δw – difference of speeds among these layers.

Viscosity of gases is much less than the viscosity of liquids and appears only where there is a great difference of speeds, for instance directly at the surface of a streamlined body. Particles of gas directly adjacent to this surface seemingly adhere to it and remain motionless.

As a result of removal from the body surface the velocities of gas particles increase fast and then remain as constants equal to the speed of flow.

A film of gas in which a build-up of speed occurs from zero to the speed of free flow is called the boundary layer. Gradual build-up of speed in the boundary layer is explained by the action of forces of internal friction in the gas.

The thickness of the boundary layer at the surface of a streamlined body increases in the direction of flow, and at large linear dimensions of the body, can attain signifucant magnitude.

Motion of gases and liquids, as it is known from physics, can be laminar and turbulent. With laminar flow the movements of gas streams are not mixed; in turbulent flow intense mixing of streams at the expense of transverse shift of small volumes of gas occurs.

Laminar flow may be both established and transient. Turbulent motion is always transient. However, because mixing of layers of liquid or gas occurs in volumes essentially smaller than the general dimensions of flow, turbulent motion can be considered established on the average.

During flow of gas along a wall, in the boundary layer for a certain length the motion is laminar and then passes into turbulent. Transition of motion from laminar to turbulent depends on conditions of gas flow.

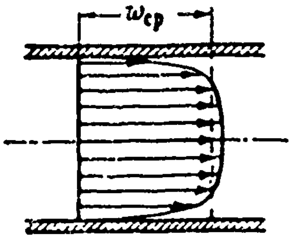


Fig. 6.2. Distribution of speeds in a cross section of turbulent flow. [cp = av = average]

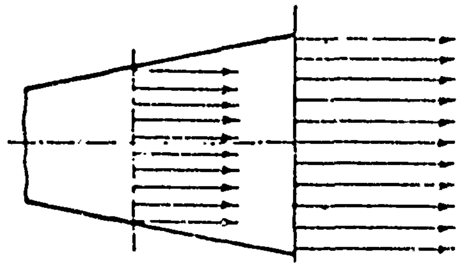


Fig. 6.3. Diagram of one-dimensional flow through the nozzle of the engine chamber.

The law of change of speed in a boundary layer by its thickness is distinguished for laminar and turbulent motions. In a turbulent boundary layer, due to intense mixing of gas layers, the speed grows significantly faster than in the laminar layer.

Conditions of gas flow through the nozzle of a rocket engine are such that the boundary layer is always turbulent with fast increase in speed to full speed of flow (Fig. 6.2). Therefore, in examining of gas motion through the nozzle of a Liquid-Fuel Rocket Engine, the boundary layer usually is not taken into account, and it is considered that in any point of a given section the speed is the same, equal to the average speed of flow w_{av}.

The nozzle of a rocket motor is a channel of variable section, in which radial flow of gas toward the axis of the channel in the narrowed part and from axis in the expanded part should appear. However, radial speeds of gas in the nozzle in first approximation can be disregarded.

Thus, in every cross section of the engine chamber speed is taken as constant and equal to the speed on the axis of the channel (Fig. 6.3). Such flow of liquid is called one-dimensional.

If radial speed is not considered, then the calculated (theoretical) speed of flow will appear higher than the real. It is possible, consequently, to consider that radial flow leads to a certain loss of exhaust velocity and specific thrust of the motor. This loss is included in total sum of losses, occurring in flow of gas through the nozzle.

Flow Rate Equation

Let us consider two sections perpendicular to the direction of speed of one-dimensional flow (Fig. 6.4), and calculate the mass of gas passing through both section during the time Δt. Mass flow rate is determined in this case by volume flow rate $Sw\Delta t$, multiplied by density ρ. Thus, the mass of gas passing through the first section of one-dimensional flow will be $\rho_1 w_1 S_1 \Delta t$, and through second — $\rho_2 w_2 S_2 \Delta t$.

But during established motion during the time Δt no variation of gas parameters in any point between the first and second sections is able to occur. Consequently, accumulation or decrease of gas mass in the volume between sections will not occur. Therefore, entrance of gas through section 1-1 should be equal to its expenditure through section 2-2, whence it follows that for one-dimensional gas flow during established motion,

$$\rho w S = \text{const.} \tag{6.3}$$

The obtained equation is called the flow rate equation. It is the expression of the law of conservation of mass for the case of gas flow.

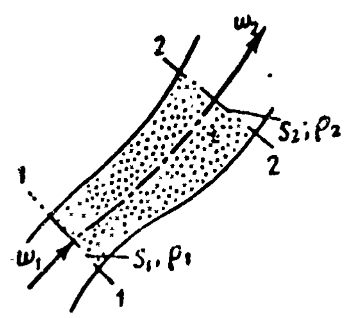


Fig. 6.4. Conclusion of the flow rate equation.

For incompressible liquids ρ = const and the equation of flow rate takes the form

$$wS = \text{const.} \tag{6.4}$$

At low speeds the gas can be considered an incompressible liquid. From equation (6.4) it follows that in this case the speed is changed reciprocally with respect to the area of the stream's cross section. For a compressible gas, due to change of density ρ the illustration is changed not only quantitatively, but also qualitatively: at supersonic speeds in the expanded channel, as we subsequently will see, the gas speed does not decrease, but increases.

Energy Equation

Let us consider energy relationships characteristic for gas flow. These relationships ensue from the law of conservation of energy. We will consider that heat exchange between flow and the walls (environment) is absent, i.e., we will consider adiabatic gas flow. In this case the general reserve of energy E, which a certain mass of gas possesses, in the process of flow cannot be changed and for any section of flow will remain constant.

However, in connection with the fact that in the flow process parameters of flow are changed, redistribution of energy occurs, with its transition from one form to another. Moreover, in dependence on the character of flow certain forms of energy do not change their magnitude. Thus, for instance, if gas flows along a horizontal channel, then the potential energy of its weight does not change. If the flow consists of gases of constant composition, then its reserve of chemical energy remains constant.

During calculation of the magnitude of the reserve of flow energy E, there is no sense in considering those of its forms which do not change their magnitude in the given flow.

In the case of motion of compressible gas considered by us the kinetic energy of flow, potential energy of pressure, potential energy of weight and, lastly, internal energy of thermal motion of gas particles can change their magnitude.

First of all, at speed w the gas possesses kinetic energy equal to

$$\frac{mw^2}{2},$$

or for 1 kg of gas $w^2/2g$, and in thermal units $Aw^2/2g$.

Potential energy of pressure of 1 kg of gas in thermal units is equal to ApV. Potential energy of weight mgz for 1 kg of substance is numerically equal to the height of location of the center of gravity of the mass of gas z, counted off from a certain level. For gas flows having small density, potential energy of weight as compared to potential energy of pressure usually will be disregarded.

Finally, internal energy of the gas is

$$U=c_VT.$$

Thus, if we disregard the potential energy of weight, the general reserve of energy per 1 kg of gas

$$E=\frac{Aw^2}{2g}+U+ApV.$$

The sum of internal energy U and potential energy of pressure ApV is called enthalpy H (see Chapter 5). Therefore,

$$E=\frac{Aw^2}{2g}+H. \qquad (6.5)$$

For adiabatic flow E = const; consequently,

$$H+\frac{Aw^2}{2g}=\text{const}, \qquad (6.6)$$

or for two arbitrary sections of a stream of gas 1-1 and 2-2 (see Fig. 6.4)

$$H_0+\frac{Aw_0^2}{2g}=H_1+\frac{Aw_1^2}{2g}. \qquad (6.7)$$

The obtained equation expresses the law of conservation of energy, affirming that energy does not disappear and does not appear anew but passes from one form into another. Equation (6.7) will be subsequently called the energy equation.

Equation (6.7) is frequently used for determination of speed of gas flow

$$w_1 = \sqrt{w_0^2 + \frac{2g}{A}(H_0 - H_1)}. \tag{6.8}$$

If exhaust of a gas from a vessel of large dimensions is considered, in which speed w_0 is small, then

$$w_1 = \sqrt{\frac{2g}{A}(H_0 - H_1)}. \tag{6.9}$$

We will present the energy equation in another form which will be useful to us subsequently. For that, using relationships derived in Chapter V, we will express enthalpy of gas through parameters of state:

$$H = c_p T = \frac{k}{k-1} ART = A\frac{k}{k-1}\frac{p}{g\rho}.$$

The energy equation can now be copied in such a form:

$$\left.\begin{aligned} \frac{k}{k-1}\frac{p}{\rho} + \frac{w^2}{2} &= \text{const}, \\ \frac{k}{k-1} gRT + \frac{w^2}{2} &= \text{const}. \end{aligned}\right\} \tag{6.10}$$

During the analysis of flow of incompressible liquid one should consider that due to absence of heat exchange with the external medium, the internal energy of the moving incompressible liquid will not be changed. Therefore, in the energy equation for motion of an incompressible liquid it is not necessary to consider the magnitude of internal energy. On the other hand, due to large specific gravity of dropping liquids, it is necessary to consider change of potential energy of weight. The equation of energy conservation for a incompressible liquid is recorded in the following form:

$$\frac{p}{\gamma}+\frac{w^2}{2g}+z=\text{const.} \qquad (6.11)$$

Equation (6.11) is the Bernoulli equation.

2. Flow of Gas Through the Nozzle

Speed of Sound in Gases

A very important characteristic of gas is the speed of propagation of sound in it.

As speed of sound, speed of propagation of longitudinal oscillations in a medium is understood. Also, here we discuss not only oscillations perceived by the human ear as sound, but also oscillations of gas, the frequencies of which lie beyond the threshold of sound.

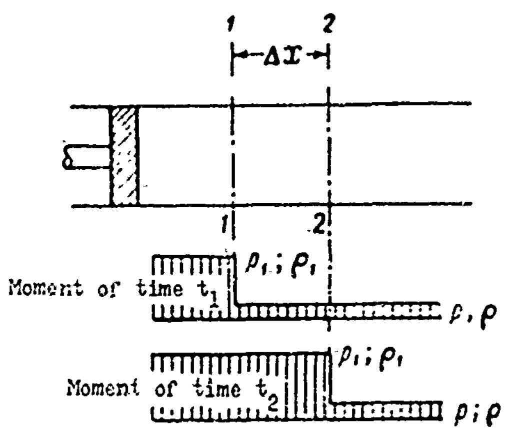


Fig. 6.5. Conclusion on the expression for speed of sound.

Let us assume that a cylindrical tube (Fig. 6.5) contains a motionless mass of gas with pressure p, density ρ, and temperature T. Let us assume that, further, in the left end of the tube a certain pulse is transmitted to the gas, for instance a short shock with the help of a mobile piston. Gas near the piston will become compressed, and then, being expanded, will induce motion to particles of gas located on the right. Along the tube from left to right a wave travels.

In a certain moment t_1 the wave will reach section 1-1. After Δt seconds (in a moment of time t_2) it will shift into section 2-2.

Past section 2-2 the pressure of the gas will be the same as it was before the gas received a pulse. Left of section 2-2 the pressure will be greater than in the undisturbed gas. We will designate it through $p_1 (p_1 > p)$.

The speed of propagation of the wave will be, obviously,

$$a=\frac{\Delta x}{\Delta t},$$

where Δx is the distance between sections 1-1 and 2-2.

The mass of gas in the tube volume, corresponding to section Δx is increased at the expense of flowing into this volume of a certain mass of gas from the left with a certain speed w. The magnitude of this mass

$$\Delta M=\rho_1 Sw\Delta t,$$

where S is the area of the tube section;

ρ_1 is density of gas left of the wave front.

On the other hand, an increase of mass of gas may be expressed through a change of density in volume $S\Delta x$:

$$\Delta M=(\rho_1-\rho) S\Delta x.$$

Equating the right side of the last two expressions, we find

$$w=\frac{\rho_1-\rho}{\rho_1} a. \qquad (6.12)$$

In order to exclude unknown speed w we will use the theorem of momentum.

The mass of gas in volume $S\Delta x$ will be $\rho S\Delta x$. This mass during the time Δt begins movement with speed w. Change of momentum here should be equal to the energy impulse,

$$\rho S\Delta x(w-0)=(p_1-p) S\Delta t,$$

whence

$$p_1-p=\rho wa.$$

Placing in this equality the expression (6.12) for w, we obtain

$$a=\sqrt{\frac{\rho_1}{\rho}\frac{p_1-p}{\rho_1-\rho}}. \tag{6.13}$$

For weak perturbations of gas, such as sonic oscillations, ρ_1 does not greatly differ from ρ, and p_1 from p:

$$\left.\begin{aligned} \rho_1&=\rho+\Delta\rho; \\ p_1&=p+\Delta p \end{aligned}\right\} \tag{6.14}$$

and

$$a=\sqrt{\frac{\Delta p}{\Delta \rho}}. \tag{6.15}$$

Magnitude $\frac{\Delta p}{\Delta \rho}$ depends on the process of compression of gas. Newton, for the first time obtaining this expression in 1687, considered that the temperature of the gas during passage of a wave through it remains constant. In this case

$$\frac{p}{\rho}=\text{const},$$

whence

$$\frac{\Delta p}{\Delta \rho}=\frac{p}{\rho}$$

and

$$a=\sqrt{\frac{p}{\rho}}.$$

This formula gives the value of the speed of sound in air almost 1[illegible]% smaller than what is obtained from experiment. Newton in his own time explained this divergence by the presence in the atmosphere of suspended hard particles and water vapors.

Much later, in 1810, Laplace indicated that the process of compression of gas during passage of a wave should be considered not as isothermal, but as adiabatic, inasmuch as during fast compressions and expansions of gas the heat exchange in gas does not succeed in occurring.

In case of an adiabatic process in a wave

$$p = \rho^k \text{const},$$
$$\Delta p = k\rho^{k-1} \Delta\rho \, \text{const}.$$

Hence

$$\frac{\Delta p}{\Delta\rho} = k\frac{p}{\rho}$$

and

$$a = \sqrt{k\frac{p}{\rho}}, \qquad (6.16)$$

or in accordance with the equation of state p = ρgRT.

$$a = \sqrt{kgRT}. \qquad (6.17)$$

The latter two formulas give for higher values the speed of sound, agreeing well with experiment.

Thus, the speed of sound in a gas depends not on the absolute value of pressure and density, but on their relationships, i.e., on temperature.

For air k = 1.4; R = 29.27 kg-m/kg deg, and the expression for the speed of sound takes the form

$$a = 20{,}1\sqrt{T}.$$

The speed of sound in air at 0°C is equal to 330 m/sec. For combustion products in the thrust chamber at T = 3000° absolute, k = = 1.2 and R = 34 kg-m/kg deg, spped of sound a = 1100 m/sec.

Speed of sound has clear physical meaning, constituting the speed of propagation of weak perturbations in a gas. Furthermore, speed of sound has also definite energy meaning. For clarification of it, we will extract an expression for the square of speed of sound and the magnitude of enthalpy of gas:

$$a^2 = kgRT;$$
$$H = \frac{k}{k-1}ART.$$

Eliminating from these two equations magnitude T, we find

$$a^2 = \frac{g}{A}(k-1)H.$$

The obtained expression means that the square of the speed of sound is a measure of enthalpy of the gas.

The idea of speed of sound has great value in aerodynamics and gas dynamics. Streamlining of bodies by gas, exhaust of gases through pipes, cap, and nozzle and in general the character of any form of motion of a gas is in the closest contact with the relationship of the speed of gas to speed of sound in gas. Depending upon the magnitude of this relationship, we refer to subsonic and supersonic conditions of exhaust and speeds of flight. The relation of speed of flow to speed of sound is designated in all aerodynamic and gas-dynamic calculations by the letter M and called "the M number,"

$$M = \frac{w}{a}. \qquad (6.18)$$

For the first time (in 1868) the relation of speed of gas (or a body moving in a gas) to the speed of sound was introduced in scientific everyday use to Russian scientists-ballistics by N. V. Maiyevskiy. Later it was used also by the Austrian physicist Mach and widely known in technology undet the name of Mach number.

We will return to expression (6.13).

If pressure p_1 and density ρ_1 noticeably differ from pressure p and density ρ in an undisturbed gas, then the wave of perturbation is called a strong, or shock, wave in distinction from a weak (sonic) wave. The speed of propagation of a shock wave

$$a_{y\text{д}} = \sqrt{\frac{\rho+\Delta\rho}{\rho}\frac{\Delta p}{\Delta\rho}}. \qquad (6.19)$$

[уд = sw = shock wave]

From comparison of expressions (6.15) and (6.19) it follows that the speed of propagation of a shock wave is always greater than the speed of sound.

If a gas is given a strong perturbation, i.e., evoking a large difference of pressures $p_1 - p$, then the formed wave during its propagation will partially disperse energy to the gas. The force of the wave, measured by the difference of pressures $p_1 - p$, will decrease. Correspondingly its speed will decrease, and the shock wave after a certain time will be turned into a weak wave, spreading with the speed of sound.

Maximum Exhaust Velocity

Let us suppose a vessel, for instance the combustion chamber of a rocket engine, inside which there is a motionless gas ($w_0 = 0$) with constant parameters p_0, ρ_0, T_0. Let us assume that from this vessel exhaust of gas occurs through a hole in the region where the gas parameters will be p, ρ, T.

From the energy equation it follows that the speed of the gas will be the biggest in those sections of the stream where its enthalpy will be least. The maximum speed will be obtained if all enthalpy will be turned into kinetic energy of the stream of the exhaust gas. Also, the absolute temperature of the gas has to become equal to zero. For an adiabatic process, as is considered exhaust of gas from a vessel, from comparison of (6.1) and (6.2) we can find an expression for temperature

$$T = T_0\left(\frac{p}{p_0}\right)^{\frac{k-1}{k}}, \tag{6.20}$$

whence it follows that for getting a temperature of the gas equal to zero, it is necessary that the pressure in the gas flow p also be equal to zero. Thus, the maximum speed of the gas may be obtained during exhaust in a vacuum, under the condition that in the gas flow itself a pressure equal to zero will be attained.

According to expression (6.9) the magnitude of maximum speed

$$w_{max}=\sqrt{\frac{2g}{A}H_0},$$

where H_0 -- enthalpy of a motionless gas,

The magnitude w_{max} can be expressed through state parameters and speed of sound in a motionless gas:

$$w_{max}=\sqrt{\frac{2k}{k-1}\frac{p_0}{\rho_0}}=\sqrt{\frac{2k}{k-1}gRT_0}=a_0\sqrt{\frac{2}{k-1}}. \qquad (6.21)$$

For air at room temperature $w_{max} \approx 750$ m/sec; for products of combustion of rocket engine fuel ($T_0 = 3000^\circ$ abs, R = 34 kg-m/kg deg, k = 1.2) $w_{max} \approx 3500$ m/sec.

Maximum exhaust velocity, as follows from formula (6.21), depends only on temperature T_0 and does not depend on pressure. From the energy point of view this is absolutely clear. Maximum speed has place at full transformation into kinetic energy of all initial enthalpy of the gas, but the magnitude of enthalpy is determined only by its initial temperature. At maximum speed there occurs a full converting of thermal chaotic motion of molecules into directed motion of flow.

At first glance it seems that with an increase of pressure the speed of flow w_{max} should increase inasmuch as, talking conditionally, the force ejecting gas from the vessel increases. However, with a growth of pressure, (at T_0 = const) density ρ_0 is increased in that same measure and, consequently, the mass, contained in a unit of volume. It is clear that increased pressure communicates by an increased mass in that same order the same speed w_{max}.

Dependence of Gas Parameters on Local Speed of Flow

Let us consider, how parameters of moving gas will change depending upon speed of flow.

We will write the energy equation for two states of flow — motionless gas and gas moving with speed w:

$$\frac{k}{k-1}gRT_0=\frac{w^2}{2}+\frac{k}{k-1}gkT.$$

Here w and T pertain to a certain arbitrary section, and T_0 — temperature of gas at w = 0. For temperature T the following expression is obtained:

$$T=T_0\left(1-\frac{w^2}{\frac{2k}{k-1}gRT_0}\right)$$

or

$$T=T_0\left(1-\frac{w^2}{w_{max}^2}\right), \tag{6.22}$$

Consequently the greater the speed of gas in the flow, the lower its temperature.

In adiabatic flow of gas [see relationships (6.1) and (6.2)]

$$\frac{\rho}{\rho_0}=\left(\frac{T}{T_0}\right)^{\frac{1}{k-1}};$$

$$\frac{p}{p_0}=\left(\frac{T}{T_0}\right)^{\frac{k}{k-1}}.$$

Consequently,

$$p=p_0\left(1-\frac{w^2}{w_{max}^2}\right)^{\frac{k}{k-1}}; \tag{6.23}$$

$$\rho=\rho_0\left(1-\frac{w^2}{w_{max}^2}\right)^{\frac{1}{k-1}}. \tag{6.24}$$

Thus, both pressure and density of the gas decrease with an increase of speed of flow. At achievement of maximum speed, in particular, the pressure and density of the gas become zero.

Dependence of Local Speed of Sound on Speed of Flow. Stalling Speed

As we already know, the speed of sound in a gas is determined only by its temperature. But the temperature of the gas in different points of the flow may be different. Consequently, the speed of sound will also be different. Therefore, for a moving flow one should consider not only local speed, temperature, pressure, and density, but also local speed of sound.

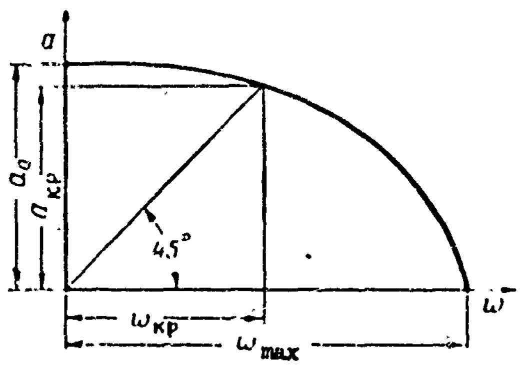


Fig. 6.6. Dependence of speed of sound in gas on speed of flow. [Кр = cr = critical]

After introduction of the idea of maximum speed, the energy equation can be recorded in the following form:

$$\frac{w^2}{2}+\frac{k}{k-1}gRT=\frac{w_{max}^2}{2}. \qquad (6.25)$$

Hence, taking into account that $kgRT = a^2$,

$$a=\sqrt{\frac{k-1}{2}(w_{max}^2-w^2)}. \qquad (6.26)$$

Dependence of speed of sound on speed of gas is shown on Fig. 6.6. At $w = w_{max}$ the speed of sound drops to zero, inasmuch as with this $T = 0$.

Speed of flow may be both greater and less than the local speed of sound. Speed of flow, equal to the local speed of sound, is called the stalling speed. It can be found from expression (6.26) putting $a = w = w_{cr}$:

$$w_{кр}^2=\frac{k-1}{k+1}w_{max}^2.$$

But since

$$w^2_{max} = \frac{2k}{k-1} gRT_0,$$

then

$$w^2_{кр} = \frac{2k}{k+1} gRT_0 = \frac{2}{k+1} a_0^2. \quad (6.27)$$

Thus, magnitude w_{cr} depends only on the temperature of the gas in the vessel from which expiration occurs.

We will find the local parameters of gas at the stalling speed. According to expression (6.22)

$$T_{кр} = T_0\left(1 - \frac{w^2_{кр}}{w^2_{max}}\right),$$

or, inasmuch as $w^2_{cr} = \frac{k-1}{k+1} w^2_{max}$,

$$T_{кр} = T_0 \frac{2}{k+1}. \quad (6.28)$$

Considering expression (6.23) and (6.24), we obtain also

$$\frac{p_{кр}}{p_0} = \left(\frac{2}{k+1}\right)^{\frac{k}{k-1}}; \quad (6.29)$$

$$\frac{\rho_{кр}}{\rho_0} = \left(\frac{2}{k+1}\right)^{\frac{1}{k-1}}. \quad (6.30)$$

In particular, for air (k = 1.4)

$$p_{кр} = 0{,}528\, p_0;$$
$$T_{кр} = 0{,}833\, T_0;$$
$$\rho_{кр} = 0{,}634\, \rho_0;$$

for products of combustion of a rocket engine (k = 1.2)

$$p_{кр} = 0{,}565\, p_0;$$
$$T_{кр} = 0{,}909\, T_0;$$
$$\rho_{кр} = 0{,}621\, \rho_0.$$

Thus, for obtaining of supersonic air flow in the chamber atmosphere, it is necessary to have a pressure, approximately twice exceeding atmospheric.

Form of Supersonic Nozzle

Till now we talked of the dependence of parameters of gas flow on speed but did not consider the question of how to create this speed.

It is widely known that for an incompressible liquid or gas an increase of speed is possible by narrowing the channel. However, experiment shows that in a narrowed channel it is impossible to obtain a speed of gas greater than critical. Supersonic speeds of gas flow can be obtained with the help of a Laval's nozzle, constituting a channel, the section of which at first decreases and then is increased (Fig. 6.7).

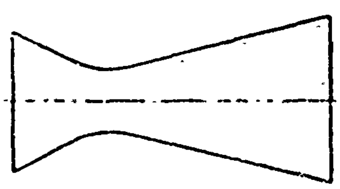


Fig. 6.7. Form of Laval's nozzle.

It has been revealed that if gas were passed through a Laval's nozzle in such a way so that stalling speed was attained in the narrowest section that subsequently, after the neck, the speed of flow w starts to increase. Thus, by experimental means it was established that for supersonic flows a rule of flow exists, directly opposite to the rule of flow in subsonic flows. In other words, flow, passing with subsonic speed, is accelerated in a narrowed channel and is delayed in an expanded channel. Supersonic flow, conversely, in a narrowed channel is delayed, and in an expanded channel is accelerated.

Let us consider this question in greater detail. We will write the equation of flow rate (6.3) in the form

$$S=\frac{\text{const}}{\rho w}. \qquad (6.34)$$

For an incompressible liquid (ρ = const) the speed is reciprocal to the area of a section of passage. For a compressible gas at small speeds the density ρ, as follows from formula (6.24) drops with an increase of speed, but so insignificantly that the character of gas flow qualitatively remains the same as for an incompressible liquid, and the area of the section of flow with an increase of speed decreases. At great, and namely at supersonic speeds, magnitude ρ with an increase of speed decreases faster than w increases. Consequently, section of flow S in this case should increase.

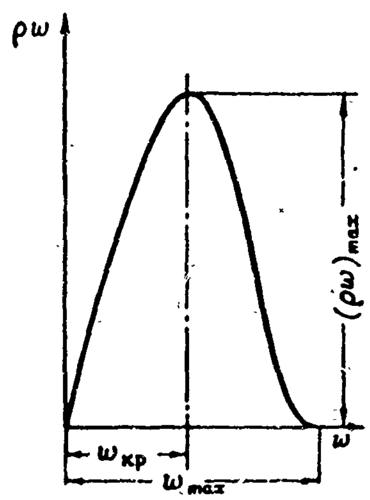


Fig. 6.8. Dependence of magnitude ρw on speed of flow w.

We will consider the dependence of ρw on w.

According to expression (6.24),

$$\rho=\rho_0\left(1-\frac{w^2}{w^2_{max}}\right)^{\frac{1}{k-1}},$$

consequently,

$$\rho w=\rho_0 w\left(1-\frac{w^2}{w^2_{max}}\right)^{\frac{1}{k-1}}.$$

Magnitude ρw can be called the mass specific flow rate, understanding by this the flow rate of a mass of gas in a unit of time through a unit of area of passage section. At w = 0 the specific flow rate becomes zero. At $w = w_{max}$ it also becomes zero, since here the density ρ becomes equal to zero. In the interval between these two limiting values of speed there exists, obviously, a maximum of the function.

On Fig. 6.8 the dependence of the specific flow rate on speed is shown.

We will consider, at what value of speed is the maximum of specific flow rate. Differentiating the expression for ρw by w and equating the derivative to zero, we obtain

$$\rho_0\left(1-\frac{w^2}{w_{max}^2}\right)^{\frac{1}{k-1}}-\rho_0 w^2\frac{1}{k-1}\left(1-\frac{w^2}{w_{max}^2}\right)^{\frac{1}{k-1}-1}\frac{2}{w_{max}^2}=0,$$

whence

$$\left(1-\frac{w^2}{w_{max}^2}\right)^{\frac{1}{k-1}-1}\left[1-\frac{w^2}{w_{max}^2}-\frac{2}{k-1}\frac{w^2}{w_{max}^2}\right]=0. \qquad (6.32)$$

Consequently, either $w = w_{max}$ or the expression standing in brackets (6.32) is equal to zero.

In the first case $\rho w = 0$, as this is shown on the curve of Fig. 6.8. In the second case at

$$w^2=\frac{k-1}{k+1}w_{max}^2 \qquad (6.33)$$

the specific flow rate ρw attains maximum value.

But the speed determined by relationship (6.33) is nothing else but the stalling speed, equal to the local speed of sound. The maximum point on the curve of Fig. 6.8 separates, consequently, subsonic speeds of flow from supersonic.

Now, considering curve ρw, it is simple to establish, how the cross section of channel should be changed for achievement of supersonic speeds. The area of the section according to (6.31) should at first decrease and then increase. Where magnitude ρw has maximum value, the area of section S should be minimum. Here local speed of sound and stalling speed are attained. Therefore, the minimum section of the nozzle is also called the critical section.

Thus, we see that in Laval's nozzle achievement of supersonic speeds is possible. However, this is not always possible. If the difference of pressure at the entrance and outlet of the nozzle are insufficient for creation of stalling speed in the narrow section, it is not always possible to attain supersonic speed.

On Fig. 6.9 possible cases of nozzle function are shown. Curves of solid lines correspond to the basic functioning case, when the relation of outlet pressure to inlet pressure is less than the critical relation,

$$\frac{p}{p_0}<\left(\frac{2}{k+1}\right)^{\frac{k}{k-1}}.$$

Here the pressure in gas continuously drops, and speed increases.

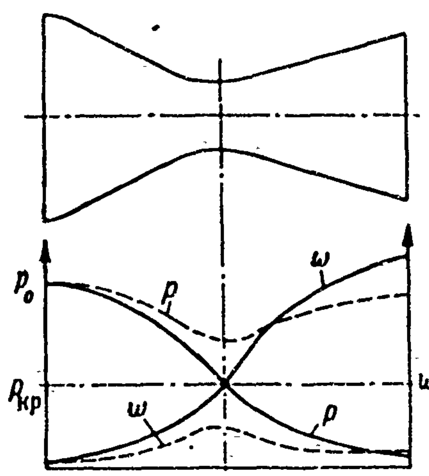


Fig. 6.9. Variation of flow parameters along supersonic nozzle at different values of gas parameters at entrance.

In those cases, when relation of pressures

$$\frac{p}{p_0}>\left(\frac{2}{k+1}\right)^{\frac{k}{k-1}},$$

i.e., when the outlet pressure of the nozzle turns out to be too large or at the entrance too small, the stalling speed in the narrow section of the nozzle will not be attained, although the speed here also will decrease, and pressure and density will increase (dotted curves on Fig. 6.9).

3. Work of a Rocket Engine Nozzle

Area of Critical Section of Nozzle

Geometric dimensions of a supersonic nozzle of rocket engine are determined by areas of critical and output sections.

We will find, first, the necessary area of critical section. For that we will express weight second flow rate of fuel (gases) G through gas parameters in the critical section of the nozzle,

$$G = g\rho_{кр} w_{кр} S_{кр}.$$

But according to (6.30) and (6.27)

$$\rho_{кр} = \rho_0 \left(\frac{2}{k+1}\right)^{\frac{1}{k-1}} \text{and } w_{кр} = \sqrt{\frac{2k}{k+1} gRT_0}.$$

Therefore

$$G = g\rho_0 S_{кр} \left(\frac{2}{k+1}\right)^{\frac{1}{k-1}} \sqrt{\frac{2k}{k+1} gRT_0}. \tag{6.34}$$

Since

$$\rho_0 = \frac{p_0}{gRT_0},$$

then

$$G = \frac{p_0 S_{кр}}{\sqrt{RT_0}} \left(\frac{2}{k+1}\right)^{\frac{1}{k-1}} \sqrt{\frac{2k}{k+1} g}. \tag{6.35}$$

Equation (6.35) connects flow rate of gases G with the area of critical section of the nozzle S_{cr} and parameters of gas p_0 and T_0 in the combustion chamber before entrance into the nozzle. These parameters, as we already know, are determined from calculation of combustion in a Liquid-Fuel Rocket Engine. In Chapter V they were designated correspondingly through p_k and T.

Equation (6.35) can be rewritten in another form,

$$\beta = \frac{p_0 S_{кр}}{G} = B_k \sqrt{T_0},$$

where

$$B_k = \left(\frac{k+1}{2}\right)^{\frac{1}{k-1}} \sqrt{\frac{k+1}{2k} \frac{R}{g}}.$$

Magnitude $\beta = p_0 S_{cr}/G$ has the dimension of specific thrust – kgsec/kg – and is called the complex of rocket engine parameters. It frequently is applied during calculations and analysis of work of engines.

It is necessary to note that magnitude β is proportional to the root of the square of temperature of combustion T_0, and coefficient B_k

depends on composition of combustion products. Consequently, magnitude β is basically determined by properties of fuel; for chemical rocket fuels, magnitude β changes in the range from 150 to 200 kgsec/kg.

For a given fuel a certain change occurs as a result of dependence of dissociation of combustion products on the pressure in the combustion chamber. Since with an increase of pressure due to decrease of dissociation the temperature T_0 increases, then β is somewhat increased. This change is usually not very great -- it does not exceed 1-2%. Therefore, the magnitude of the complex of parameters is considered a characteristic of the fuel, and in the table of basic properties of rocket fuels the values of β are shown (see Table 4.5). These values can be used for approximate analysis of work of rocket engines. The exact value β for given parameters of the engine is determined from calculation of combustion and exhaust.

Knowing the magnitude of the complex of parameters β, it is easy to find the magnitude of critical section S_{cr} necessary for transmission 1 second G kg of combustion products of fuel at a chamber pressure p_0 kg/cm^2. It is possible to also resolve another problem: to determine what pressure will exist in the combustion chamber if G kg/sec of a given fuel is expended in an engine with a given section S_{cr}. Using magnitude β, it is possible to resolve also a great number of practically important problems.

From the equation for magnitude β it is clear that an increase of fuel flow rate of G in an engine with a constant critical section of the nozzle S_{cr} leads to a proportional increase of pressure in the combustion chamber p_0. This is a direct result of the fact that the speed in the critical section [see formula (6.27)] does not depend on flow rate of fuel and at a constant temperature of combustion T_0 is constant. Below we will see that a similar conclusion remains just for every cross section of nozzle, i.e., the change of flow rate G in general does not render an influence on the law of change of flow speed through the length of the nozzle. Also, the temperature of flow in any cross section of the nozzle, as shows relationship (6.22), also does not depend on the flow rate. Thus, with change of mass the volume flow rate of gas through the critical section S_{cr}, as through any cross section of the nozzle, will not change, if only the temperature in the combustion chamber remains constant. This means that for transmission of a large flow rate through the nozzle G, it is necessary to increase the density ρ of the moving gas.

At constant temperature of the gas its density may be increased only at the expense of increase of pressure in the chamber. As a result the pressure, and consequently also the density of gas will be increased proportionally in all sections of flow, including in the critical.

Let us note that an increase of temperature T_0 at constant flow rate of gas also leads to growth of pressure in the chamber and in the nozzle.

The composition of combustion products affects the necessary area of nozzle section through magnitudes of the adiabat index k and gas constant R. The substances contained in the combustion products with a small number of atoms in the molecule and with small molecular weight, for which large values k and R are characteristic, the less the flow rate of gases through the nozzle for the given S_{cr} and p_0 (magnitude B_k as a result increases due to change of R).

With increase of pressure p_0 in the chamber, the magnitude of the given flow rate G may be obtained for a smaller dimension of critical section S_{cr}, which leads to a decrease also of all other nozzle dimensions.

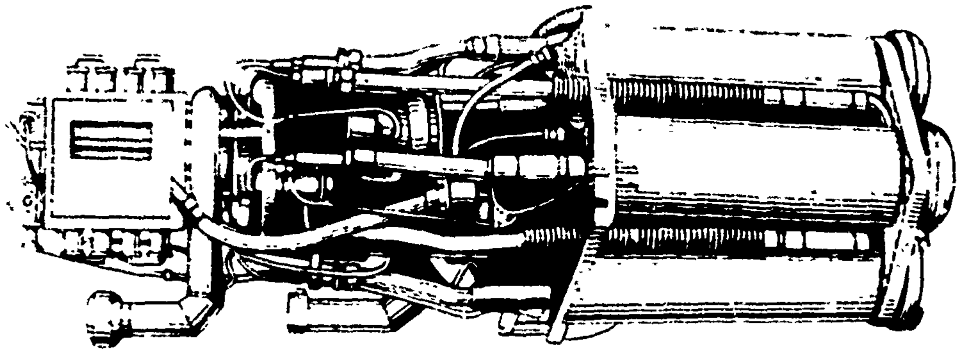


Fig. 6.10. Exterior of a four-chamber engine.

In many types of rocket engines (in particular, in aircraft) it is necessary to regulate engine thrust. The most simplest way of all to do this, is by changing the flow rate of fuel G. However, such adjustment of thrust, especially if it is carried out in wide limits of flow rate change, leads to significant change of pressure in the chamber. Decrease of pressure in the chamber unfavorably affects the burning process (in particular, it may cause vibration burning); it leads also to impairment of nozzle function (the nozzle starts to work in conditions of overexpansion – see p.265). Therefore, if it is necessary to regulate thrust in wide limits, then for preservation of constant pressure in the chamber, the engine's nozzle should be made with a variable critical section.

Mechanization of the nozzle (adjustment of the critical section in the process of engine function) is a very difficult problem. Therefore, this question is resolved practically by means of step adjustment of thrust in propulsion systems with several combustion chambers (Fig. 6.10). With necessity of decrease of thrust one or several chambers can be turned off, and the remaining continue to work with normal or close to normal pressure in the combustion chamber.

The magnitude of the complex of parameters β can be calculated not only theoretically, but also can be determined experimentally during testing of the rocket engine in a testing unit. For that it is necessary to determine the chamber pressure p_0, flow rate of fuel G, consisting of flow rate of fuel and flow rate of oxidizer, and to measure the area of critical section S_{cr}. Thus, it is possible to determine the real value β with great accuracy. If one were to compare the real magnitude β and its theoretical value, it is possible to analyze how fully burning occurs in the engine.

If the real value β is considerably less than its theoretical magnitude, then, obviously, theoretical temperature T_0 is by far not attained in the combustion chamber, calculated taking into account dissociation, testifying to a great physical unburning of fuel and bad work of the cap and combustion chamber of the engine. Thus, using the magnitude of the complex of parameters β, it is possible by relatively simple means to control the quality of combustion in the engine.

Area of Output Section of the Nozzle

Above it was shown that obtaining of a maximum exhaust velocity w_{max} is connected with full expansion of gas to $p = 0$ and, consequently, requires an infinitely large area of output section of the nozzle S_a. This is a very essential case, since now we see that for obtaining of maximum speed it is necessary to cool the gas to $T = 0$ in the actual flow, i.e., to obtain a total vacuum not in the surrounding medium, but on the nozzle section of the engine.

The tendency to increase exhaust velocity can lead to impractical growth of the dimensions and weight of the nozzle. For this reason maximum speed remains practically unattainable, and velocity of expiration is established taking into account rational dimensions of the nozzle of the engine.

Let us find the connection between dimensions of output section of the nozzle S_a, exhaust velocity w_a, and gas pressure on the nozzle section p_a. For this we express exhaust velocity w_a using equation (6.9) through enthalpy of gas in the combustion chamber and on its exit out of the nozzle,

$$w_a=\sqrt{\frac{2g}{A}(H_0-H_a)}.$$

Since

$$H_a=c_pT_a;$$
$$T_a=T_0\left(\frac{p_a}{p_0}\right)^{\frac{k-1}{k}};$$
$$c_p=\frac{k}{k-1}AR,$$

then the exhaust speed

$$w_a=\sqrt{2g\frac{k}{k-1}KT_0\left[1-\left(\frac{p_a}{p_0}\right)^{\frac{k-1}{k}}\right]}. \tag{6.36}$$

We will next express the secondal flow rate of fuel G by the gas parameters in the output section of the nozzle,

$$G=g\rho_a w_a S_a.$$

Considering that the equation of thermodynamic process gives

$$\rho_a=\rho_0\left(\frac{p_a}{p_0}\right)^{\frac{1}{\varkappa}},$$

$[\varkappa = k]$

and also using expression (6.36), we find

$$G=g\rho_0 S_a\left(\frac{p_a}{p_0}\right)^{\frac{1}{k}}\sqrt{2g\frac{k}{k-1}RT_0\left[1-\left(\frac{p_a}{p_0}\right)^{\frac{k-1}{k}}\right]}. \tag{6.37}$$

Equating the right side of (6.37) with the right side of formula (6.34) we get

$$\frac{S_a}{S_{кр}}=\frac{\sqrt{\frac{k-1}{2}\left(\frac{2}{k+1}\right)^{\frac{k+1}{k-1}}}}{\sqrt{\left(\frac{p_a}{p_0}\right)^{\frac{2}{k}}-\left(\frac{p_a}{p_0}\right)^{\frac{k+1}{k}}}}. \tag{6.38}$$

As can be seen from formula (6.38), for combustion products of a given fuel having fully definite composition and, consequently, magnitude of adiabat index k, the relationship of pressures p_a/p_0 depends only on the relationship S_a/S_{cr}. The latter we call broadening of the nozzle.

Relationship (6.38) allows us to determine gas parameters in any section of the nozzle S_x. For that in (6.38) one should replace S_a by S_x, and p_a by p_x. Knowing magnitudes S_x and S_{cr}, it is possible to calculate p_x/p_0, in a given section; then by the formula analogous to (6.36), we determine the velocity w_x, and next by relationships (6.22), (6.23), and (6.24), find all remaining parameters of flow in a section with area S_x.

Thus, we see that speed, and consequently also temperature in any section of gas flow of constant composition (k = const) do not depend on change of mass flow rate. They are determined by dimensions of and by the law of change of S_x along the length of the nozzle, and also by the magnitude of temperature in the combustion chamber.

One more important conclusion can be made. Inasmuch as the relationship of pressures p_a/p_0 does not depend on flow rate, and pressure in the combustion chamber p_0 is directly proportional to flow rate, then the pressure in the flow along the nozzle section p_a will change also proportionally to the change of flow rate.

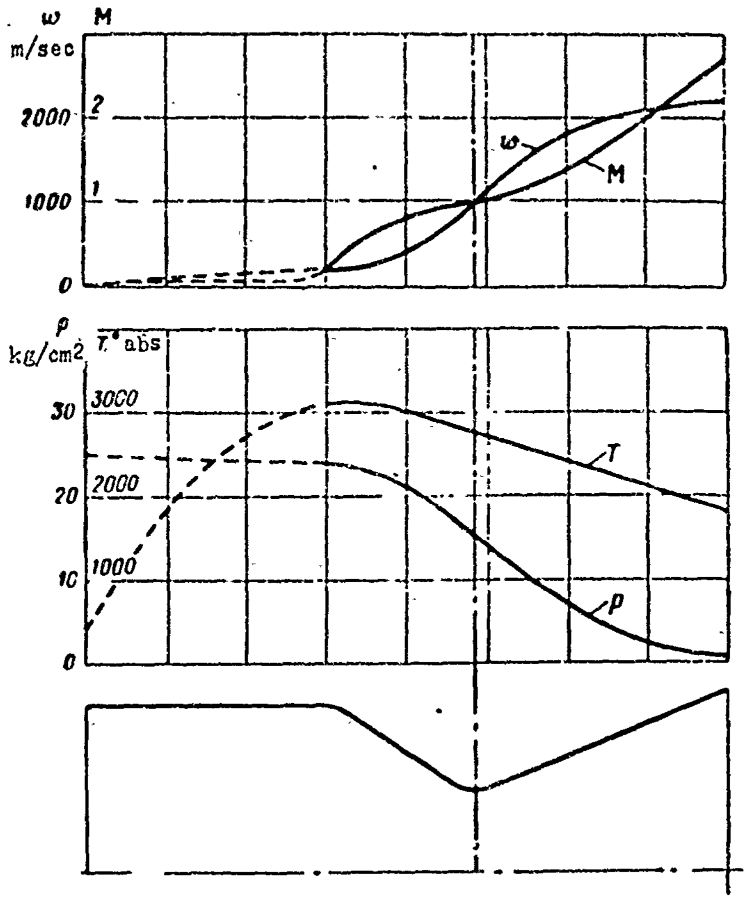


Fig. 6.11. Variation of parameters of gas flow along length of rocket engine using nitric acid and kerosene ($k = 1.2$).

Equation (6.38) fits both for the subsonic and for the supersonic parts of the nozzle. Thus, we can establish a law of variation of parameters of gas flow along the length of the whole nozzle.

A typical graph of variation of parameters of gas flow along the axis of a rocket engine is represented on Fig. 6.11. Variation of gas parameters along the length of the combustion chamber is shown by a dotted line, inasmuch as there these dependences are not sufficiently clear. From the graph it is clear that the fastest variation of parameters (excluding the Mach number) occurs in the region of critical section of the nozzle. In the divergent section of the nozzle change of parameters occurs more and more slowly.

Broadening of Nozzle. Thermal Efficiency of the Rocket Engine

Expressions obtained above for parameters of gas flow allow us to estimate the degree of use of gas enthalpy in the rocket engine.

During motion along the nozzle, the speed, and consequently also the kinetic energy, of the flow increase. Growth of kinetic energy is accompanied by a corresponding decrease of gas enthalpy, and the higher speed the gas obtains on getting out of the nozzle, the greater the share of its enthalpy converts into useful kinetic energy of exhaust.

Total transformation of all thermal energy into kinetic, i.e., achievement on a nozzle section of the maximum speed w_{max}, as we already know, is impossible. This would demand creation of a nozzle with infinitely large dimensions of the output section. Consequently, in real engines it is necessary to be limited to the use of only part of all the enthalpy of the gas. Besides, the greater the broadening of the nozzle, i.e., relation S_a/S_{cr} the lower are the temperature and pressure of gas on getting out of the nozzle, and the smaller the share of enthalpy is lost with gas flow leaving the nozzle.

The degree of enthalpy transformation of gas into kinetic energy is characterized by the thermal efficiency of the engine η_t. It is natural to determine the thermal efficiency as a relation of the kinetic energy of gas flow to that enthalpy which the gas has on entrance into the nozzle:

$$\eta_t = \frac{A\frac{w_a^2}{2g}}{H_0}.$$

Since, according to the energy equation

$$A\frac{w_a^2}{2g} = H_0 - H_a,$$

then

$$\eta_t = \frac{H_0 - H_a}{H_0} = 1 - \frac{H_a}{H_0} = 1 - \frac{T_a}{T_0}.$$

Usually thermal efficiency is expressed through the relationship of pressure on entrance into the nozzle and on getting out of it p_0/p_a, or the reciprocal of p_a/p_0. By the equation of adiabat

$$\frac{T_a}{T_0} = \left(\frac{p_a}{p_0}\right)^{\frac{k-1}{k}}$$

and

$$\eta_t = 1 - \left(\frac{p_a}{p_0}\right)^{\frac{k-1}{k}}. \qquad (6.39)$$

Dependence of the magnitude of thermal efficiency of the engine on the drop of pressures and adiabat index is represented on Fig. 6.12.

We note also that thermal efficiency may be expressed also as the relation of the square of exhaust velocity to the square of maximum speed.

Relation of pressures p_0/p_a is directly connected to the magnitude S_a/S_{cr} by formula (6.38). The value of the index of adiabat of combustion products has much influence on the character of the connection of these magnitudes. On Fig. 6.13 the dependence of p_0/p_a on S_a/S_{cr} for two values of k is represented.

Relation p_0/p_a is one of the initial parameters for designing of the nozzle.

Curves of Fig. 6.12 and 6.13 show that magnitude k strongly affects the relationship of nozzle dimensions and the thermal efficiency. As a result, growth of the adiabat index of combustion products leads to a decrease in dimensions of the supercritical part of the nozzle and an increase of thermal efficiency. Thus, an increase of k beneficially affects the dimensions of the nozzle and degree of use of thermal energy in the engine.

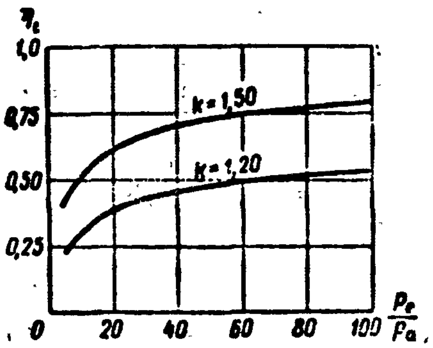


Fig. 6.12. Dependence of thermal efficiency of a rocket engine on the relationship of pressure on entrance and getting out of the nozzle p_0/p_a for two values of k.

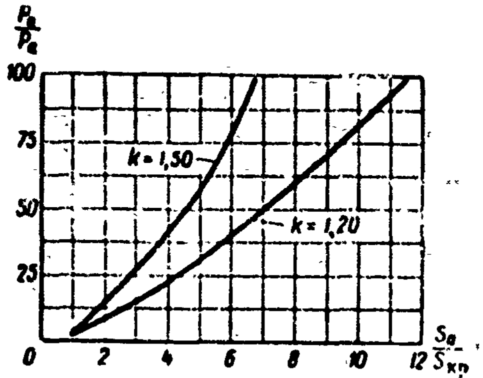


Fig. 6.13. Dependence of relationship of pressure on entrance and getting out of the nozzle p_0/p_a on broadening of the nozzle for two values of k.

Let us note that an increase of pressure in the chamber p_0 at a constant flow rate leads not only to reduction in area of the nozzle's critical section S_{cr}, but also to a decrease in the area S_a even under the condition of maintenance of constant pressure p_a on the nozzle section. This circumstance favorably affects the perfection of the rocket engine for several reasons. First, an increase of p_0 at a constant p_a leads to an increase in the engine's thermal efficiency, i.e., to the best use of gas enthalpy and an increase of exhaust velocity w_a. Thus, in these conditions the specific engine thrust increases and, consequently, for obtaining of a given thrust P, it is possible to decrease flow rate of fuel G. This leads to an additional decrease of nozzle dimensions and promotes simplification of the propulsion system, calculated for a given thrust.

Furthermore, with a decrease of S_{cr} and S_a (with preservation of the law of profile for the nozzle) the lateral surface of the supercritical (supersonic) part of the nozzle decreases also; consequently, the weight and heat-transfer surface of the nozzle decrease.

Earlier we already noted that an increase of pressure in the combustion chamber allows us to decrease its volume, not decreasing the time of stay τ and thereby not worsening the quality of fuel combustion. An increase of pressure in the combustion chamber to a certain degree improves the combustion process, since, as a result the share of chemical energy of the fuel converting into heat is increased (due to lowering losses of heat from dissociation of combustion products). Consequently, an increase of pressure p_0 leads to an increase of physical enthalpy of gases in the combustion chamber before entrance into the nozzle H_0, which additionally increases the exhaust velocity and the specific engine thrust.

As can be seen, an increase of pressure in the combustion chamber allows the possibility to decrease the dimensions of the chamber of a Liquid-Fuel Rocket Engine. Therefore, an increase of p_0 is a constant in the development of liquid-propellant rocket engines. It is limited only by an increase of weight of the supply system and difficulties in creation of light and reliable supply systems, which can ensure the necessary pressure of fuel components on entrance into the engine chamber.

Influence of Recombination of Gases and Burning of Fuel on the Flow of Gas Along the Nozzle of a Rocket Engine

Till now we talked about flow of gas of constant composition along the nozzle. However, during movement of combustion products along the nozzle the process of partial recombination of dissociated gases and burning of fuel not succeeding to burn in the combustion chamber occurs in them. In contemporary Liquid-Fuel Rocket Engines owing to rational construction of the cap, physical non-burning of fuels is very insignificant, and the basic influence on the expansion process of combustion products in the nozzle is rendered by recombination of combustion products and transition of chemical energy into other forms of energy evoked by it.

The energy equation derived earlier in connection with the above should be modified in the appropriate way. In the general energy reserve it is necessary to also introduce chemical energy. For that enthalpy of combustion products H in equation (6.5) should be replaced by total enthalpy I and considered, as also earlier,

$$E = I + \frac{Aw^2}{2g} = \text{const}$$

or for two states corresponding to sections 1-1 and 2-2 (see Fig. 6.4)

$$I_0 + \frac{Aw_0^2}{2g} = I_a + \frac{Aw_a^2}{2g}.$$

Equation (6.8) for calculation of speed will have the form

$$w_a = \sqrt{w_0^2 + \frac{2g}{A}(I_0 - I_a)}$$

In particular, for the exhaust velocity from the rocket engine nozzle we get

$$w_a = \sqrt{\frac{2g}{A}(I_0 - I_a)}.$$

Total enthalpy of combustion products in the chamber $I_0 = I_{cp}$ is determined from calculation of the combustion process. In the absence of thermal losses

$$I_{\text{п.с}} = I_{\text{топл}}.$$

[пс = cp = combustion products; топл = fuel = fuel]

Total enthalpy of combustion products on nozzle section I_a is determined just as total enthalpy in the chamber, i.e., by temperature and composition of gases.

It is absolutely obvious that exhaust velocity, taking into account recombination and burning, will be large as compared to magnitude w_a, calculated with the assumption of invariability of composition of the gas mixture flowing along the nozzle. Thermal efficiency of the engine may be in these conditions defined by the formula

$$\eta_t = \frac{I_0 - I_a}{K_G},$$

where K_G is the weight caloricity of the fuel.

The phenomenon of recombination leads not only to change (increase) of exhaust velocity, but also to change of all other parameters of gas flow as compared to parameters of flow of gas of constant composition.

Flow parameters are changed because the expansion process recombining chemically active combustion products occurs not with the index of adiabat $k = c_p/c_V$, but with the average index of isentropy n_{is}. Thus, the equation of adiabatic process in reacting gas will have the form

$$\frac{p}{\rho^{n_{из}}} = \text{const.}$$

[из = is = isentropy]

As magnitude n_{is}, usually the averaged index of isentropy, constant for the whole expansion process is understood, although sometimes its magnitude in the process of gas flow through the nozzle can be changed significantly enough. The magnitude of index n_{is} is determined by the results of calculation of combustion and exhaust.

In connection with this all formulas describing the phenomenon of gas flow derived by us earlier remain just also in the case of flow along the nozzle of chemical active combustion products, if the index in those equations k is replaced by n_{is}.

Let us see how such replacement reflects on the basic parameters of gas flow.

In the first place speed of sound will be changed. In reacting gas it will be equal to

$$a=\sqrt{n_{is}\frac{p}{\rho}}=\sqrt{n_{is}gRT}.$$

This expression for speed of sound will be valid only if in a sound wave passing through the gas, the necessary chemcial reactions succeed in occurring, due to which the gas composition is changed in accordance with change of temperature and pressure in the sonic wave.

Measurements of pressure in the combustion chamber of a Liquid-Fuel Rocket Engine, which at constants G and S_{cr} depends on the index of the expansion process, imply that indeed the process of expansion occurs with an index close to the magnitude of n_{is}. This also convinces us that in the chamber nozzle the gas composition changes in accordance with equilibrium conditions with variables T and p. In particular, this implies that speed of sound indeed is equal to $\sqrt{n_{is}gRT}$; i.e., in a sonic wave chemical reactions necessary for supporting of equilibrium succeed in occurring.

Calculation of the exhaust process, taking into account recombination just as in calculation of combustion taking into account dissociation, is complicated enough. However, to get computed values of specific thrust close to real independence upon the dimensions of the engine chamber nozzle, it is necessary to use these complicated calculations; as a result they are widely applied during designing and analysis of rocket engine tests.

Form of Rocket Engine Nozzles

Liquid-propellant rocket engines known at present, the data on which is published by foreign press, are characterized by the following basic parameters.

Pressure in the combustion chamber is within the limits of 20-40 kg/cm^2. Increase of pressure in the chamber, other things equal, leads, as already was said, to an increase of specific thrust, to a decrease of the dimensions of the nozzle's critical section, and consequently, all its other dimensions, and also to a certain decrease of dimensions of the combustion chamber. Therefore, it is subsequently possible to expect an increase of pressure in the combustion chambers of rocket engines.

Relations of pressures p_0/p_a for contemporary engines are equal to 20-60. Thus, broadening of the nozzle S_a/S_{cr} is equal to 4-9. It is possible to expect a further increase in p_0/p_a both due to growth of pressure in the chamber and also at the expense of lowering the pressure on the nozzle section, which is profitable for engines of long-range rockets and certain types of aircraft engines.

The elementary theory of the supersonic nozzle, considered above, shows that the exhaust velocity of gases from the nozzle and the pressure on a nozzle section do not depend on its form. It is necessary only, that the nozzle in the beginning is narrowed and then becomes expanded. Change of profile of the nozzle, i.e., law of change of section area along the length, affects only the law of variation of parameters of gas flow along the length. Pressure and speed at exit are determined only by the relation of area of the exit section S_a to the area of the critical section S_{cr}.

The geometric form of the nozzle should be made in such a manner so that the nozzle does not have large losses of speed. At the same time its surface should be least. With an increase of surface, the weight of nozzle and quantity of heat transmitted into the liquid coolant increase.

Losses in the nozzle occur as a result of friction, molecular collisions, and eddy formations in the gas flow. Also, part of the kinetic energy of the directed motion becomes gas enthalpy unutilized for creation of tractive force. Let us consider in connection with the above, what shape the separate sections of the nozzle should take.

The entrance part up to the critical section should be as short as possible and also smooth to avoid collisions of gas against the walls, in order to decrease losses due to friction.

In the supercritical part, for decrease of nozzle surface, it is desirable to make it shorter and correspondingly to increase the angle of expansion.

However, it can happen that at a large angle of expansion the flow of gas, passing with great speed, will not succeed in being expanded and will not fill all of the nozzle section. Breaking away of flow from the walls will occur, and loss through eddy formation will sharply increase. Thus, the angle of expansion of the nozzle should be limited.

The allowed angle of expansion of nozzles in Liquid-Fuel Rocket Engines is sufficiently large. In the beginning of the divergent section of the nozzle it can be as high as 35-40° (Fig. 6.14). Here intense reactions of burning and recombinations, which are accompanied by liberation of heat, still occur. Due to this, the conditions of radial expansion of flow are simplified, and breaking away of the stream from the walls, even with such large angles of expansion, will not occur.

At the exit of the nozzle the angle of expansion should be decreased. Here the reactions of recombination and burning are significantly less intense, evolution of heat decreases, and the conditions of expansion of flow are hampered. Furthermore, at the nozzle exit the angle of expansion should decrease still more in order to lower the radial component of the flow speed (see Fig. 6.14) which does not give thrust. It is desirable that the total flow has a direction coinciding with the rocket axis.

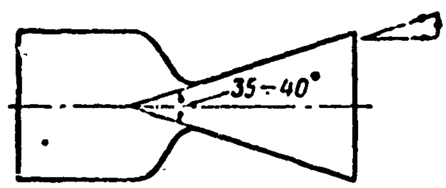


Fig. 6.14. Loss of kinetic energy due to creation of radial gas flows.

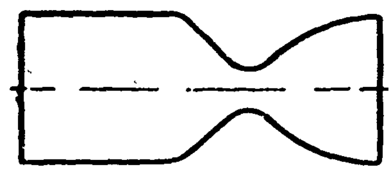


Fig. 6.15. Efficient form a nozzle with large broadening.

Through technological considerations it is more convenient to make the divergent section of the nozzle in the form of a cone. However, in this case the angle of opening of the nozzle must be comparatively small (20-30°) in avoidance of losses, and the resulting nozzle is long with a large area of lateral surface.

It is more profitable to consider the nozzle whose share is shown on Fig. 6.15. Transition from a large angle of expansion for the critical section to a small angle on this section is done along the arc circumference. The nozzle length and loss to radial flow are reduced to a minimum. Such a nozzle shape, however, does not exclude the possibility of appearance of local losses. In order to bring these losses to a minimum, it is necessary to make the nozzle shape correspond to the trajectory of gas particles passing near the wall. This trajectory is definite if the broadening of the nozzle and distribution of speeds in the critical section and on the nozzle section are given. Nozzles such as this are called gas-dynamically shaped nozzles.

Methods of shaping, known from aerodynamics, give the nozzle a very great length, and consequently, a great area of lateral surface. Therefore, such methods of shaping of nozzles in rocket engines are not applied.

4. Peculiarities of the Supersonic Nozzle. Characteristics of Rocket Engines

Peculiarities of the Supersonic Nozzle and the Conditions of its Work

The supersonic nozzle possesses a number of interesting peculiarities that influence the work of a rocket engine.

The first of such peculiarities is the independence of parameters of combustion products in the chamber and nozzle on external atmospheric conditions. This is explained by the fact that change of atmospheric conditions during supersonic exhaust cannot penetrate into the nozzle toward the flow, which is moving with a speed, faster than sound. Thus, with ascent of the rocket the pressure on the nozzle section (under constant pressure in the chamber) remains constant. On the other hand, if the engine works with a variable flow rate of fuel, then independently of atmospheric conditions after a pressure change in the chamber, the pressure on the nozzle section will be changed.

Speed on getting out of a nozzle with a given broadening will remain constant under any conditions of exhaust, unless there is a change in temperature and gas composition in the combustion chamber. Pressure in the combustion chamber influences exhaust velocity, but only by changing the degree of dissociation of combustion products.

Let us consider so-called efficient and inefficient conditions of the work of the nozzle.

Efficient conditions of the nozzle are understood to be such conditions of its work, when the pressure in the flow of gases on the nozzle section is equal to atmospheric. If pressure on the nozzle section is not equal to atmospheric, then it is said that the nozzle works under inefficient conditions.

Two inefficient conditions of work of the nozzle are distinguished: conditions of underexpansion and conditions of overexpansion.

Under conditions of underexpansion, the pressure in a gas flow on the nozzle section is greater than the pressure in the atmosphere. Underexpansion appears, for instance, during ascent of a rocket into altitudes, where the pressure in the atmosphere becomes less than the pressure on the nozzle section. Gas ensuing from the nozzle at excess pressure is expanded in the atmosphere and is mixed with the external medium. As a result, the speed of flow during expansion outside the nozzle is practically not increased due to formation of turbulence on the stream boundaries.

In case of overexpansion, pressure on the nozzle section is less than the external pressure. As a result, beyond the borders of the nozzle there occurs, naturally, a braking of flow and deceleration to subsonic. Transition to subsonic speed, as will be shown latter (see Chapter VII, Section 3), is accompanied by an intermittent change of pressure. Because of growth of the external pressure, jumps of pressure approach the nozzle section and, at sufficiently large excess pressure in the atmosphere, enter inside the nozzle.

In this case the normal operating conditions of nozzle are disturbed, since jolts inside the nozzle lead to breaking away of flow from the walls, appearance of powerful turbulence, and large losses of kinetic energy. According to certain data, formation of pressure jumps inside the nozzle occurs, if external pressure is 2.5-5.5 times greater than the pressure on the nozzle section.

Overexpansion in the nozzle appears during work of the engine at such altitudes and such conditions, when the pressure in the surrounding atmosphere is higher than the pressure on the nozzle section (for instance, with decrease of flow rate of fuel in the engine and corresponding lowering of chamber pressure).

The most interesting aspect in appraisal of operating conditions of the nozzle is the comparison of these conditions with the magnitude of specific thrust developed by the engine.

We will compare the three nozzles, shown on Fig. 6.16. The first nozzle is shortened, and pressure p_a on its section will be greater than the pressure of the surrounding medium p. The nozzle works under conditions of underexpansion. The second nozzle has large dimensions and works under efficient conditions (pressure p_a is equal to external pressure). Finally, the third nozzle has still larger dimensions, and pressure on nozzle section turns out to be less than the pressure of the surrounding medium. In this case conditions of overexpansion appear.

If one were to return to formula (1.6)

$$P = mw + S_a(p_a - p),$$

then immediately it is difficult to say, in which of the three cases we will obtain a large thrust at identical flow rate. In the first case the exhaust velocity $w = w_a$ will be smaller, but then the difference p_a-p has a positive value. In the third case there will be a high exhaust speed, but tne difference p_a-p becomes negative.

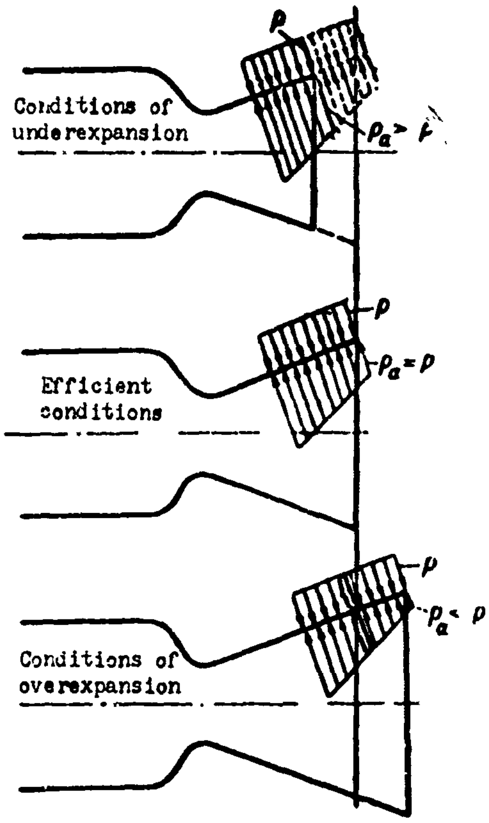


Fig. 6.16. Comparison operating conditions of a nozzle.

The given question, however, is resolved rather simply, if one were to remember that thrust force is a resultant of forces of pressure, distributed along the internal surface of the engine chamber and external surface of the rocket. On Fig. 6.16 the distributive law of pressures near the nozzle section is shown in the three considered cases. As we can see, an increase of nozzle length under underexpansion will cause an increase of the axial component forces of pressure. These forces are shown on the first figure by a dotted line. In conditions of overexpansion the atmospheric pressure on the final section of the nozzle predominates over pressure of exhaust gases, and the projection of the resultant force on the engine axis will be directed against the thrust force. Thus, it becomes evident that optimum operating conditions of the nozzle at constant flow rate of fuel are the efficient conditions. In other words, during work of the nozzle under efficient conditions, the engine will have maximum specific thrust.

Characteristics of Rocket Engines

The dependence of thrust on conditions of work of the engine is understood by the characteristics of rocket engines. The basic characteristics are high-altitude and throttle.

The high-altitude characteristic is a dependence of engine thrust on the altitude, at which it works, at a constant flow rate of fuel.

To obtain the high-altitude characteristic we will consider the formula for engine thrust (1.7)

$$P = P_0 + S_a(p_0 - p_h).$$

At a constant secondal flow rate of fuel G, the magnitude of the tractive force on the earth's surface P_0 also remains constant. Therefore, the only variable in the formula for thrust is pressure p_h. Total change (increase) of thrust of an engine in ascent from earth, where $p_0 = 1.033$ kg/cm^2, to such an altitude, where atmospheric pressure can be assumed to be equal to zero ($p_h = 0$), will be $1.033S_a$ kg, if the area of the nozzle's output section is expressed in cm^2. Thrust in these conditions (in vacuum) is

$$P_п = P_0 + S_a p_0.$$

[п = vac = vacuum]

High-altitude characteristics of engines have the form of curves depicted on Fig. 6.17.

It is interesting to clarify what thrust engines will give, having identical flow rates of fuel G, identical areas of critical nozzle section, and identical chamber pressures, but different areas of the nozzles's output sections S_a and, consequently, different pressures on sections p_a.

From the considerations given in the preceding division, it is obvious that an engine having a large efficiency altitude h (for which $p_a = p_h$), i.e., having a large output section S_a, will develop great thrust during work in upper layers of the atmosphere (curve B). An engine, having a lower efficiency altitude will have a great thrust in lower layers of the atmosphere (curve A).

In this is included the cause of the fact that many engines of long-range and zenith rockets, working at great heights a large part of the time, and also aircraft engines, are made so that the pressure on the nozzle section is less than atmospheric pressure on the surface of the earth (0.7-0.8 kg/cm^2). The nozzles of such engines are called high-altitude. The relative change of thrust during ascent depends on the height of the nozzle, and for contemporary engines it is 10-25%.

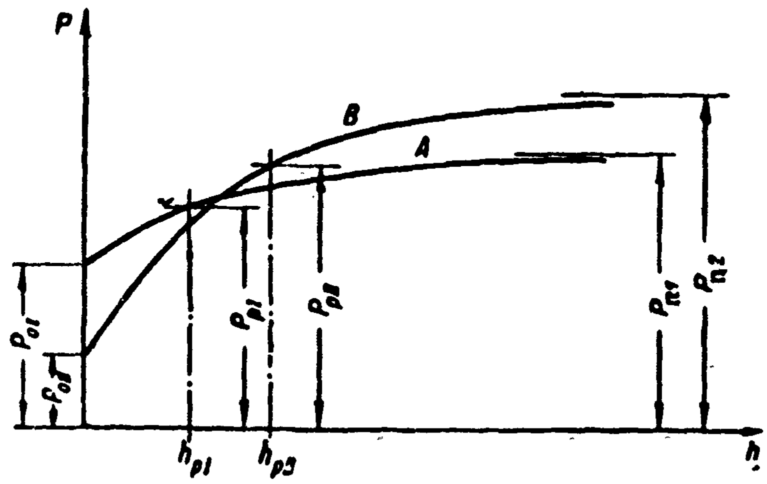


Fig. 6.17. High-altitude characteristic of a rocket engine.

The throttle characteristic expresses the dependence of engine thrust on change of flow rate of fuel at a constant altitude at which the engine works. During change of flow rate of fuel G and corresponding change of pressure in the combustion chamber of the engine p_0, in general both temperature and magnitude of the gas constant change; however, these changes are usually not considered in plotting of the throttle characteristic.

The equation of the throttle characteristic is easy to obtain from the equation of tractive force of the rocket engine if one considers that for a given engine the exhaust velocity w_a does not depend (at constants R and T_0 in the chamber) on flow rate G, and the pressure on the nozzle section decreases proportionally with fuel expenditure. The last affirmation follows from the fact that the relation of pressures p_a/p_0 for a nozzle of given dimensions, i.e., with the given relationship S_a/S_{cr}, is constant. In connection with this the pressure on the nozzle section can be expressed as a constant share from the chamber pressure,

$$p_a = \chi p_0.$$

The actual chamber pressure at constants T_0 and R is proportional to expenditure of fuel G; consequently,

$$p_a = \chi' G,$$

whence

$$p_a = p_{a\,\text{ном}} \frac{G}{G_{\text{ном}}},$$

[ном = nom = nominal]

where G_{nom} – nominal (efficient) flow rate for the given engine;

$p_{a\ nom}$ – pressure on nozzle section with this flow rate.

Considering the above remarks, expression (1.6) for tractive force of an engine

$$P = mw + S_a(p_a - p_h)$$

can be rewritten in the following form:

$$P = G\left(\frac{w_a}{g} + S_a \frac{p_{a\,\text{ном}}}{G_{\text{ном}}}\right) - S_a p_h. \tag{6.40}$$

The obtained equation of the throttle characteristic is an equation of a straight line, cutting off on the ordinate axis section $S_a p_h$ (Fig. 6.18). The slope of this straight line is determined, obviously, by velocity w_a and is increased with an increase in S_a.

In certain cases, using that circumstance that chamber pressure is proportional to expenditure, the throttle characteristic is derived not by expenditure, but by the magnitude of chamber pressure. This is especially convenient in derivation of experimental characteristics obtained during testing of the engine, since chamber pressure is significantly simpler to measure than flow rate of fuel.

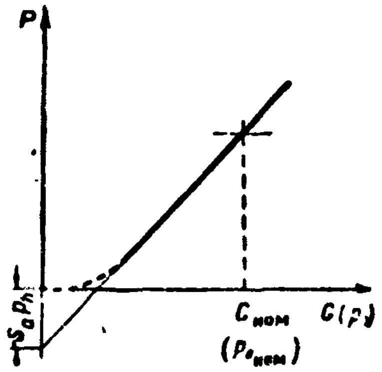


Fig. 6.18. Throttle characteristic of a rocket engine.

The rectilinear form of the throttle characteristic, determined by equation (6.40), is well confirmed by experiments with flow rates, greater than nominal and with such lowered flow rates, where over-expansion is not as great, so that pressure jumps could enter into the nozzle. Thus, the throttle characteristic has the form of a straight line to flow rates of 30-40% of the nominal. At lower flow rates relationship (6.40) becomes false, and the throttle characteristic is approximately as is shown on Fig. 6.18 by the dotted line. At small chamber pressures a significant increase of the degree of dissociation of combustion products and a corresponding decrease of temperature in the chamber also becomes prevalent, additionally effecting the form of the throttle engine performance.

5. Cooling of Liquid Rocket Engines

Heat Exchange in Rocket Engines

High temperature of combustion products and high speed of their movement through the chamber and especially along the nozzle leads to a very intense transmission of heat from gas to the walls of the rocket engine.

This seemingly contradicts what was said above because of processes occurring in the engine's nozzle. Earlier we considered expansion of gas adiabatic, i.e., occurring without heat exchange between gas and walls. Now we discuss intense transmission of heat to the walls. However, the fact is that the thermal energy transmitted to the walls and then into the cooling medium is a small share of the total enthalpy of gas and cannot essentially influence the character of flow. In absolute magnitude this share constitutes a very large quantity of heat, which without proper cooling cannot be absorbed by the wall without a burnt place and destruction.

Heat is transmitted from the gas to the wall by two ways: convection and radiation.

Convectional heat exchange plays a basic role in contemporary rocket engines. The essence of it consists in the fact that particles (molecules and atoms) of gas in their chaotic motion passage along the wall exchange thermal energy with it. Thus, in convectional heating exchange practically only the layer of gas flowing by the wall participates — the so-called boundary layer of gas.

Closest to the wall the flow of gas is laminar, and intense motions of small volumes of gas are absent in it, characteristic for turbulent motion. However, feeding of heat, and consequently also the temperature of the laminar layer are determined by turbulent motion of gas of the region of the boundary layer farther away from the wall. Therefore, the conditions of turbulent gas flow in the boundary layer also affect heat exchange between gases and the wall.

The quantity of heat transmitted to the wall depends on the number of collisions of molecules against the wall and on the difference of temperatures of the gas and wall. At equal temperatures on the average the gas obtains as much heat from wall of the engine as it returns to it.

The influence of the number of blows of gas molecules against the wall on heat exchange is considered by the coefficient of heat radiation α, which is the quantity of heat transmitted to a unit of surface of the wall in a unit of time at a difference of temperatures of the gas and wall of one degree. The number of blows of moving gas molecules against the wall is proportional to the number of molecules in the volume of gas passing past the wall in a unit of time, i.e., gas density ρ and speed of gas w.

Thus, the number of collisions, which means also the quantity of heat, transmitted to the wall is proportional to the product ρw. The latter magnitude is the mass specific flow rate, which is equal to the relationship of the mass secondal flow rate to the area of passage section,

$$\rho w = \frac{G}{gS}.$$

The quantity of heat transmitted by the gas to a unit of wall surface in a unit of time, i.e., the specific heat flow is

$$q = \alpha_{\text{г}} (T_{\text{газ}} - T_{\text{г.ст}}). \quad (6.41)$$

[$\alpha_{\text{г}} = \alpha_g$ = coefficient of heat transfer from gas to wall; газ = gas = = gas; г.ст = g.w = gas wall]

where $T_{g.w}$ – temperature of the wall surface in contact with gases;

T_{gas} – temperature of gas;

α_g – coefficient of heat radiation from the gas to wall.

Inasmuch as gas flow near the wall is braked, it has in all sections of the nozzle a temperature close to the temperature of braking, i.e., a temperature corresponding to zero speed of gas flow. Formula (6.22) shows that the braking temperature will be equal to the temperature in the chamber T_0, if the speed of gas in the combustion chamber is disregarded. The wall temperature is considerably less than the gas temperature. Due to this, a large difference of temperatures is obtained $T_{gas} - T_{g.w}$. On the other hand, owing to high pressures and high speeds, tha mass specific flow rate of gas is great, and the coefficient of heat radiation α_g in a Liquid-Fuel Rocket Engine turns out to be very high. All these circumstance also lead to the position where a significantly larger quantity of heat is discharged into the walls of the rocket engines than into the walls of any other thermal machine.

Along the length of the engine's nozzle both coefficient α_g, and the difference of temperatures $T_{gas} - T_{g.w}$ are changed. As a result, due to change of specific mass flow rate ρw along the length of the engine the coefficient of heat radiation α_g is changed sharply. The biggest values of ρw and α_g are attained in the critical section of the nozzle. In this section the magnitude of heat flow q also has a maximum (Fig. 6.19). In contemporary engines the magnitude of convectional specific heat flow in the critical section reaches $10 \cdot 10^6$ kcal/m^2 hr.

With an increase of temperature and pressure in the chamber, heat flow into the walls of the nozzle significantly increases.

To convectional thermal flow is added heat flow from radiation of gases. The magnitude of radiant heat flow is less significant than that of convection. The most intense radiation occurs, where the temperature of gases is great, i.e., in the combustion chamber. The magnitude of radiant heat flow here is equal to $(1.5 \text{ to } 2) \cdot 10^6$ kcal/m^2 hr. In the critical section in connection with lowering of gas temperature the magnitude of radiant flow is less.

Heat flow, obtained by the walls from within the chamber, should be transmitted to the surface of the wall touching the liquid coolant. As a result, the surface of the walls absorbing and returning heat have temperature connected among themselves by the equation of thermal conduction

$$T_{г.ст} - T_{ж.ст} = q\frac{\delta}{\lambda}, \qquad (6.42)$$

[ж.ст = l.w = liquid wall]

where $T_{l.w}$ — temperature of wall surface, bathed by liquid coolant;

δ — thickness of wall;

λ — coefficient of thermal conduction of wall material.

Magnitude δ/λ becomes the thermal resistance of the wall. The thicker the wall and the worse its material conducts heat, the greater the thermal resistance of the wall.

Heat flow should next be transmitted to the liquid coolant, having a temperature T_{liq}. The magnitude of heat flow is connected with the temperatures of the surface bathed by the cooler and the liquid by the equation

$$q = \alpha_{ж}(T_{ж.ст} - T_{ж}), \qquad (6.43)$$

[ж = liq = liquid]

where α_{liq} is the coefficient of heat radiation from the wall to the liquid coolant.

Excluding $T_{l.w}$ from equations (6.42) and (6.43), we get for the temperature of the wall surface, touched by the gas, the following expression:

$$T_{г.ст} = T_{ж} + q\left(\frac{1}{\alpha_{ж}} + \frac{\delta}{\lambda}\right). \qquad (6.44)$$

Change of temperature during transmission of heat from the gas to the wall and then to the liquid coolant is shown on Fig. 6.20.

Wall temperature $T_{g.w}$ does not have to exceed that upper limit at which the wall material significantly loses its durability properties. In this sense it is desirable that magnitude $T_{g.w}$ is as low as possible. To limit temperature $T_{g.w}$ is possible by increasing the coefficient of heat radiation to the liquid and decreasing the thermal resistance of wall.

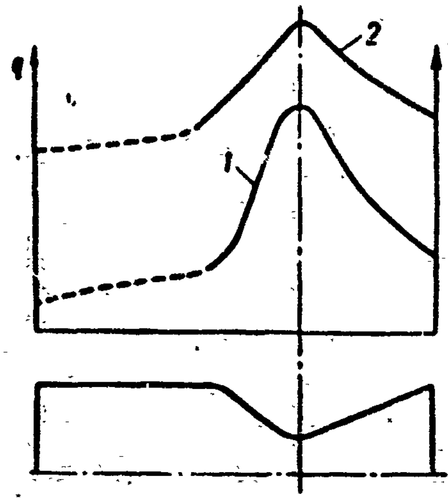


Fig. 6.19. Distribution of heat flow and temperature of a hot wall along the length of the engine. 1) heat flow through wall, 2) temperature of hot wall.

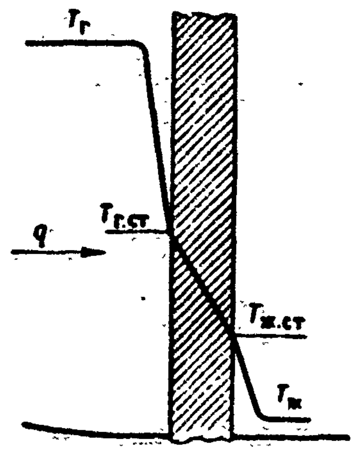


Fig. 6.20. Distribution of temperatures in the boundary layer of the gas, in the wall and in the liquid coolant.

The magnitude of the coefficient of heat radiation to liquid should be still greater because in the cooling system of the engine it is impossible to allow excessive increase of $T_{l.w}$ above the boiling point of cooling liquid. A small increase above the boiling point is allowed and even useful, since it increases the coefficient of heat radiation α_{liq}. Intensification of heat radiation is obtained due to formation of vapor bubbles on the wall surface. These bubbles are continually washed off by the flow of liquid coolant and, getting into the mass of colder liquid, are condensed in it, issuing the heat obtained from the wall.

The magnitude of the coefficient a heat radiation α_{liq}, just as the magnitude of α_g, depends on the flow of the mass of liquid bathing the wall. But since the density of the liquid is practically constant, the basic factor determining magnitude α_{liq}, is the speed of the cooling liquid. Intense cooling of the chamber walls and nozzle of a Liquid-Fuel Rocket Engine requires large values of coefficient α_{liq}. In connection with this, the speed of the liquid in the cooling system should be significant.

Organization of Engine Cooling

The problem of cooling the engines consists of the temperature of the walls during all work of the engine not exceeding a definite magnitude. In different types of rocket engines this problem is resolved in different ways.

In powder engines there is no liquid, which could cool the engine. Therefore, the time of work of the engine here is usually limited so that its walls do not succeed to be heated to such a temperature, that could cause a loss in durability.

For deceleration of heating of the walls and increase of the time of work of a powder engine, heat-insulating coatings, which are applied on the internal surface of engine's walls, are used. Analogously, wishing to prevent overheating of walls, the same is done in liquid-propellant rocket engines of brief action.

In a Liquid-Fuel Rocket Engine, however, reliable cooling is hampered by higher temperatures in the combustion chamber and great duration of the time of work of the engine. On the other hand, organization of cooling is facilitated by the fact that the wall can be protected by means of an abundant supply of one of the fuel components to the internal surface of the combustion chamber wall. It is true that such use of fuel components worsens the quality of combustion, and consequently, lowers the specific thrust of the engine. However, in engines requiring limiting simplicity, it is nevertheless possible.

Most frequently cooling of a Liquid-Fuel Rocket Engine is carried out by one of the fuel components, expelled through a cooling jacket. This method of cooling is more profitable. It does not require any additional liquids and distribution of the tanks in the rocket for their storage, i.e., it allows us to obtain a propulsion system of minimum weight. Furthermore, heat removed by liquid coolant will return into the combustion chamber and is not lost uselessly.

However, there are also certain difficulties in the organization of cooling of an engine by this method. They are determined mainly by the properties and mass flow rate of the components. It occurs that one of the components, for instance liquid oxygen, is unfit for use as a liquid coolant due to small difference of boiling point and initial temperature of the component. The second component may be insufficient for absorption of all heat transmitted through the wall of the engine during the time of its work.

Another difficulty consists in the fact that for achievement of the necessary magnitude of the coefficient of heat radiation α_{liq} it is necessary to create high speeds of the liquid coolant in the jacket space. For that the gap between the walls of the chamber should be very small. It is especially difficult to ensure the necessary speed of cooling liquid in engines of small dimensions. Thus, in the maneuver accelerator engine shown on Fig. 5.8, the height of the slot in the critical section must be made equal to only 0.6 mm. For that in the cooling jacket it is necessary to place a mechanically adapted insert and to guarantee preservation of the gap by spacer belts. In large engines the necessary magnitude of the gap increases, and technological difficulties partially drop.

It is necessary to say that high speeds in the cooling jacket are accompanied by hydraulic losses and force us to increase the feed pressure of the cooling component. This in turn complicates the work of the supply system and makes it heavier.

Returning to expression (6.44), we see that lowering of $T_{g.w}$ may be attained not only by an increase of speed of the cooling liquid and a corresponding increase of α_{liq}, but also by a decrease of thermal resistance of the wall δ/λ. It follows from this that the thickness δ should be as small as possible, and the coefficient of thermal conduction of the wall material λ should be as large as possible.

With an increase of thickness δ the temperature $T_{g.w}$ is increased so sharply that the durability of the wall is lowered. With an increase of heat flow in the wall the latter must be made much thinner. Thus, the special conditions of work of the wall material during a large heat flow consist of the fact that, it is impossible to increase the durability of the wall by increasing its thickness.

For decrease of gas temperature the internal surface of the wall can be prepared from a material with high thermal conduction, for instance, from copper or aluminum. It is necessary, however, to consider that heat-conducting materials, as a rule, are less heat-resistant.

The given circumstances, especially the necessity of creation of a thin internal wall, lead to great difficulties in construction of the engine chamber and the nozzle.

A significant force acts on the internal wall of the engine evoked by a difference of pressures of the cooling liquid in the jacket and gases in the chamber and nozzle. This difference of pressures is especially great at the end of the nozzle. Therefore, for creation of a sufficiently durable and stable internal wall, it is necessary to strengthen it some how. Let us take as an example the design of an engine, in which the internal jacket is made in the form of a spiral by copper tubes of small diameter, inside which passes the cooling liquid. Tubes of small diameter sustain difference of pressures of liquids and gases well. The internal pressure in the chamber is absorbed by a thick steel jacket protected from the action of high temperature by a belt of tubes.

The structure of a chamber is known, in which the walls are prepared from shaped tubes of right-angle section, located along the generators of the chamber and soldered together. The general durability of chamber is attained also using steel bands, girdling the chamber in many sections. Other structures of Liquid-Fuel Rocket Engine chambers are also possible. Thus, for instance, the chamber can be of double-walled construction, the internal and external wall of which are fastened together. Owing to installation of connections strong rigidity is attained of the gap, through which passes the cooling liquid, and also essentially increased is the rigidity of the inner shell, which may be filled by the thin-walled material. In the last case the number of fastenings should be sufficiently large. Connections fastening the walls can be longitudinal, spiral and point. Longitudinal connection is exemplified by external and internal walls which are united together on the generators by means of welding rigid longitudinal belts to both walls, and located between them. Spiral connection is formed through fastening of walls by helical belts. Channels for passage of liquid coolant in this case are also helical. Point connections are made by means of spot welding of shells through dents, stamped into the external shell.

Difficulties in construction of the engine, connected with the presence of large heat flow, force us to look for methods of decreasing the latter. The basic method depends on the fact that, in the layer of gases adjoining the wall, in the so-called boundary layer an excess quantity of fuel moves. Such cooling is called internal. Thus, for instance, in the engine of a long-range rocket the fuel in the boundary layer is fed by several rows of holes (see Fig. 5.10). As a result a curtain of fuel protects the wall. The action of this fuel curtain consists not only in the fact that it, being heated and being evaporated, takes away heat from the wall, but also and in that it lowers the temperature of that layer of gas which directly touches the walls of the engine. So that the boundary layer could decrease convectional heat flow in the wall of chamber, it should be greater than the boundary layer, in which processes are developed, determining transmission of heat into the wall. Protective action of the boundary layer overenriched with fuel, is maintained for a certain length. Gradually the boundary layer is washed out by the basic flow of combustion products, and the excess of fuel in it burns, when brought into contact with the oxidizer penetrating from the basic flow; the temperature in the boundary layer is increased, and its protective action decreases. Therefore, in large engines it is necessary to make several series of holes to feed excess fuel.

Introduction of fuel for internal cooling through radial holes is not expedient, since a large part of it passes into gas layers remote from the wall. A more rational method of feeding of fuel is, for instance, in the engine whose chamber is shown on Fig. 5.8. Here the fuel enters through small holes into a narrow gap between the cap and chamber and moves along the walls, not immediately entering into the basic mass of gas.

Distribution of peripheral atomizers of fuel on the flat cup, also serves the goal of internal cooling, creating for the walls a boundary layer of gas of lowered temperature at the expense of additional feeding of fuel.

Expenditure of fuel for creation of a boundary layer of lowered temperature, of course, lowers the specific thrust of the engine, since part of the fuel does not burn completely and removes with itself part of the chemical energy. Therefore, a combined system of cooling is frequently applied, by which through moderate internal cooling the heat flow into the wall decreases, but then the heat is absorbed by external cooling with comparatively small hydraulic losses in the cooling system.

A significant decrease of flow rate of fuel to internal cooling can be expected, if as material for the internal wall of the chamber a porous material is used — a so-called perspiring wall, passing evenly over the entire surface a very small quantity of liquid, which continuously enriches the boundary layer and lowers its temperature.

There are also proposals to organize cooling of the engine by water, bringing it not only to boiling, but completely evaporating it and obtaining superheated steam on exit. This steam may be used in a TNA turbine, and then condensed in a heat exchanger cooled by one of the components. Similar systems of cooling can be useful only in very big engines.

CHAPTER VII

FORCES AND MOMENTS ACTING ON A ROCKET IN FLIGHT

1. System of Forces Acting on a Rocket in Flight, and Differential Equations of Motion

Coordinates Determining the Position of a Rocket in Space

In solution of different problems connected with calculation of rocket trajectory, with stability of motion, and with thermal and durability calculations, it becomes necessary to introduce systems of reference of time and position of the rocket in space.

Reading of time, used in rocket technology, is the single factor independent of the peculiarities of the considered processes and is conducted from the moment of start of the rocket in seconds.

The position of a rocket in space is determined first of all by three coordinates x, y, z of its center of gravity in the so-called terrestrial (starting) system of coordinates. For the origin of this system the point of start of the rocket is taken. For long-range rockets for the x axis a straight line is used tangent to the arc of the great circle connecting the start with the target (Fig. 7.1). Axis y is directed upwards, and axis z — perpendicular to the first two axes.

We will introduce also the so-called connected, or mobile, system of coordinates. Its origin is placed in the center of gravity of the

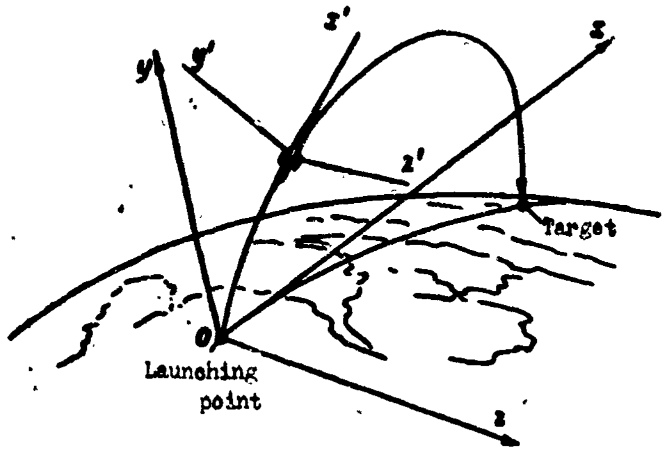


Fig. 7.1. Terrestrial and associated coordinate systems.

rocket, and axes x', y', and z' are connected with the characteristic elements of the rocket. Axis x' is directed along the axis of the rocket and called its longitudinal axis.

Axes y' and z' are disposed in a plane perpendicular to the longitudinal axis of rocket in such a way so that in the launching position the plane x'-y' coincides with plane x-y of the terrestrial system of coordinates, while axis z' has a direction similar to axis z. Axis y' is called the transverse, and axis z' – the lateral axis (Fig. 7.1 and 7.2).

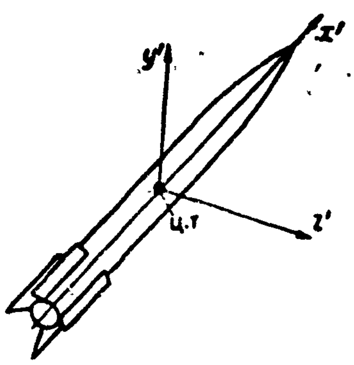


Fig. 7.2. Connected system of coordinates. [Ц.Т = c.g. = center of gravity]

For total determination of a rocket's position in space as a rigid body, in supplement to coordinates x, y, z we introduce three more angles, determining the mutual direction of axes of connected and terrestrial coordinate systems.

The angle between the rocket axis x' and plane x-z, i.e. the angle of inclination of the rocket axis with respect to the "starting" horizon, we will designate by φ. This angle is called the pitch angle of the rocket (Fig. 7.3).

The angle which is formed by the axis of the rocket with plane x-y will be designated by ψ. This angle is called the rove angle. It characterizes the deflection of the rocket axis from the vertical plane passing through the target.

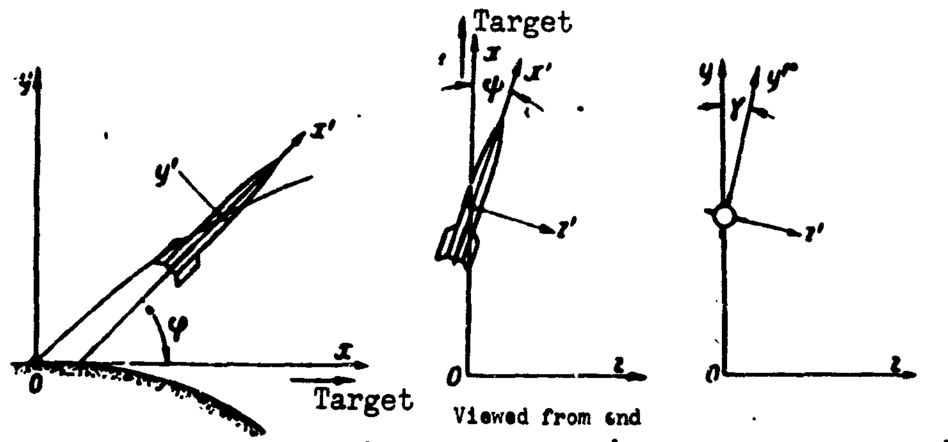


Fig. 7.3. Angles of pitch φ, rove ψ, and yaw γ.

We introduce, finally, one more angle determining the turn of the rocket body about its longitudinal axis — angle of yaw γ between axis y' and plane x-y.

Thus, three coordinates x, y, z and three angles φ, ψ, and γ (see Fig. 7.3) completely determine the position of a rocket in space.

The described coordinate systems are used in the first place in ballistic calculations and for solution of questions on stability of rocket motion. Along with this, for convenience of conducting aerodynamic calculations, for atmospheric sections of flight a so-called continuous system of coordinates x'', y'', z'' is also introduced. The origin of this system is placed in the center of gravity of the rocket. Axis x'' is directed along the vector of flight speed v, i.e. along the tangent to the trajectory; axis y'' has the direction of the principal normal of the trajectory, and axis z'' is directed along the binormal to the trajectory* (Fig. 7.4).

*The principal normal of the trajectory is determined by a line crossing the plane perpendicular to the tangent of the trajectory in a given point from the plane of the trajectory itself. The binormal is perpendicular to the plane passing through the tangent and principal normal.

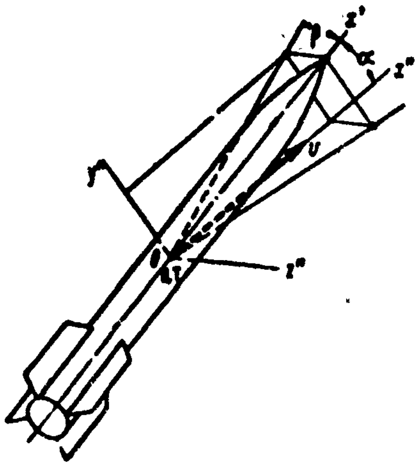


Fig. 7.4. Continuous coordinate system.

The connected and continuous coordinate systems during the rocket's flight, in general, do not coincide. Rocket axis x' forms with speed vector v angle α, called the angle of incidence, and with plane x''-y'' angle β, called the angle of slide (see Fig. 7.4). On a section of controlled flight in the atmosphere these angles are minute (angle α usually does not exceed 5-6°). For ballistic rockets angle β is much less than the angle of incidence α. Therefore, in ballistic calculation of flight trajectory the slide angle β is not taken into account and is considered only during the analysis of stability of motion.

Forces Acting on the Rocket

Let us consider the force system acting on a rocket in flight.

The basic forces determining rocket motion are tractive force, force of weight, and aerodynamic forces.

A rocket, as any body moving in air, experiences on the part of the latter a definite influence in the form of pressure distributed on the surface in a determined way. The resultant of pressure forces is called the total aerodynamic force.

During the analysis of the laws of rocket motion and aircraft in general, the total aerodynamic force is spread usually on the continuous axes x'', y'', and z'' on the components X, Y, Z.

Component X of the aerodynamic force on the tangent to the trajectory of motion of the body's center of gravity (or projection of total aerodynamic force in the direction of the speed vector) is called the drag force. This force is always directed opposite the motion.

Component Y of the total aerodynamic force on the continuous axis y'' is called <u>lift</u>.

Component Z is called the <u>drift force</u>.

On Fig. 7.5 are shown the tractive force P directed along the rocket axis, the force of gravity of the rocket $G = Mg$ and also components of aerodynamic force X and Y.

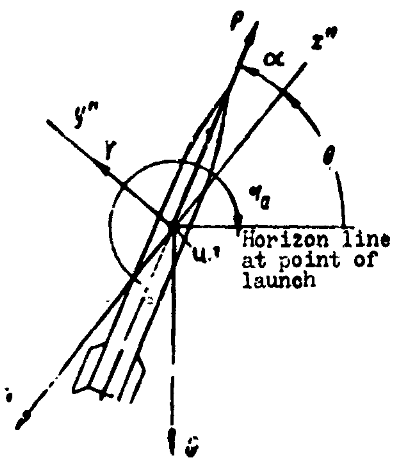


Fig. 7.5. Forces and moments acting on a rocket (with reduction of aerodynamic forces to the center of gravity).

The total aerodynamic force may be distributed also not on the continuous but on the connected axes x', y', z'. Such distribution is used, for instance, during the analysis of loads acting on elements of the structure.

Total aerodynamic force components corresponding with axes x', y', z' are designated by R, N, and T and are called axial, transverse, and lateral aerodynamic forces. On Fig. 7.6 axial R and transverse N forces are shown. From comparison of Fig. 7.5 and 7.6 it is easy to establish that in the absence of a slide angle (at $\beta = 0$)

$$R = X\cos\alpha - Y\sin\alpha,$$
$$N = X\sin\alpha + Y\cos\alpha,$$

and at small angles of incidence α

$$\left.\begin{aligned}R &\approx X - Y\alpha,\\ N &\approx X\alpha + Y.\end{aligned}\right\} \qquad (7.1)$$

Distributed on the rocket surface the system of aerodynamic forces according to laws of mechanics is reduced not only to the

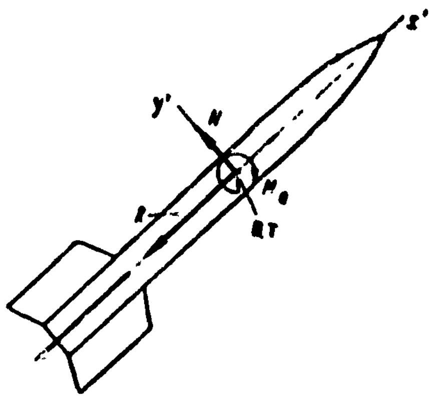


Fig. 7.6. Aerodynamic forces R and N in the connected coordinate system.

total aerodynamic force, but also to the moment, the magnitude of which depends on selection of the point of reduction of forces. Reducing the force system to the rocket's center of gravity, besides the mentioned forces X, Y, X (or R, N, T), we obtain a resultant aerodynamic moment, which in general may be analogous to the vector of total aerodynamic force distributed on the coordinate axes of system x'', y'', z'', or x', y', z'.

On Fig. 7.5 and 7.6 the components of this moment M_a, are shown acting in planes x''-y'' or x'-y' of the continuous and connected systems of coordinates.

Each of the moment components is considered usually as the sum of two moments — static and damping:

$$M_a = M_{ст} + M_{д}.$$

[ст = st = static, д = da = damping]

These moments will be discussed below in more detail.

If a rocket has controls (air or gas current steering mechanisms or turning engine chamber), then to the force system shown on Fig. 7.6 the force system from the controls should be added.

For every separate control it is convenient to reduce these forces to the axis of joint of its turning part in the form of component forces and moments in a connected coordinate system. The moment acting on the controls and calculated by means of reduction of forces to the axis of control rotation, is called the hinge moment.

The force system and hinge moments acting on all controls can be assigned to the point of crossing of their hinge axes with the rocket's longitudinal axis. We will consider that this point is located at a distance c from the rocket's center of gravity.

Thus, the force system on the controls for each of the coordinate planes of the connected system is reduced to two component forces and moments. Thus, in plane x'-y' one of the components X_{con} (Fig. 7.7) is directed along the rocket axis; the second Y_{con} is perpendicular to it.

The first of these forces X_{con} is called the loss of thrust on the controls. This component is not the control force, inasmuch as it does not create a moment about the rocket's center of gravity. The second force Y_{con} is the control force. It creates a control moment with respect to axis z'.

Moment M_h is the hinge moment.

In the case where controls are steering wheels, force X_{con} by analogy with aerodynamic forces is called the resistance of steering wheels, while force Y_{con} – the lift of the steering wheels.

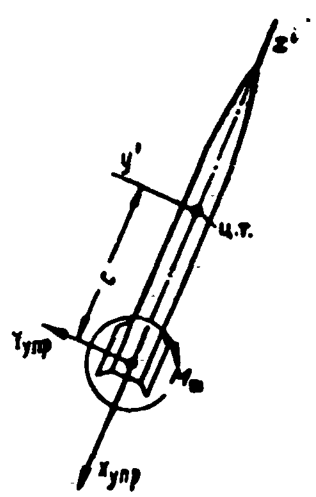


Fig. 7.7. Control forces. [ш = h = hinge, упр = con = control]

It is absolutely obvious that the analogous control force Z_{con} appears also in plane x'-z'. This force gives a control moment with respect to axis y'. Finally, in plane y'-z' the control forces on separate controls will ensure obtaining of a control moment with respect to the longitudinal rocket axis x'.

We will now turn to derivation of equations of rocket motion.

Differential Equations of Motion

Let us consider the simplest case of motion of a rocket in one plane.

On Fig. 7.8 is shown the total force system acting on a rocket during its flat motion. Here, planes $x'-y'$ and $x''-y''$, obviously, will coincide with the vertical plane x-y.

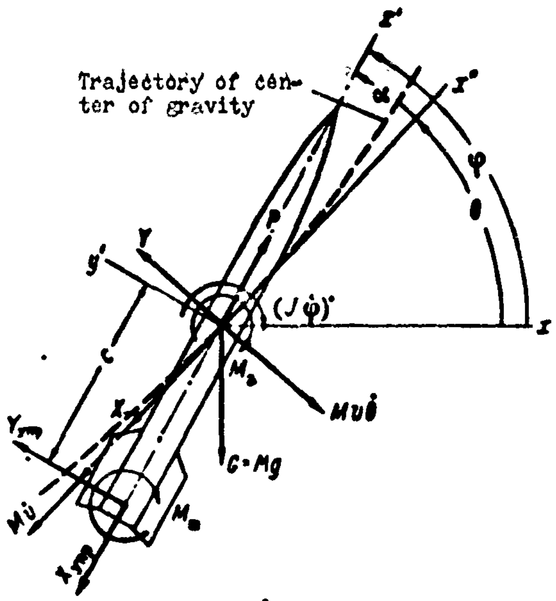


Fig. 7.8. Conclusion on motion equations.

In order to derive an equation of motion, we will use d'Alembert principle and introduce inertial forces.

Tangential acceleration of the rocket (acceleration on the tangent to the trajectory) will be

$$\frac{dv}{dt}=\dot{v}.$$

The corresponding inertial force is directed opposite acceleration and is equal to

$$M\dot{v}.$$

Normal acceleration caused by the curvature of the trajectory, is equal as it is known, to

$$\frac{v^2}{r},$$

where r is the radius of curvature of the trajectory.

But

$$\frac{1}{r}=\frac{d\theta}{ds}=\frac{d\theta}{dt}\frac{dt}{ds}=\frac{\dot{\theta}}{v},$$

where θ is the angle of inclination of the tangent to the trajectory counted off from the starting horizon (see Fig. 7.5).

Thus, normal acceleration, directed to the center of curvature, turns out to be equal to

$$v\dot{\theta}.$$

The inertial force directed in the opposite direction is equal to

$$Mv\dot{\theta}.$$

Forces $M\dot{v}$ and $Mv\dot{\theta}$ are shown on Fig. 7.8.

We will introduce, finally, inertial moment, equal to the derivative of the rocket's angular momentum by time and directed opposite of the angular acceleration $\ddot{\varphi}$. Here φ is the pitch angle, determining the position of the rocket's longitudinal axis (see Fig. 7.8):

$$\varphi = \theta + \alpha.$$

The inertial moment will be

$$[J(\dot{\theta}+\dot{\alpha})]^{\cdot} = (J\dot{\varphi})^{\cdot}.$$

The moment of inertia of the rocket J, relative to the central axis z', and perpendicular to the plane of the trajectory, is also a function of time. Therefore, moment $(J\dot{\varphi})^{\cdot}$ may be represented in the form of a sum

$$(J\dot{\varphi})^{\cdot} = J\ddot{\varphi} + \dot{J}\dot{\varphi}.$$

The first factor is inertial moment. The second factor is the result of change of the rocket's moment of inertia in time. This component, proportional to the angular velocity, may be included in the internal damping moment (see p. 349). In derivation of the motion equations we will combine it with moment M_a which was considered earlier. Subsequently during the analysis of damping moments we will return to this question again.

Projecting all forces applied to the rocket onto the tangent to the flight trajectory we get

$$M\dot{v}+X+(P-X_{ynp})\cos\alpha+Mg\sin\theta+Y_{ynp}\sin\alpha=0.$$

In view of the smallness of angle α, the value of $\cos\alpha \approx 1$. Disregarding $Y_{con}\sin\alpha$, we get

$$\dot{v}=\frac{P-X_{ynp}-X}{M}-g\sin\theta. \tag{7.2}$$

Projecting all forces onto the normal to the flight trajectory, we get

$$Y+Y_{ynp}\cos\alpha-Mv\dot{\theta}+(P-X_{ynp})\sin\alpha-Mg\cos\theta=0.$$

At small values of α

$$\dot{\theta}=\frac{1}{Mv}[(P-X_{ynp})\alpha+Y+Y_{ynp}]-\frac{g}{v}\cos\theta. \tag{7.3}$$

In these equations magnitude g should be understood as the acceleration of terrestrial gravitation on the height of flight.

Finally, taking the sum of moments about the center of gravity, we get

$$J\ddot{\varphi}+M_{a}+Y_{ynp}c+M_{m}=0. \tag{7.4}$$

The system of differential equations (7.2), (7.3), and (7.4) describes motion of the rocket only in one plane and does not consider the possibility of motion along the trajectory. Nonetheless, in a number of cases it is possible to use these equations freely for determination of parameters of the rocket trajectory, especially ballistic.

In those cases, when the flight trajectory is a space curve, as, for instance, for a zenith guided rocket pursuing a maneuvered aircraft, this system of equations is significantly complicated. In derivation of this system it is necessary to consider forces and moments on three coordinate axes and additionally to introduce into consideration angular shifts of the rocket through rove and bank. As a result, instead of three equations, six would be obtained.

It is important to note that for a guided rocket, equations giving the space location of the rocket during flight also have to be added to the differential equations of motion. The form of these equations depends on the methods of guiding and control in the rocket.

Thus, for instance, for ballistic long-range rockets an equation of the program of change of pitch angle

$$\varphi_{\text{пр}} = f(t),$$

[пр = pro = program]

is introduced where f(t) is a given function of time.

This condition of control is determined by the fact that the rocket is oriented in a determined way in the section of governed flight relatively constant in the direction of a coordinate system of gyroscopes.

These types of equations are called <u>program equations</u>.

Finally, in order to complete the presentation on the motion of a guided rocket, it is necessary to consider, furthermore, so-called <u>equations of control</u>, giving the magnitude of control forces (or turning angles of the controls) depending upon the rocket's space orientation and angular deflections of it from the assigned position. The form and numerical values of parameters of control equations depend on the fundamental scheme of the control system and on parameters of its adjustment.

We will discuss all these questions more specifically in Chapters VIII and IX.

In order to integrate the equation of motion, it is necessary to know how the magnitudes entering into these equations change in time and, in general, on what they depend.

It is absolutely clear that aerodynamic forces for a given rocket depend on speed, altitude of flight and angular orientation of the rocket in space. The tractive force of the engine does not remain constant; it changes as if because the external atmospheric pressure, entering in the expression of thrust, is changed on the flight trajectory. The mass of the rocket decreases in time in accordance with expenditure of fuel. Finally, even acceleration due to gravity g, if we consider large altitudes, should be considered a variable.

All these questions we will also consider below. We will start with properties of the atmosphere.

2. Terrestrial Atmosphere and Its Properties

Let us consider the properties of atmosphere, since the magnitudes of aerodynamic forces acting on a rocket in flight depend on these properties.

The basic parameter of the atmosphere affecting aerodynamic forces is air density ρ. Its temperature T also renders a certain influence, inasmuch as with a change of temperature, as we already know, the speed of sound

$$a=\sqrt{kgRT},$$

is changed, and depending upon the relation of flight speed to speed of sound (on the Mach number), as will be shown below, the character of streamlining is changed and as a result the magnitude of aerodynamic forces.

It is necessary also to consider that the speed of sound depends on change of chemical composition of the atmosphere with altitude. This will affect gas constant R and index of adiabat k. Change of chemical composition of the atmosphere is observed, however, only at very great altitudes.

Parameters of state of the air, especially in the lower, and also in significant measure in the upper layers of atmosphere, are changed depending upon the time of day, time of year, upon latitude of the site and, finally, on the general meteorological situation. However, all parameters oscillate near certain average values determined on the basis of results of atmospheric observation for many years.

The mean values of the parameters of the atmosphere themselves essentially depend on altitude. Thus, for instance, air pressure p due to increase of altitude should decrease, inasmuch as the magnitude of pressure is determined by the weight of air layers located higher. The character of pressure change with respect to altitude may be determined analytically. For this we will derive an equilibrium equation of unit column with base area dF and altitude dh separated from the atmosphere (Fig. 7.9).

Pressure p acts on the bottom of the column; from above in conformity with an increase of altitude dh, the pressure is p + dp. Let us assume that γ is the specific gravity of air at height h. Then the equilibrium condition will be the following:

$$dp\,dF+\gamma\,dh\,dF=0,$$

whence

$$\frac{dp}{dh}=-\gamma$$

or

$$\frac{dp}{dh}=-g\rho. \tag{7.5}$$

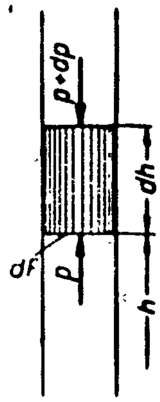


Fig. 7.9. Derivation of the dependence of air pressure on altitude.

Derivative dp/dh is negative, signifying a drop of pressure due to increase of altitude. It is not difficult to show that in the absence of absorption of heat from without, the temperature will also drop.

With a shift of any air mass upwards expansion and corresponding lowering of gas temperature occurs. Conversely, air travelling downwards is compressed, and its temperature is increased. Thus, a temperature equilibrium is established in the atmosphere, as a result of which the underlying layers will have a higher temperature than those above. Naturally. this affirmation is true only to the extent that it is possible to consider that heat exchange between air layers by means of radiation is small.

Temperature distribution by altitude can be considered dependent on that thermodynamic process which corresponds to the mentioned expansion and compression of gas during slow vertical mixing with preservation of temperature equilibrium. Let us assume that this expansion occurs on a polytrope with index n not depending on altitude h:

$$\frac{p}{p_0}=\left(\frac{\rho}{\rho_0}\right)^n. \tag{7.6}$$

where p_0 and ρ_0 are the pressure and air density at the earth's surface.

Using this law, we have the possibility of establishing a law of change of pressure, density and temperature of the air by height.

With the help of expression (7.6), we find density ρ, and. placing it in the equilibrium equation (7.5), we get

$$\frac{dp}{dh}=-g\rho_0\left(\frac{p}{p_0}\right)^{\frac{1}{n}}.$$

We divide the variables

$$p^{-\frac{1}{n}}\,dp = -g\rho_0 p_0^{-\frac{1}{n}}\,dh,$$

whence after integration we find

$$\frac{n}{n-1}p^{\frac{n-1}{n}} = -g\rho_0 p_0^{-\frac{1}{n}}h + C.$$

Constant C is determined from the conditions: at h = 0 pressure $p = p_0$. Consequently,

$$C = \frac{n}{n-1}p_0^{\frac{n-1}{n}}$$

and

$$p^{\frac{n-1}{n}} = p_0^{\frac{n-1}{n}} - g\rho_0 p_0^{-\frac{1}{n}}\frac{n-1}{n}h,$$

or

$$p = p_0\left(1 - g\frac{\rho_0}{p_0}\frac{n-1}{n}h\right)^{\frac{n}{n-1}}.$$

Since

$$\frac{p_0}{\rho_0} = gRT_0,$$

then finally the dependence of pressure on altitude we get in the form

$$p = p_0\left(1 - \frac{n-1}{nRT_0}h\right)^{\frac{n}{n-1}}. \qquad (7.7)$$

Density will be related to altitude by the relationship

$$\rho = \rho_0\left(1 - \frac{n-1}{nRT_0}h\right)^{\frac{1}{n-1}}. \qquad (7.8)$$

In accordance with the assumed polytropic process

$$T = T_0\left(\frac{p}{p_0}\right)^{\frac{n-1}{n}}$$

or, taking into account (7.7)

$$T = T_0 - \frac{n-1}{nR} h. \qquad (7.9)$$

Thus, temperature drops with altitude by linear law. The gradient of temperature drop is equal to

$$\frac{n-1}{nR}.$$

If one were to take n = k = 1.4 for air, i.e. to assume that in a state of temperature equilibrium, expansion and compression of air in vertical mixing occurs adiabatically, then we get

$$\frac{n-1}{nR} = \frac{1,4-1}{1,4 \cdot 29,27} = 0,0098 \text{ degree/meter},$$

i.e. temperature drop of air is approximately 1° for each 100 meters altitude. In fact, temperature drops in lower layers of atmosphere on the average 0.65° per 100 meters altitude, which corresponds to n = 1.23 and heat absorption during expansion.

Magnitude n, however, is changed with height; therefore dependences (7.7)-(7.9) should be considered approximate.

Experimental investigations of the atmosphere show that up to an altitude of h = 11 to 12 kilometers the derived laws of variation of air parameters by altitude will agree with the observed values sufficiently well. At high altitudes sharp deflections from the derived laws take place, and further regularities connecting the change of atmospheric properties with altitude are not open now to theoretical analysis; they are studied experimentally, primarily with the help of rockets and artificial earth satellites. Inasmuch as rocket investigations began relatively recently, in data on atmosphere properties there are quite a lot of contradictions, especially for altitudes over 100 km [kilometers].

At present it is possible to derive the following approxim[illegible] model of the atmosphere.

The atmosphere spreads to altitudes of 2000-3000 km (earlier it was considered that the border of the atmosphere is located below 1000 km), where it gradually passes into interplanetary gas.

The lowest layers of atmosphere, to 11 km, where the equations derived above are valid, is called the troposphere. Layers located higher than 11 km, are called stratosphere. Layers of atmosphere starting from 90 km are called the ionosphere, while the highest layers (over 1000 km) are the exosphere.

The chemical composition of the atmosphere (i.e. relationship between content of molecular nitrogen and oxygen) to a height of 90 km remains constant. At great heights the air becomes strongly ionized, as a result of which it is called the ionosphere. From a height of 90 km noticeable dissociation of oxygen into atoms begins, while from altitudes of 220 km nitrogen also starts to disintegrate into atoms. Simultaneously nitrogen oxide (NO) appears in the composition of the atmosphere. All these gases are greatly ionized. Also, ionized molecules and atoms in the upper layers of atmosphere can contain a noticeable quantity of free electrons.

The chemical composition of the atmosphere, and also the types and quantity of charged particles contain in it, along with other causes strongly affect atmospheric temperature.

Starting from an altitude of 11 km, the air temperature remains approximately constant and on the average equal to -56°C. Then from an altitude of 30 km an increase of temperature is observed to the maximum corresponding to altitudes near 50 km. Further starts a new lowering of temperature, and at a height of 80-100 km the temperature again attains a minimum. At great heights the temperature constantly increases. Thus, the temperature with respect to altitude is changed approximately as is shown on Fig. 7.10a.

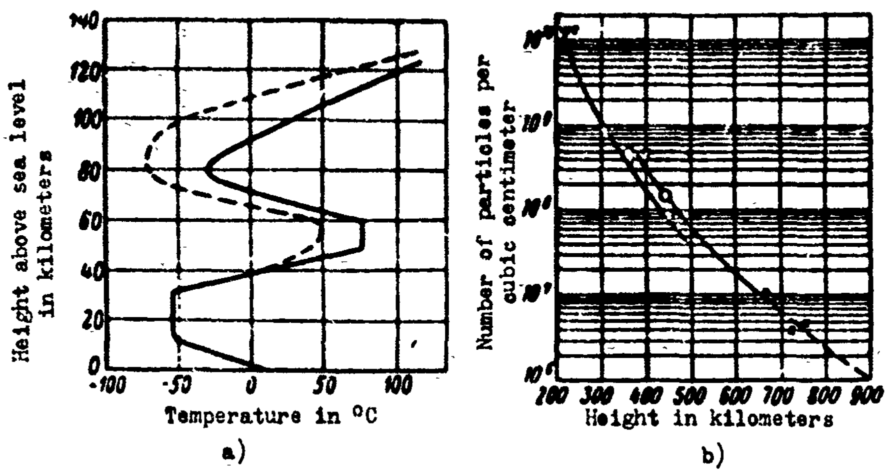


Fig. 7.10. Data on temperature and density of high layers of atmosphere. a – distribution of temperature by altitude in the terrestrial atmosphere. Dotted curve on the left – distribution of temperature obtained by meteorological rockets (March, central latitude of northern hemisphere). Solid curve – standard distribution of temperatures. b – curves of density change of neutral particles with altitude, obtained by different methods. ● – results of determination of density by study of braking of Soviet satellites. O – result of determination of density by diffusion of a sodium cloud. ★ – data, given in international literature by the study of braking of Soviet and American satellites. Solid lines correspond to results obtained with the help of manometers fixed on the third Soviet satellite and by radio signals of the first Soviet satellite.

The first increase of temperature at altitudes of 30-50 km, is explained by the fact that in layers of air located at these altitudes there is a small quantity of ozone, which absorbs short wave (ultra-violet) solar radiation very intensely. Further lowering of temperature is possible to explain by the same causes as in the first low layers of atmosphere. Finally, temperature increase in the highest layers, starting from altitudes of 80-100 km, is connected with bombardment of the terrestrial atmosphere by cosmic particles and with solar radiation. Therefore, the temperature of these layers of atmosphere greatly oscillates during the day and is noticeably changed depending upon the time of year and latitude of the site. Recent measurements of temperature with help of satellites showed that at an altitude of 300 km the temperature is 800-1000° absolute and then increases to 2000-3000° absolute.

It does not follow, however, to consider that such a high temperature can in any way evoke complications during mastering of this section of the atmosphere. The temperature of an aircraft, being at these altitudes for a prolonged time, will remain essentially less than the temperature of the surrounding medium. In view of extraordinary rarefaction of gas, heat transfer from air to the apparatus will be very insignificant, and the balance between heat transmitted by the gas, and heat lost by the body through radiation, will be established at a comparatively low temperature of the aircraft. In this sense temperatures appearing as a result of high speeds of a rocket in the atmosphere present danger incomparably great. (We will specially discuss this question later.)

Numerical data characterizing the change of average parameters of air by altitude are given in Table 7.1, which is of primary importance for ballistic calculations.

In Table 7.1 data of the so-called International Standard Atmosphere (ISA) is used up to an altitude of 20 km for average parameters of air, i.e. under the conditions of the introduced atmosphere, the parameters of which are considered independent of the time of year and day and of the latitude of the site. The International Standard Atmosphere is determined proceeding from the following assumptions:

1. Air is an ideal gas with gas constant $R = 29.27$ kg.m/kg.deg.

Table 7.1. Value of Parameters of Air Depending Upon Altitude Above Sea Level

h м	$\frac{p}{p_0}$	$\frac{\rho}{\rho_0}$	$\sqrt{\frac{T_0}{T}}$	h м	$\frac{p}{p_0}$	$\frac{\rho}{\rho_0}$	$\sqrt{\frac{T_0}{T}}$
0	1,000	1,000	1,000	6600	0,423	0,487	
200	0,976	0,980		6800	0,412	0,476	
400	0,953	0,960		7000	0,400	0,465	1,078
600	0,930	0,940		7200	0,389	0,454	
800	0,907	0,921		7400	0,378	0,443	
1000	0,885	0,902	1,010	7600	0,367	0,433	
1200	0,863	0,884		7800	0,356	0,422	
1400	0,842	0,866		8000	0,346	0,412	1,091
1600	0,821	0,848		8200	0,336	0,402	
1800	0,800	0,831		8400	0,327	0,393	
2000	0,780	0,813	1,021	8600	0,317	0,383	
2200	0,761	0,796		8800	0,308	0,374	
2400	0,742	0,779		9000	0,299	0,365	1,104
2600	0,723	0,763		9200	0,291	0,356	
2800	0,705	0,747		9400	0,282	0,347	
3000	0,687	0,731	1,032	9600	0,274	0,339	
3200	0,670	0,715		9800	0,266	0,330	
3400	0,653	0,700		10000	0,258	0,323	1,118
3600	0,636	0,685		10200	0,250	0,314	
3800	0,619	0,670		10400	0,243	0,306	
4000	0,603	0,655	1,043	10600	0,235	0,298	
4200	0,587	0,641		10800	0,228	0,290	
4400	0,572	0,627		11000	0,221	0,282	1,133
4600	0,557	0,613		11200	0,214	0,274	
4800	0,542	0,600		11400	0,207	0,266	
5000	0,527	0,586	1,054	11600	0,201	0,258	
5200	0,513	0,573		11800	0,195	0,251	
5400	0,499	0,560		12000	0,189	0,243	1,133
5600	0,486	0,548		12200	0,183	0,236	
5800	0,473	0,535		12400	0,178	0,229	
6000	0,460	0,523	1,066	12600	0,172	0,222	
6200	0,448	0,511		12800	0,167	0,215	
6400	0,435	0,499		13000	0,162	0,208	1,133

Table 7.1. (continuation)

h, м	$\frac{p}{p_0}$	$\frac{\rho}{\rho_0}$	$\sqrt{\frac{T_0}{T}}$	h, м	$\frac{p}{p_0}$	$\frac{\rho}{\rho_0}$	$\sqrt{\frac{T_0}{T}}$
13200	0,157	0,202		28000	$0,159 \cdot 10^{-1}$	$0,198 \cdot 10^{-1}$	1,114
13400	0,152	0,195		29000	0,137	0,169	1,112
13600	0,148	0,189		30000	0,118	0,145	1,109
13800	0,143	0,183		31000	0,102	0,125	1,107
14000	0,139	0,177	1,133	32000	$0,876 \cdot 10^{-2}$	0,107	1,105
14200	0,135	0,172		33000	0,756	$0,919 \cdot 10^{-2}$	1,102
14400	0,130	0,167		34000	0,652	0,790	1,100
14600	0,126	0,162		35000	0,563	0,679	1,098
14800	0,122	0,157		36000	0,487	0,573	1,084
15000	0,118	0,152	1,133	37000	0,423	0,485	1,071
15200	0,114	0,148		38000	0,369	0,413	1,058
15400	0,111	0,143		39000	0,322	0,352	1,046
15600	0,107	0,139		40000	0,283	0,302	1,031
15800	0,104	0,134		41000	0,249	0,260	1,022
16000	0,101	0,130	1,133	42000	0,219	0,224	1,011
16200	$0,980 \cdot 10^{-1}$	0,126		43000	0,194	0,194	1,000
16400	0,950	0,122		44000	0,172	0,168	0,999
16600	0,923	0,118		45000	0,153	0,147	0,979
16800	0,895	0,115		46000	0,136	0,128	0,969
17000	0,867	0,111	1,133	47000	0,122	0,112	0,960
17200	0,841	0,108		48000	0,109	$0,984 \cdot 10^{-3}$	0,951
17400	0,815	0,105		49000	$0,977 \cdot 10^{-3}$	0,866	0,942
17600	0,790	0,101		50000	0,878	0,764	0,933
17800	0,766	$0,980 \cdot 10^{-1}$		55000	0,508	0,473	0,963
18000	0,742	0,952	1,133	60000	0,284	0,282	0,997
19000	0,635	0,814	1,133	65000	0,152	0,163	1,034
20000	0,543	0,697	1,133	70000	$0,774 \cdot 10^{-4}$	$0,895 \cdot 10^{-4}$	1,075
21000	0,465	0,594	1,130	75000	0,372	0,468	1,122
22000	0,398	0,507	1,128	80000	0,167	0,229	1,176
23000	0,341	0,432	1,125	85000	$0,721 \cdot 10^{-5}$	$0,990 \cdot 10^{-5}$	1,176
24000	0,299	0,369	1,123	90000	0,310	0,429	1,176
25000	0,251	0,316	1,121	95000	0,134	0,185	1,176
26000	0,216	0,270	1,118	100000	$0,580 \cdot 10^{-6}$	$0,600 \cdot 10^{-6}$	1,176
27000	0,185	0,231	1,116				

2. Parameters of air at sea level:

pressure p_0 = 760 mm Hg;

density ρ_0 = 0.125 $kgsec^2/m^4$;

temperature T_0 = 288° absolute.

3. The temperature gradient in the troposphere is constant and is equal to 0.0065 degrees/meter.

4. The temperature in the lower layers of the stratosphere is constant and equal to -56.5°C.

As a result, for the troposphere (i.e. within limits of height up to 11,000 m) the laws of change of pressure, density and temperature by height according to expressions (7.7), (7.8) and (7.9) have the form

$$p=p_0\left(1-\frac{h}{44\,300}\right)^{5,256};$$

$$\rho=\rho_0\left(1-\frac{h}{44\,300}\right)^{4,256};$$

$$T=T_0-0{,}0065h.$$

Values of atmospheric parameters for altitudes greater than 20 km are obtained by means of experimental measurements with subsequent theoretical generalization of the data obtained.

Data on density of still higher layers of atmosphere before launching of Soviet earth satellites were for altitudes 150-250 km extremely contradictory, while the density of the atmosphere higher than 300-500 km was actually unknown.

Measuring change of time of rotation of a satellite around Earth due to its braking by the atmosphere, it is possible to determine sufficiently accurately the perigee of orbit magnitude, proportional to the density of the atmosphere. On the third artificial earth satellite furthermore for the first time, special manometers of ionic and magnetic types were installed with help of which density in the region of altitudes 225-500 km was measured.

Data on these and certain other measurements of density of the atmosphere at great heights are shown in Fig. 7.10b.

Parameters of high layers of the atmosphere (to 300 km), calculated on the basis of the model of the atmosphere considered earlier, are given in Table 7.2.

Table 7.2.

Altitude in km	Temperature, deg absolute	Pressure, dynes/cm^2	Density, g/cm^3	Altitude in km	Temperature, deg absolute	Pressure, dynes/cm^2	Density, g/cm^3
100	237	$5{,}69\cdot10^{-1}$	$8{,}29\cdot10^{-10}$	190	601	$5{,}72\cdot10^{-4}$	$2{,}74\cdot10^{-13}$
110	267	$1{,}58\cdot10^{-1}$	$1{,}97\cdot10^{-10}$	200	647	$3{,}73\cdot10^{-4}$	$1{,}66\cdot10^{-13}$
120	301	$5{,}32\cdot10^{-2}$	$5{,}61\cdot10^{-11}$	220	732	$1{,}73\cdot10^{-4}$	$6{,}82\cdot10^{-14}$
130	340	$2{,}13\cdot10^{-2}$	$1{,}90\cdot10^{-11}$	240	798	$8{,}74\cdot10^{-5}$	$3{,}11\cdot10^{-14}$
140	380	$9{,}72\cdot10^{-3}$	$7{,}57\cdot10^{-12}$	250	827	$6{,}38\cdot10^{-5}$	$2{,}15\cdot10^{-14}$
150	418	$4{,}88\cdot10^{-3}$	$3{,}40\cdot10^{-12}$	260	853	$4{,}74\cdot10^{-5}$	$1{,}52\cdot10^{-14}$
160	461	$2{,}63\cdot10^{-3}$	$1{,}65\cdot10^{-12}$	280	887	$2{,}74\cdot10^{-6}$	$7{,}93\cdot10^{-15}$
170	505	$1{,}51\cdot10^{-3}$	$8{,}61\cdot10^{-13}$	300	901	$1{,}68\cdot10^{-5}$	$4{,}42\cdot10^{-15}$
180	553	$9{,}08\cdot10^{-4}$	$4{,}73\cdot10^{-13}$				

During calculations of stability of rocket flight, and also for determination of lateral loads acting on the body of the rocket in flight, it is necessary to consider the peculiarities of wind distribution in the atmosphere. Depending upon the probable magnitudes of change of air flow speeds with altitude, the possible lateral brief aerodynamic perturbational forces are determined.

Nominal wind speeds for the earth's surface are determined on the basis of statistical treatment of results of meteorological observations in the region of rocket launching for more or less a prolonged interval of time.

In the upper layers of atmosphere, winds differ by high speeds and comparative constancy. The basic perturbation factors in formation of stratospheric winds are semidiurnal ebb-flow motions under the action of the Sun and Moon, and the heating of air for a day with subsequent cooling at night.

Tentative distribution curves of wind speeds (eastern or western) by altitude for the central latitude of the northern hemisphere are represented on Fig. 7.11. From these curves it is clear that a speed of 100 m/sec for high altitudes can be considered common. However, the power influence of such currents on the body of the rocket is significantly lowered due to great rarefaction of the atmosphere.

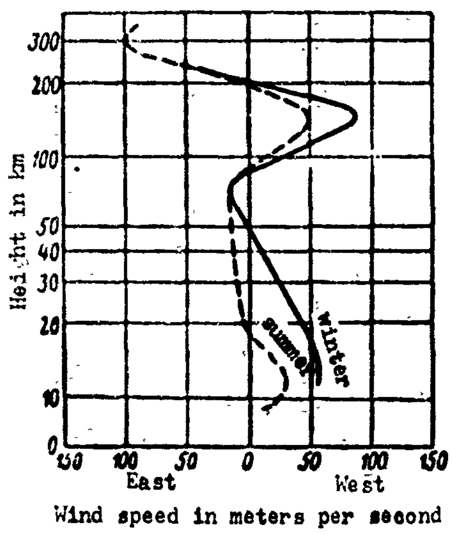


Fig. 7.11. Distributive law of predominant winds at great heights.

3. Aerodynamic Forces

Coefficient of Aerodynamic Forces

We will start the consideration of aerodynamic forces with the drag force X, inasmuch as during calculation of flight trajectory of rockets the role of this force as compared to other aerodynamic forces is the most essential.

The first and most natural attempt of determining drag was an attempt to express it through the magnitude of impact pressure.

If one considers reversed motion and incompressible gas, then for simple bodies having a form close to that of a plate, the magnitude of drag force X, one would think, can be determined in the following way.

We will designate by v the speed of undisturbed air flow and use the Bernoulli equation [see formula (6.11)] for a stream of incompressible gas in section 1-1, sufficiently remote from the body ($v_1 = v$) and in section 2-2 by the front surface of the body (Fig. 7.12)

$$\frac{\rho v_1^2}{2}+p_1=\frac{\rho v_2^2}{2}+p_2.$$

But $v_2 = 0$; therefore,

$$p_2 - p_1 = \frac{\rho v_1^2}{2} = \frac{\rho v^2}{2}.$$

If one were to consider that the pressure behind the plate is equal to ambient pressure, i.e. p_1, then we get the drag force, multiplying $p_2 - p_1$ on plate area S:

$$X = (p_2 - p_1) S = \frac{\rho v^2}{2} S. \qquad (7.10)$$

Experiment, however, does not confirm the obtained relations even in this simplest case. This is understandable. In the derivation a simplifying assumption is made that $v_2 = 0$. This is true only in the critical front point, whereas spreading from this point the air has a speed other than zero. Then it was assumed that directly after the plate the pressure was equal to the pressure of undisturbed flow. This is also incorrect. Pressure here will be somewhat lower.

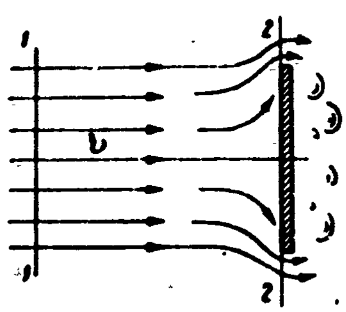


Fig. 7.12. Transverse streamlining of a plate.

For a round plate in an experiment at small speeds the value of X is obtained approximately 11% higher than that which formula (7.10) gives. As a result, it turns out that this variation at not very high speeds remains constant and independent of flow speed and absolute dimensions of the plate.

Thus, if one were to introduce a correction factor 1.11, then for a round plate the drag force is

$$X = 1{,}11 \frac{\rho v^2}{2} S.$$

In general, introducing correction factor c_x, called the drag coefficient, we can write

$$X = c_x \frac{\rho v^2}{2} S, \tag{7.11}$$

where c_x at low speeds depends only on the form of the plate and its location with respect to the flow.

By formula (7.11) the drag of any body is determined. For area S, here for an axially symmetric body, the area of projection of the body on a plane perpendicular to the axis of symmetry is taken (midship section), or any other so-called characteristic area.

Arbitrariness in selection of area S influences magnitude c_x. Therefore, in every case where numerical values of c_x are used, it is necessary to explain what area is used as characteristic.

Analogous to expression (7.11) are the formula both for lift and for drift force:

$$\left.\begin{aligned} Y &= c_y \frac{\rho v^2}{2} S, \\ Z &= c_z \frac{\rho v^2}{2} S, \end{aligned}\right\} \tag{7.12}$$

where c_y and c_z are coefficients of lift force and drift force;

S is the same characteristic area as in expression (7.11).

Expressions of aerodynamic forces have precisely the same form also in a connected coordinate system:

$$R = c_R \frac{\rho v^2}{2} S,$$
$$N = c_N \frac{\rho v^2}{2} S,$$
$$T = c_T \frac{\rho v^2}{2} S.$$

Inasmuch as in all formulas magnitude $(\rho v^2/2)S$ is general, between coefficients of aerodynamic forces the same relationships exist as between the forces themselves, and expressions (7.1), written for the case of absence of a slide angle β, take the form

$$c_R = c_x - c_y\alpha,$$
$$c_N = c_x\alpha + c_y.$$

At not very high speeds of flight, corresponding to Mach numbers not exceeding 0.5-0.6, the coefficients of aerodynamic forces can be considered not depending on speed. The shown limit of speeds to recent time exactly corresponded to speeds attained in aviation, and aerodynamic coefficients c_x, c_y, c_z were defined as constants for a given form of aircraft as a whole or for any of its parts.

These coefficients were before thus considered as coefficients of body form. Subsequently, however, it was clarified that forgetting the form of the body they depend also on speed, and more exactly on the relation of the speed to speed of sound, i.e. the Mach number. Through approach of flight speed to speed of sound their values sharply increase. At high, supersonic, speeds they decrease, asymptotically nearing to a certain constant value. The character of changes in c_x and c_y, depending upon the Mach number and angle of incidence α, is shown on Fig. 7.13 and 7.14 for a ballistic rocket of definite form. It is necessary at the same time to indicate that c_x and c_y in practical cases in some measure depend also on other

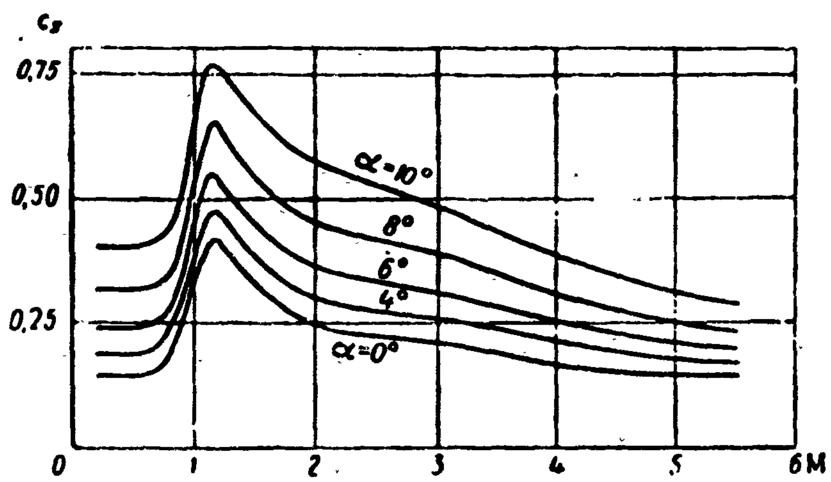


Fig. 7.13. Dependence of drag coefficient c_x for a ballistic rocket on the Mach number and the angle of incidence α (coefficient c_x refers to the midship section of a rocket).

factors — on absolute dimensions of the body, air density, and so forth.

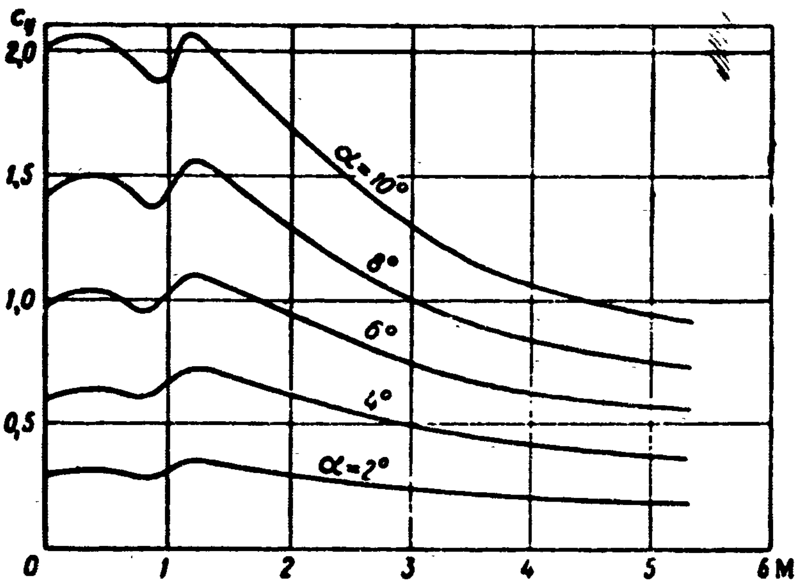


Fig. 7.14. Dependence of lift coefficient of a rocket c_y on the Mach number and angle of incidence α (coefficient c_y refers to midship section of the rocket).

In either case, in rocket technology where a full range of speed changes from low to great supersonic must be considered, aerodynamic coefficients are considered as functions of speed v or more exactly the Mach number. In this sense formulas (7.11) and (7.12) lose their basic content, establishing proportionality between aerodynamic forces and impact pressure. However in force of their simplicity, convenience, and certain established traditions in aviation, they remain as the basic calculating formulas independent of velocity.

As we have seen, aerodynamic forces depend also on the angle of incidence α. This dependence is expressed through coefficients c_x and c_y in function α at a constant S. For an axially symmetric rocket, obviously, this function for c_x will be even, and for c_y — odd:

$$c_x(-\alpha)=c_x(\alpha),$$

$$c_y(-\alpha)=-c_y(\alpha).$$

Dependences of c_x and c_y on the angle of incidence α are shown on Fig. 7.15 (for a V-2 rocket at M = 2), and also on the curves in Fig. 7.13 and 7.14.

It is essential to note that at small α the drag coefficient can be considered depending on the angle of incidence very little. The lift coefficient depends on the angle of incidence linearly.

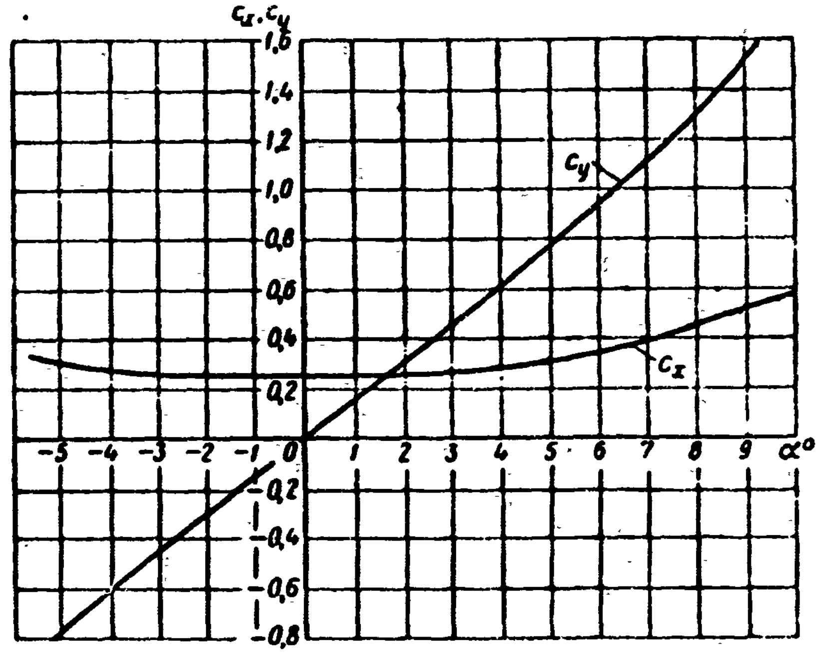


Fig. 7.15. Dependence of coefficients of drag and lift of a ballistic rocket on the angle of incidence at M = 2.

Component Parts of Aerodynamic Forces and Subsonic Streamlining

The character of streamlining and of aerodynamic forces connected with it are different depending upon whether the flight speed is subsonic or supersonic. However, in both cases the drag force can be divided into two basic components, frictional force and pressure force:

$$X=X_{тр}+X_{д}.$$

[тр = fri = friction, д p = pressure]

Force X_{fri} is understood as the resultant of resistance forces p_t, tangent to the body surface; while force X_p – as the resultant forces p_n normal to the surface (Fig. 7.16):

$$X_{тр} = \int_S p_t \sin\chi \, dS;$$

$$X_{д} = \int_S p_n \cos\chi \, dS,$$

where dS – element of rocket surface;

χ – local angle of surface normal.

Forces p_t and p_n refer here to a unit of surface area of the body.

It is possible to write further that

$$X_{тр} = c_{x\,тр} \frac{\rho v^2}{2} S,$$

$$X_{д} = c_{x\,д} \frac{\rho v^2}{2} S,$$

and to obtain hence an expression for the general drag coefficient in the form

$$c_x = c_{x\,тр} + c_{x\,д}.$$

The drag coefficient of friction is determined by air viscosity. In the simplest cases, for instance for longitudinal streamlining of a plate, it is possible to calculate magnitude $c_{x\,fri}$ at different flow speeds. There exist several approximate semi-empirical formulas, which give the possibility of estimating the magnitude of coefficient $c_{x\,fri}$ for bodies of any form, if one were to assume that the magnitude of the "mcistened" surface determines the frictional force independently of the form of this surface.

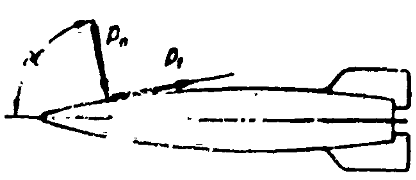


Fig. 7.16. Normal and tangent aerodynamic forces acting on the rocket surface.

The influence of viscosity on drag is not limited by simple friction, as was discussed above. Presence of viscosity can essentially influence also the distributive law of normal pressures, forming a second component of resistance X. Depending upon viscosity, especially for poorly streamlined bodies, a breakdown of flow occurs in the boundary layer at larger or smaller speeds, which directly affects the distributive law of pressures.

A theory considering a liquid ideal, without viscosity, can lead to results unexpected at first glance. In particular, the paradox of Euler — d'Alembert is known: any body at not very fast motion in an ideal liquid does not experience drag. Thus, for instance, a cylinder with transverse streamlining (Fig. 7.17a) shows forces of pressures symmetric about a plane perpendicular to the direction of flow. These forces come to equilibrium and have a resultant force. The presence of viscosity leads not only to appearance of tangents of forces p_t, but what is still more important, to a change of the actual character of streamlining. In a certain zone the flow breaks away from the surface of the body, as a result of which intense eddy formation appears behind the cylinder. Pressure acting on the cylinder from behind turns out to be less than ambient pressure (Fig. 7.17b). The pressure forces form a resultant directed against the motion of the cylinder.

The appearance of a frontal force in this case and its absence in the first case is understandable also from the power point of view. During motion of the cylinder in an ideal liquid, streams of the latter being interrupted before the cylinder, converge behind it and remain motionless. Consequently, during uniform motion of the cylinder in an ideal liquid work is not produced. Otherwise the second case holds. Here behind the cylinder a vortex trail will be formed possessing kinetic energy. On transmission of this energy to particles of liquid or gas, work of a force is expended during uniform motion of a body in a viscous medium.

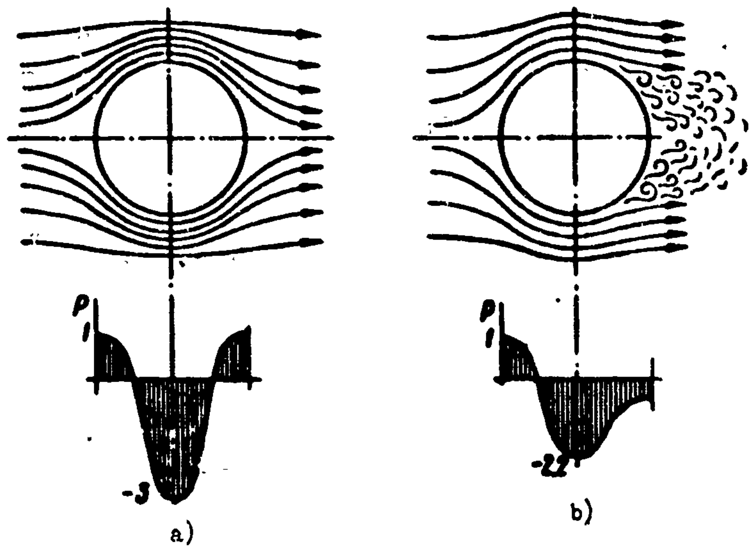


Fig. 7.17. Appearance of resistance to pressure during subsonic speeds of streamlining. a) ideal transverse streamlining of cylinder without breakaway of flow, b) real transverse streamlining of cylinder with breaking away of flow and asymmetric distribution of pressure.

From what has been said it follows that separation of the boundary layer and eddy formation increase drag. For decrease of the drag force in subsonic flow it is necessary to give the profile of a body smooth outlines that promote streamlining without breakdown of flow, i.e. to make, so to speak, the form of the body streamlined. On Fig. 7.18 such streamlined shapes are shown: profile of an aircraft wing intended for flight at subsonic speeds, aircraft fuselage, and a cylinder, enclosed in a streamlined unit.

It is important to note that all streamlined bodies have smoothly convergent outlines in the rear, not allowing breakdown of flow, and rounded outlines on the front. Pointed shapes are less profitable in subsonic flow. A sharp edge turbulates the flow and thereby increases drag.

At subsonic speeds of flight the resistance of friction for a streamlined body is the main part of the general resistance, and the resistance of pressure here plays a secondary role. For unstreamlined bodies as, for instance, a sphere or cylinder, the resistance of pressure is essentially greater than the resistance of friction.

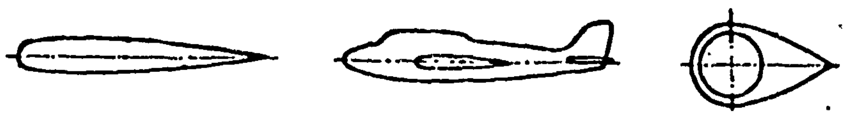


Fig. 7.18. Examples of bodies having a streamlined shape.

Considering the diagram of distribution of pressures shown on Fig. 7.17, we notice that excess pressure on the body surface (with respect to pressure of undisturbed flow) may be both positive and negative. In connection with this, for solids of rotation (body of a rocket) the resistance of pressure is divided into two parts, head resistance (resistance from excess pressure distributed on the body) and base, appearing due to rarefaction under the base (base suction). In the same way resistance is divided also at supersonic speeds.

During flight of rocket the air ahead of it becomes condensed. Here heightened excess pressure is created, the magnitude of which determines "pressure resistance." This resistance increases with growth of speed without limit. Behind the rocket a rarefaction will be formed — a base vacuum, determining suction resistance. This component of resistance also increases with speed, but not infinitely. It cannot be larger than the magnitude corresponding to an absolute base vacuum.

Let us remember that the force influence of absolute pressure of undisturbed atmosphere on a rocket is included in the thrust force; in aerodynamics the distribution of excess pressures is considered.

Figure 7.19 shows the distributive law of pressure on the surface of the rocket body.

With the presence of angle of incidence α, the character of streamlining of the rocket will be apart from that which has place at $\alpha = 0$. In this case lateral overflowing of air appears from a zone of heightened pressure into a zone of lower pressure, leading to additional vortex formation and additional expenditure of energy on their formation. The corresponding additional resisting force is called induced drag. On Fig. 7.20 lines of additional currents of air and vortexes forming from them are shown by arrows.

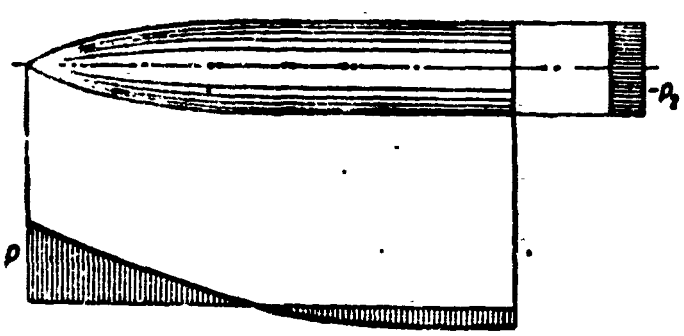


Fig. 7.19. Distribution of aerodynamic pressure on a rocket body.

The division of the resisting force into separate components shown here remains valid both for subsonic and supersonic streamlining. However, during transition beyond the speed of sound an additional component of resistance of pressure forces appears; the character of streamlining sharply changes, and accordingly the approach to selection of the most rational aerodynamic forms is changed.

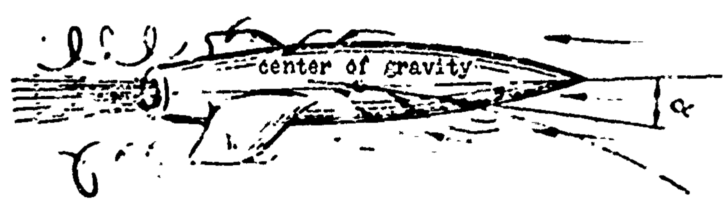


Fig. 7.20. Formation of vortexes, leading to induced drag during flight of a rocket with an angle of incidence.

Peculiarities of Supersonic Streamlining

Let us return to the transformations which were conducted in Chapter VI [formula (6.19)] for determination of the speed of sound. For speed of propagation of a compressional wave the expression

$$a_{yд}=\sqrt{\frac{\rho+\Delta\rho}{\rho}\frac{\Delta p}{\Delta\rho}}.$$

[уд = s.w = shock wave]

At very weak (sonic) perturbations, when ρ + Δρ ≈ ρ, this expression becomes the formula for the speed of sound

$$a=\sqrt{\frac{\Delta p}{\Delta\rho}},$$

whence in assumption of adiabatic process of gas compression during sonic oscillations

$$a=\sqrt{k\frac{p}{\rho}}=\sqrt{kgRT}$$

is obtained.

From a comparison of expressions for the speed of propagation of a shock wave and the speed of sound, it follows that strong perturbations always spread faster than sound.

Let us assume that in a motionless gas medium there is a constant point source of weak perturbations. We will suppose that in a certain moment this source creates a local condensation of gas. Due to this a sonic spherical wave appears, which will start to spread evenly in all directions with the speed of sound a. After time t the wave front will be separated from perturbation source by a distance at. If the source of perturbations sends signals periodically, the fronts of all waves will fall into concentric spherical surfaces with the center at the source of perturbations (Fig. 7.21). The picture of propagation of these waves is analogous to propagation of waves on the surface of water from the place of drop of a stone cast into the water.

We will now suppose that the perturbation source is motionless, and the gas medium moves about it with speed v, where v < a. During the time t between two signals from the perturbations source, the first wave will travel to a magnitude vt, and the centers of the spherical waves will be displaced relative to each other by that same magnitude vt. At the same time, inasmuch as v < a, the waves will not cross over each other (Fig. 7.22).

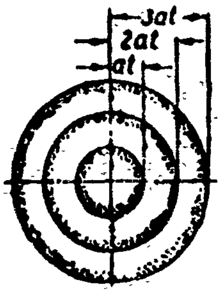


Fig. 7.21. Propagation of weak waves in a motionless medium.

Otherwise the condition v > a holds, i.e. in that case when a flow streamlining a body will be supersonic. In this case the family of spherical waves gets a conical enveloping surface (Fig. 7.23).

Thus, the character of propagation of perturbations turns out to be different depending upon the flow speed.

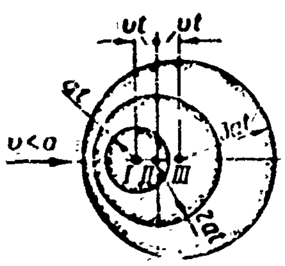


Fig. 7.22. Propagation of weak waves in medium moving with a speed v < a.

While the speed of flow was below the speed of sound, the perturbations spread in all directions, although not nonuniformly, but both with the flow and against it. When the speed of flow turned out to be larger than the speed of sound, the perturbation began to spread only in one way — with the flow. The enveloping conical surface — a so-called Mach cone — is here the surface that limits the region of propagation of weak perturbations in space. This surface is called the threshold wave of weak perturbations.

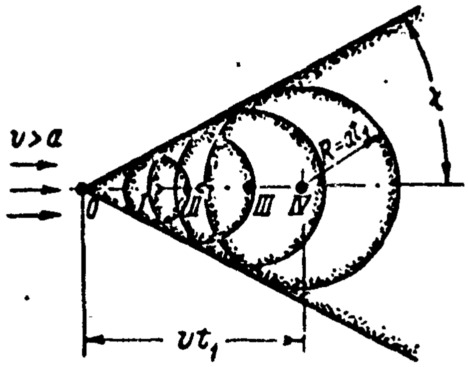


Fig. 7.23. Propagation of weak waves in a medium moving with a speed v > a.

The angle at the summit of a cone of weak perturbations (Mach angle) depends on the relation of flow speed to speed of sound. From Fig. 7.23 it is clear that

$$\sin\chi = \frac{a}{v} = \frac{1}{M}.$$

With a decrease of flow speed, angle χ is increased, and at v = a becomes equal to 90°. This means that the cone of perturbations is turned into a plane. As a result, signals sent by the source of perturbation attain any point located behind the source.

We will assume now that the source of perturbations is able to continuously send not only weak acoustic, but also strong shock waves. Let us see how these waves spread in supersonic flow.

A shock wave, we already know, spreads with a speed $a_{s.w}$, higher than the speed of sound. Therefore, in the first moment a wave departed from the source of perturbations can spread with sufficient initial power against the supersonic flow. Due to propagation, this wave will weaken and its speed will decrease, nearing in the end to the speed of sound. Under these conditions the envelope of the family of spherical waves will no longer be a simple conical surface. This will be a surface with its front end similar to a hyperbola and then transform into a cone of weak perturbations. On Fig. 7.24 such an enveloping surface is shown. Where the force of the shock wave is larger (directly ahead of the source), the envelope is traced by a thicker line. Appearance of threshold waves, i.e. concentration of perturbations in a certain region of space, is a distinctive peculiarity of supersonic flows.

Threshold waves of weak perturbations are formed during streamlining by supersonic flow of sufficiently small obstacles. Practically, the source of weak perturbations may be any point of a streamlined surface.

Shock waves appear always during streamlining by supersonic flow of bodies which no longer can be considered small obstacles.

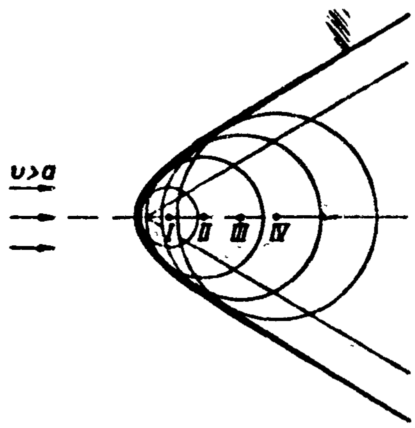


Fig. 7.24. Envelope of a family of spherical shock waves during motion with a speed $v > a$.

During streamlining of such bodies, in definite sections the flow experiences a local braking, as a result of which its compression occurs. Thus, for instance, supersonic flow will experience compression directly ahead of the streamlined body, and also in places of streamlining of concave sections on its lateral surface. As before every element of surface, will be a source of weak perturbations; however, it is possible to show that threshold waves of these perturbations now will be concentrated in one wave front with formation of a shock wave.

Let us consider streamlining of a certain blunt body with supersonic flow (Fig. 7.25) thoroughly.

Perturbations in the environment of points located on the front flat part of a streamlined body, form a zone of heightened density ahead of the body. As a result of this a shock wave appears, possessing the property to spread with a speed greater than the speed of sound. Thus, a wave appears moving toward the flow. However, it cannot go far forward. As soon as the distance between the wave front and the

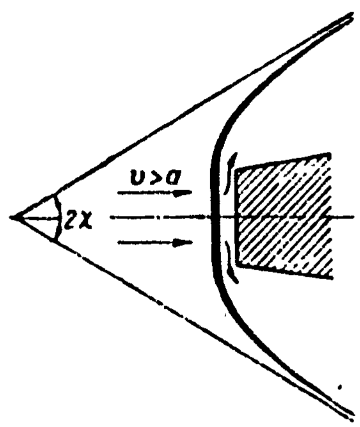


Fig. 7.25. Shock wave before a blunt body at supersonic flight speed.

leading edge is increased, a possibility appears of overflowing of gas from the zone of heightened pressure into the sides (as this is shown by arrows on Fig. 7.25). Here the force of the shock wave, i.e. the drop of pressures appearing on it, will fall, and the speed of its propagation will decrease. Thus, in conditions of established movement in front of a body subjected to supersonic flow, there will constantly be a shock wave, moving forward with the speed of supersonic flow. The distance between the wave and body depends on the form of the body, determining the possibility of overflowing of gas into the sides, and on the approaching stream's velocity.

Beyond the borders of the nose part of the streamlined body, the front of the shock wave is bent, and intensity of the wave decreases. As a limit, the angle of inclination of a shock wave made by the direction of speed becomes equal to the Mach angle, and the shock wave itself transforms into a common acoustic wave.

Gas flow during transition through the front of a shock wave sharply changes its parameters: the flow speed drops, pressure and density are increased, temperature increases. These changes occur so sharply that it is permissible to consider them intermittent. Because of this, supersonic aerodynamics shock waves are also called shock waves.

Shock waves appear not only before, but also from behind a streamlined body.

Let us consider the tail end of a body, passed by a supersonic flow, or a body flying with supersonic speed (Fig. 7.26).

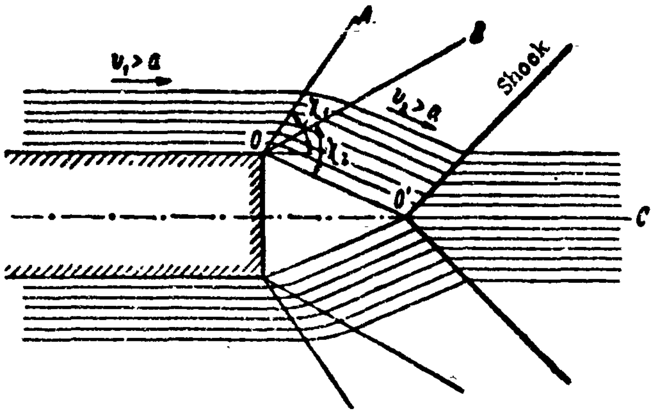


Fig. 7.26. Diagram of formation of tail shock wave.

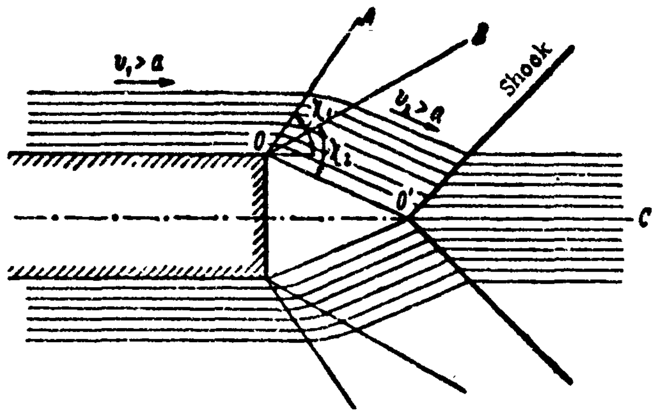
Conditionally we will consider streamline flow to be flat.

We will take the corner of the rear section of the body for a source of weak perturbations.

Let us assume that flow approaching point O has a speed v_1, where $v_1 > a$. Before line OA, the incident flow will remain undisturbed. The position of line OA is determined by angle $\chi_1 = \text{arc sin } a/v_1$. With rounding of the corner the flow should be expanded. For supersonic flow, expansion, as it is known, is accompanied by an increase of speed. Consequently, after line OA the flow speed is increased, and the flow will turn. After a certain line OB the flow will completely turn and will flow in a new direction with speed $v_2 > v_1 > a$.

The presented illustration is somewhat simplified, inasmuch as the boundaries of streams of the turned flow will not be parallel, but it is indisputable that during streamlining of the external corner we do not get shock waves. Lines OA and OB are waves of weak perturbations.

We will trace the further behavior of the turned flow. We notice that the flow should again turn with the same angle but in the opposite direction, encountering precisely the same flow, streamlining the lower side of the symmetric body. Therefore, flow passing with speed v_2 will act as if it collides at a certain angle with a rigid obstacle O'C (see Fig. 7.26). In the zone of compression of flow a shock wave appears.



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wave), it should possess a greater force than in the case of motion slanted to the flow (oblique shock wave). In the limit, when the wave and flow create a Mach angle its force in general may be insignificantly small.

For lowering of wave impedance it is expedient to give such a form to the rocket, so that shock waves, since they are inescapable, are slanted, and the angle of their slope with respect to the rocket axis is as close as possible to the Mach angle.

After the above it becomes clear that so-called streamlined bodies, having the least frontal resistance in subsonic flow, in supersonic become poorly streamlined. On Fig. 7.28 streamline flow is shown of a subsonic wing profile by a supersonic flow. Before the wing a curvilinear jump is established, close to straight, as a result of which the wave impedance of such a wing turns out to be significant.

With a peaked profile or coneshaped heat of a rocket, the frontal shock wave is slanted, and at a very acute angle of the leading edge it approaches a wave of weak perturbations. Losses of energy in such a shock wave will be much smaller.

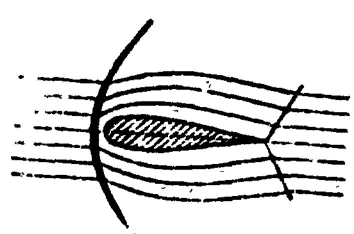

Fig. 7.28. Streamlining of a subsonic profile by supersonic flow.

On Fig. 7.29 change of the form of the shock wave is shown depending upon the angle of sharpening of the frontal edge at a constant streamlining speed.

Due to thickening of the leading edge or increase of the angle of a cone the shock angle is increased. When angle of the cone or wedge for the leading edge becomes equal to the so-called critical angle, the shock wave in the central part will become straight and will depart from the point. Wave impedance with such a jump will be significantly greater.

Thus, for the purpose of lowering of wave impedance, it is desirable to give as sharp a form as possible to the front of rocket.

Also, however, it is necessary to consider several other circumstances. With lengthening and sharpening of the front section only the wave impedance is lowered, but at the same time, inasmuch as the surface is increased, the resistance of friction increases. Furthermore, excessive lengthening of the head section can lead also to difficulties of construction, inasmuch as the difficulty of positioning the stearing equipment is increased.

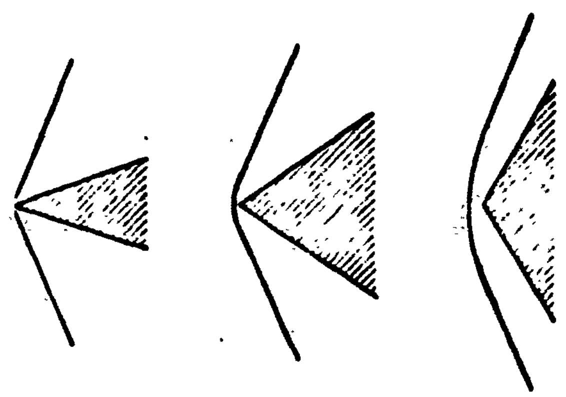

Fig. 7.29. Dependence of the form of jump on the angle of sharpening of the leading edge at a constant flow speed.

In most cases the front section of rockets and missiles is given either a conical or a so-called ogival form, which is outlined by arcs of a circle or arcs of a parabola (Fig. 7.30).

At high flight speeds of the order of M = 5 and higher, the problem of finding aerodynamic forms of frontal sections of rockets cannot be resolved apart from thermal phenomena connected with the appearance of the frontal shock wave. At high temperatures dissociation of gases entering into the composition of air occurs, and their thermodynamic properties change. This affects the character of streamlining. In recent years the question of application for intercontinental rockets (at M > 10) of head sections with a blunted nose outline has been discussed.* As a result of appearance of a powerful shock

*See, for example, Aviation Week, 1957, 20, Vol. 66, p. 51.

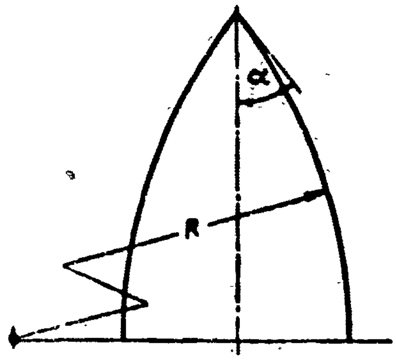

Fig. 7.30. Form of the ogival outline of the head section.

wave a significant part of the thermal energy, as is assumed, is dispersed in the atmosphere, and only a small share of it will be absorbed by the head section. Thus, profiling of the head section of a rocket in certain cases is determined not by the condition of achieving good streamlining by its supersonic flow, but by the necessity of protecting the material of the head from thermal action at its entrance into the atmosphere.

Shock Stall

Wave impedance, which is the result of the appearance of shock waves, has place also at subsonic speeds of flight close to the speed of sound. The fact is that in these conditions the local flow speed during streamlining of wings, tail group or other parts of the aircraft or rocket, can exceed the speed of sound, and in corresponding places shock waves can appear.

The flight speed at which appears a local speed of sound at the surface of the body is called the critical speed of flight, and the phenomenon of sharp increase at which the first shock waves appear, is called shock stall.

In order to understand the appearance of shocks at subsonic flight speed, we will be satisfied at first with the fact that supersonic flow passes into subsonic only through a shock wave. Flow in which a gradual transition from supersonic speed to subsonic is carried out is unstable.

Actually, in order to decrease the speed of supersonic flow, the area of flow section should be decreased. After the flow speed becomes equal to the speed of sound, the section area should be increased for

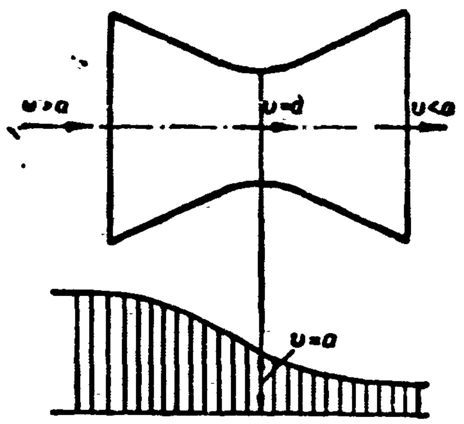

Fig. 7.31. Imaginary smooth braking of supersonic flow in a reversed Laval nozzle.

further braking of flow. Thus, we arrive at a diagram of a reversed Laval nozzle (Fig. 7.31). Flow through a Laval nozzle was described earlier in Chapter VI.

Let us assume that the flow speed smoothly changes from supersonic to subsonic (see Fig. 7.31). In the critical, the narrowest, section of the nozzle the flow speed will be equal to the local speed of sound.

Every point in the wall of the nozzle, and the gas itself, can be considered as sources of weak perturbations. Let us take any point in the subsonic region of flow, for instance point 0 on Fig. 7.32. Perturbations emanating from this point will spread ~~in~~ all directions to any distance along the flow, and only as far as the critical section against the flow. The same condition holds for perturbations from all points located in the subsonic part of flow. As a result, perturbations from different sources will be summated in the environment of the critical section, and the wave appearing thus obtains the ability to move toward the flow with a speed greater than the speed of sound.

In a certain section the speeds of flow and wave (shock) are equal, and the position of the shock should be stabilized.

In practical cases, for a diffuser the described braking of supersonic flow leads to the situation where a straight shock front exceeds the bounds of the diffuser and is located directly before the entrance, but after the shock, subsonic speed is established immediately (Fig. 7.33).

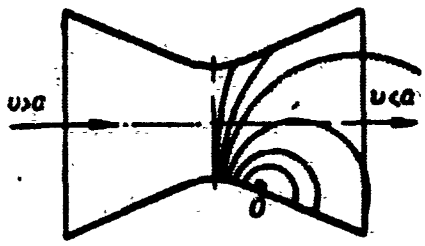

Fig. 7.32. Propagation of weak waves of perturbations in the subsonic part of flow.

Thus, transition from supersonic speed to subsonic occurs by a shock wave. This is explained by the fact that in the flow there is always a sufficient quantity of perturbation sources, forming by an integral way a wave able to move toward the supersonic flow.

We will return to the question of shock stall.

During streamlining of a body, certain streams of flow at first are narrowed, and then are expanded as is shown, for instance, on Fig. 7.34. In such streams growth of speed to supersonic, as in a Laval nozzle is possible with a sufficient drop of pressures. However, inasmuch as the general speed of flow is less than sonic, reverse transition should take place to a speed lower than the speed of sound. But now we already know that this transition occurs through a shock wave. This is how shock waves appear at subsonic flight speeds.

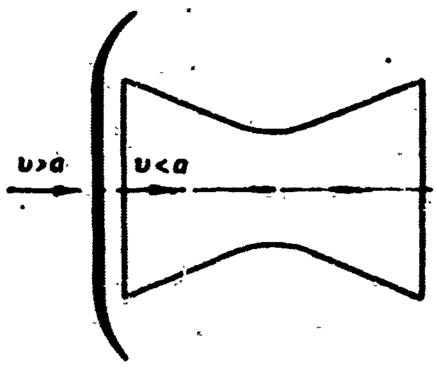

Fig. 7.33. Shock before entrance into the diffuser.

On Fig. 7.35 shock waves are shown forming at the surface of a flying bullet. These shocks are formed at speeds close to the speed of sound (M = 0.900 and M = 0.946).

In contemporary aviation, having mastered subsonic, sonic and supersonic flight speeds, the question of shock stall is very serious. At appearance of the first shock waves the force of aerodynamic drag sharply increases, aerodynamic loads grow, and the character of streamlining is qualitatively changed with redistribution of pressure forces on the aircraft surface. All this leads to large power

Fig. 7.34. Change of stream section during streamlining of a body with a wing profile.

overloads of the structure and a change of the degree of stability and controllability of the aircraft.

The appearing difficulties are being surmounted at present, on the one hand, by investigation and solution of questions of stability and controllability in the stage of shock stall, and on the other, -- by application of structure shapes eliminating the appearance of shock stall. To the latter pertains, for instance, transition to thin symmetric wing profiles and to arrow-shaped wings.

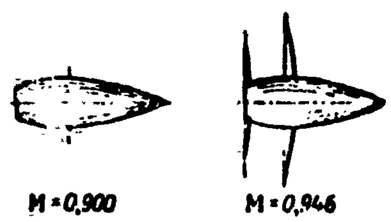

Fig. 7.35. Shock waves forming at the surface of a bullet at large subsonic flight speeds.

For those rockets, a great portion of the trajectory of which is passed with supersonic speed (the majority of rockets), elimination of shock stall does not have meaning. Here it is necessary to detect the most rational forms for conditions of transition through speed of sound, and also for supersonic flight.

Determination of Aerodynamic Properties

From all that was said above, it is clear that the mechanism of appearance of aerodynamic forces is very complicated. To find them by means of calculation even in the simplest cases is often very difficult. Therefore, during determination of aerodynamic forces (i.e. magnitudes of aerodynamic coefficients c_x and c_y) at present approximate calculations are used which then are corrected by experiment.

During computation of aerodynamic coefficients, in the first stage the design of the rocket body is separated into simple elements — head part, cylinder, base, stabilizer. For each of these elements the constituents of the aerodynamic coefficients can be determined either by theoretical calculations (if the geometric form of the element is simple), or on the basis of comparing results of wind tunnel tests conducted earlier for bodies of similar form. Summation of coefficients with correction for mutual influence of components on the rocket elements gives the aerodynamic coefficient of the actual rocket.

Results obtained in such a way need more precise determination. The basic means of more precise determination is the testing of a geometrically similar model of the rocket in a wind tunnel.

The wind tunnel is a channel in which a gas flow with a given regulated speed is artificially created.

The simplest machine is a pipe of the closed type. On Fig. 7.36 a diagram of a such pipe for subsonic speeds is depicted. In this pipe flow is created with the help of a blower 1. In the grid 2 the flow is straightened and enters into a narrow operating canal 3, in which the tested object is located. In the diffuser 4 braking of flow occurs (subsonic flow is braked during expansion), after which the air again enters the blower. As a result of the fact that the pipe is closed, the work of the blower goes only to replenishing losses in the flow, inasmuch as the impact pressure in the diffuser before entrance into the blower is restored in the form of pressure energy. For an open pipe the energy of flow would be lost completely.

The described closed circuit is useful for creation not only of subsonic, but also of supersonic flows. For that, obviously, before passing the object, gas should pass through a Laval nozzle, as shown on Fig. 7.37. It is necessary, of course, to ensure supercritical drop of pressure on entrance and exit out of the nozzle. Speed of flow also depends on the broadening of the nozzle. For

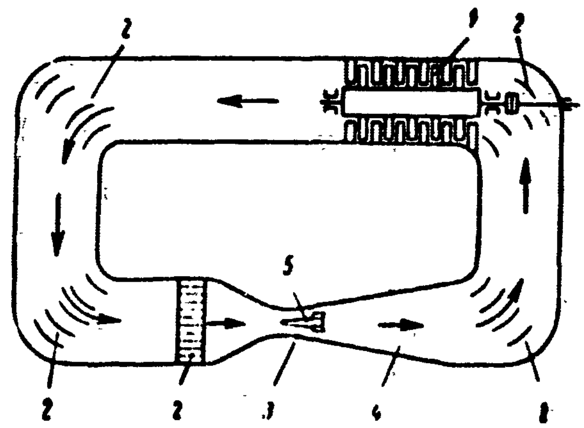

Fig. 7.36. Diagram of wind tunnel for subsonic speeds of the closed type. 1) blower, 2) straightening grids, 3) operating canal, 4) diffuser, 5) model.

change of speed it is necessary, obviously, either to make different nozzles, or in construction of the pipe to anticipate the possibility of smooth change of the geometric form of the channel. In contemporary tunnels the first method is applied more frequently.

The object being tested in the tunnel should occupy a small part of the operating section of the channel, so that the flow speed in this section essentially does not change and constriction of the tunnel, so to speak, does not occur. Since testing of the natural size can be done only with small rockets, experiments are conducted in a wind tunnel with models that are geometrically similar to the investigated rockets. Inasmuch as at high flight speeds the character of streamlining depends primarily on the compressibility of gas, during the test a condition of similarity by Mach number should by all means be observed. In other words, the model should be tested in the tunnel at the same Mach number at which it is necessary to obtain aerodynamic coefficients in nature.

Observance of similarity by Reynolds number, considering the influence of forces of viscosity is also very desirable:

$$\mathrm{Re}=\frac{\rho v l}{\mu},$$

where μ – coefficient of viscosity of air or gas in which testing is done;

l – certain characteristic linear dimension of the rocket.

Fulfillment of the last condition, i.e. equality of Re for the

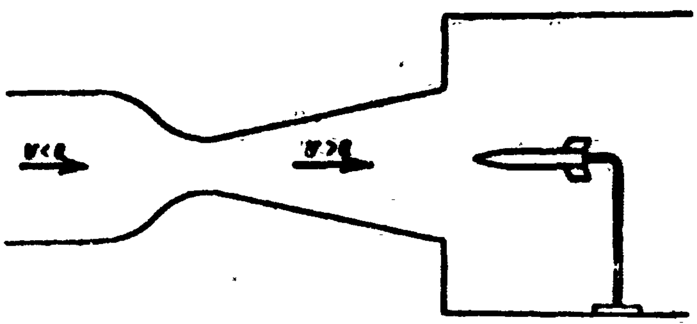

Fig. 7.37. Model tested in a supersonic wind tunnel.

model and nature, is met with great difficulties, and in practice of wind tunnel tests is usually not sustained. However in treatment of results of the experiment, this circumstance is considered by introduction of necessary corrections.

In wind tunnels measurements of component forces and moments acting on the model in the flow are produced. For this purpose a special wind tunnel balance, on which the tested model is placed, is used.

Wind tunnel balances have various construction and can register any number of the six components of the total aerodynamic force and moment. Depending upon this they are called two-, three-, four-, or six-component. For an axially symmetric model of a rocket usually there is no necessity to measure all six components, inasmuch as a change of the plane of the angle of incidence α leads to identical results. Here for the most part three-component balances are used. Construction of such balances is shown on Fig. 7.38.

In the rear part of the model a hole is drilled, with help of which the model is tightly fitted onto the cylinder 1. The cylinder has an elastic arc 2, reacting to the axial force. Under the action of this force arc 2 is bent, as a result of which a tensometeric transducer of resistance D_1, glued on its surface, is extended, and its resistance changes. The transducer through electric leads 5 is plugged into the bridge circuit, allowing determination of drag under conditions of preliminary calibration testing.

Forces perpendicular to the axes are transmitted from the model

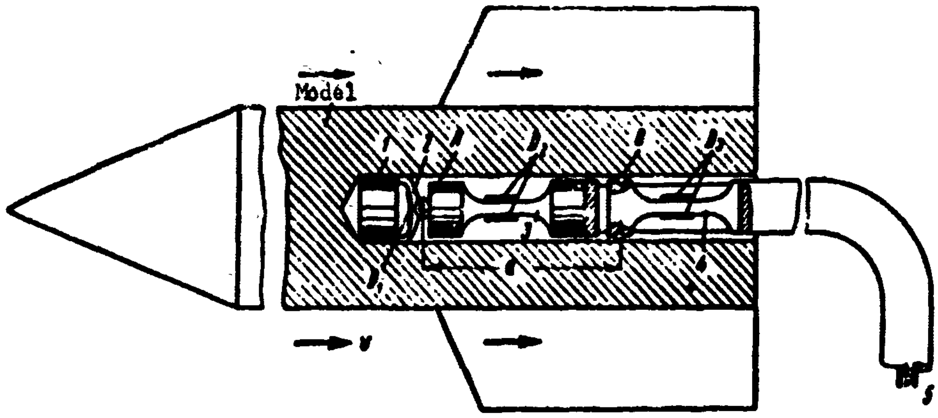

Fig. 7.38. Diagram of a three-component wind tunnel balance. 1) front cylinder, 2) elastic arc, 3) front elastic neck, 4) rear elastic neck, 5) electric leads, A and B – hinge points, D_1, D_2, D_3 – transducers of resistance.

onto the holder through points A and B on base a. Forces acting on points A and B create bending moments in the elastic necks 3 and 4. On these necks transducers of resistance (D_2 and D_3) are also glued. By the indication of these transducers two forces are determined in points A and B. Their sum gives the normal force. Since the position of points A and B with respect to the model are known, then by means of reduction of forces to the center of gravity it is easy to determine the aerodynamic moment in a vertical plane also. Consequently, with the help of the described circuit components R, N, and M_a are determined.

If measuring of the aerodynamic force for the model is performed, it is possible to consider the corresponding aerodynamic coefficient found. For instance, for axial forces acting on the model and in nature, we have

$$R_{\text{мод}}=c_{R\,\text{мод}}\left(\frac{\rho v^2}{2}\right)_{\text{мод}}S_{\text{мод}};$$

$$R_{\text{нат}}=c_{R\,\text{нат}}\left(\frac{\rho v^2}{2}\right)_{\text{нат}}S_{\text{нат}}.$$

[мод = mod = model, нат = nat = nature]

Under the condition of observance of similarity in the M and Re numbers

$$c_{R\,\text{мод}} = c_{R\,\text{нат}}.$$

In those cases, when it is necessary to determine not the actual aerodynamic forces, but the distributive law of pressure on the rocket surface, testing of a hollowed model is done on surface of which holes are drilled in many points. Through these holes, pressure is brought to manometers (Fig. 7.39). As a result of testing a picture of distribution of pressure forces on the body surface is revealed. After that by means of summation there is the possibility to find all components of aerodynamic pressure forces, and also the components of both their head and base drag.

At great supersonic speeds the method of testing models is met by difficulties connected with creation of wind tunnels ensuring high speed of flow. The fact is that the power of the blower, necessary for creation of an operational flow, at a constant area of passage section of the tunnel's working part and constant gas density in the working part, increases proportionally to the cube of flow speed. Therefore, for achievement of a speed exceeding a few times the speed of sound, it becomes difficult to ensure the necessary power for the blower. Thus, for instance, for obtaining of a flow with M = 5 at standard atmospheric pressure and at a diameter of the tunnel's working part of 1 meter, a power compressor of about 500,000 kilowatts would be necessary. Therefore in supersonic wind tunnels usually it is necessary to sharply decrease the dimensions of the working part and of the model correspondingly and to take all possible measures for lowering losses in flow.

At present there are many wind tunnels of brief action, in which the blower works for several hours, creating in the sealed pipe a

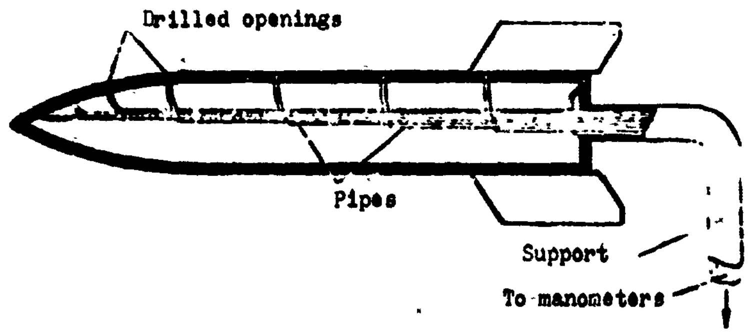

Fig. 7.39. Hollowed rocket model.

pressure of about 200 atm(tech). Then air is released within 20-30 seconds through the operating canal of the tunnel into the atmosphere. During that time, with the help of recording instruments, the characteristic of the model can be recorded. In such a type of wind tunnel the power of the blower may be comparatively small; however, the whole complex of equipment, especially for obtaining high flow speeds, becomes extraordinarily bulky.

Along with application of wind tunnels a flying test of models, fixed on the nose of a carrier rocket is possible. With such a test, data of measurements during flight are transmitted by radio and registered by high-speed equipment. Rescue of the model and rocket at their return to earth is possible.

Inasmuch as the flight of ballistic rockets passes through complicated conditions of variables in speeds and density of the medium, the most reliable information about the peculiarities of aerodynamic forces are obtained in natural tests. For that, a system of transducers of pressures and temperatures is established on the rocket. Indications of these transducers, included in a system of telemetries (see Chapter X), give the possibility to establish a full picture of the laws of change of aerodynamic influence on the rocket in its trajectory.

Influence of the Stream of a Rocket Engine on Aerodynamic Forces

The presence of a gas stream from a working engine introduces specific peculiarities in rocket aerodynamics.

As a result of the action of the stream the conditions for formation of the boundary layer, change especially near the tail part of the rocket, and an additional ejected flow along the component appears (Fig. 7.40). The degree of its influence on the resistance of pressure and friction turns out to be small and greatly depends on the conditions (overexpansion or underexpansion) of the nozzle's work.

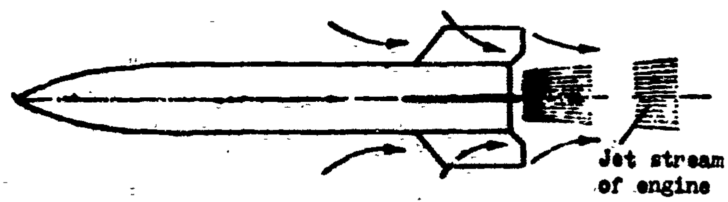

Fig. 7.40. Increase of external flow by the jet stream of the engine.

The jet stream has a basic influence on base drag. Behind the rocket with a working engine there is no rarefied space opposite the nozzle hold. Therefore, only part of the base drag, corresponding to the base area surrounding the nozzle is included in the resistance of the rocket. Thus, for instance, during calculation of the drag force for a V-2 rocket, with a working engine only 50% of the base drag is kept.

At low flight speeds this circumstance lowers drag by a noticeable magnitude. Changes of resistance due to ejection of circumfluent flow are negligible. At high speeds a decrease of drag, connected with the presence of a jet stream, becomes negligible, inasmuch as base drag plays a comparatively small role here.

Thus, during calculations of the rocket trajectory, drag becomes different depending upon whether all other conditions are equal in the sections of the trajectory passed by working or not working engines. It is necessary to note that since usually the tractive force exceeds drag many times for rockets, then during appraisal of aerodynamic forces for ballistic calculations, there is no need for more precise determinations. In this sense a rocket noticeably differs from an aircraft, for which during uniform flight along the horizontal the tractive force is equal to the resisting force.

Heating of the Rocket Body at High Flight Speeds

The aerodynamic influence of the medium on the body of a flying rocket is not solely power. At high speeds a thermal influence appears through heating of the rocket surface.

From the energy equation derived above (in Chapter VI) it follows that during motion of a gas in a pipe or during its streamlining of any bodies a transformation of some forms of energy into others occurs. Kinetic energy transforms into thermal and conversely. Decrease of kinetic energy (deceleration) obligatorily leads to an increase of gas, temperature and consequently also of the temperature of the streamlined body.

The gas speed can decrease with encounter with a streamlined body, and also due to friction in the boundary layer. Also, increase of gas temperature does not depend on causes evoking braking of flow.

Let us consider a body streamlined by a gas (Fig. 7.41).

At the leading edge of this body the flow divides into two parts. Part of the stream goes upwards, and part — downwards. It is obvious that in a certain point A for the front edge of the body the flow speed is equal to zero. Here, so to speak, full braking of flow occurs, and all the kinetic energy of the gas transforms into enthalpy. This

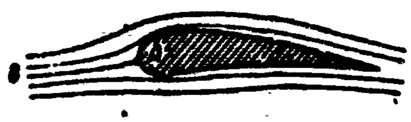

Fig. 7.41. Determination of braking temperature.

point is called the point of braking, or critical point. It is not difficult to determine the gas temperature at this point.

Let us separate from the total flow a stream of gas AB, where point B pertains to a section of undisturbed flow, and let us write for this stream the energy equation in the form (6.10):

$$\frac{v_B^2}{2}+\frac{k}{k-1}gRT_B=\frac{v_A^2}{2}+\frac{k}{k-1}gRT_A.$$

But $v_B = v$ and $v_A = 0$, therefore an increase of temperature during braking of flow at point A will be

$$\Delta T=T_A-T_B=\frac{v^2}{2}\,\frac{k-1}{kgR}.$$

The temperature in the critical point T_A is the <u>temperature of braking</u>.

For air k = 1.4, R = 29.27 kgm/kg deg, and

$$\frac{k-1}{2kgR}v^2\approx\frac{v^2}{2000};$$

here the temperature of braking is

$$T_{торм}=T+\frac{v^2}{2000}. \tag{7.13}$$

[торм = brake]

At low speeds the braking temperature is small. However, at supersonic speeds accessible by the rocket it turns out to be very significant. Thus, for instance, already at a flight speed of 1500 m/sec

$$T_{торм}=T+\frac{1500^2}{2000}=T+1125^\circ;$$

at an environment temperature of $T = 275^{\circ}$ absolute this will be 1400° absolute.

It is necessary to note that at high temperatures the derived formula is not quite true, inasmuch as dissociation of gas occurs. Therefore, magnitudes k and R entering into the written expressions at temperatures greater than 3000° absolute cannot be considered constants. As a result of dissociation, the braking temperature essentially drops. Furthermore, at high temperatures a significant yield of heat through radiation into surrounding space occurs, in consequence of which the gas temperature is also lowered. This lowering becomes all the more noticeable the greater the temperature.

If one considers not only the critical point, but also in general, all points of the surface of a streamlined body, then one should say, that here also, strictly speaking, total braking of flow occurs, inasmuch as the relative velocity of gas particles directly touching the motionless body is equal to zero. However, due to the fact that the braked particles are in close thouch with the rapid flow, intense discharge of heat through the boundary layer occurs; therefore, the temperature in so-called points of incomplete braking turns out to be essentially lower than the temperature in the critical point.

Registering of the distribution of flow temperatures on the surface of a streamlined body is a very complicated problem and is connected with analysis of peculiarities of the boundary layer. Detailed investigation of this question goes beyond the limits of this course. It is possible to indicate only that the gas temperature at the surface of a streamlined body in its different points, depending upon the geometric form of the body, density of the atmosphere, i.e. in general, on conditions of streamlining, oscillates usually within limits of 20-70% of the temperature in the critical point. Thus, considering everything said above, the conclusion can be made that formula (7.13) obtained above gives us only the limit of possible flow temperatures near the streamlined body.

In spite of calculation of all lowering factors noted above, the braking temperature sharply increases with flight speed, inescapably leading to intense heating of the structure. The body temperature of an aircraft here remains noticeably lower than the temperature of braked flow due to partial transmission of heat into the body, and also due to thermal radiation of the heated surface of the body into the surrounding space. Nonetheless, starting from speeds of 1000 m/sec the necessity appears of acceptance of constructive measures in connection with an increase of an aircraft's body temperature.

The shown influence of temperature essentially depends on duration of flight in the atmosphere. For aircraft apparatus, forced in virtue of the basic principle of motion, to be constantly in comparatively dense layers of atmosphere, external heating is a constantly acting factor, and during prolonged flight with great speed the temperature condition of the airplane fuselage is established. Therefore, the existence of terminal velocity becomes evident, directly defining thermostability indices of construction materials.

The ballistic long-range rocket has two sections of atmospheric flight — on takeoff and during drop (see Chapter VII). The first atmospheric section is surmounted with comparatively small speeds, and therefore, here protection of the structure from excessive heating does not present great difficulties. In the atmospheric section of drop, the speed of the ballistic rocket is very great, where with increase of distance the speed inevitably increases. In these conditions thermal protection of the structure is a sufficiently complicated technical problem.

If one were to return to the question of exhaust of combustion products from the nozzle of the rocket engine, then one should note that braking of flow occurs, and due to this, an increase of temperature for the nozzle walls in the boundary layer occurs. This, as already was mentioned, leads to an increase of heat flow through the wall.

During full braking of gas in the nozzle of a liquid-fuel rocket engine and absence of thermal losses, the kinetic energy of flow will be converted back into the enthalpy which the gas had in the combustion chamber. Consequently, in the absence of thermal losses the braking temperature of flow in the nozzle of a liquid-fuel rocket engine will be equal to the temperature in the combustion chamber. Similar full braking of gas and corresponding heating up take place, for instance, on the leading edges of jet stream steering mechanisms.

4. Static and Damping Moments

Static Aerodynamic Moment

It was already indicated above that th system of aerodynamic forces distributed on the surface may be given to any point of the rocket in the form of a resultant force and moment, the magnitudes of which depend on what point is used as the origin of the force system.

If the system of aerodynamic forces were located at the center of gravity of the rocket, then at an angle of attack not equal to zero, we, besides the above considered forces, will obtain a resultant moment — the so-called static aerodynamic moment M_{st}. If this moment is directed in the direction of decrease of the angle of incidence α, it is called the restoring or stabilizing moment (Fig. 7.42). In the opposite direction this moment is called overturning or destabilizing.

The magnitude of static moment is determined by a formula analogous to formulas by which forces of drag X and lift Y were determined:

$$M_{ст} = c_m \frac{\rho v^2}{2} Sl, \qquad (7.14)$$

where ρ, v, and S correspond to air density, flight speed and characteristic area;

l — certain characteristic linear dimension (usually length of rocket);

c_m — dimensionless coefficient of moment.

Coefficient c_m in first approximation (at small α) is proportional to the angle of incidence:

$$c_m = c_m^{\alpha}\alpha.$$

In contrast to all remaining aerodynamic properties, the magnitude of coefficient c_m^{α} depends not only on the Mach number which corresponds to speed and altitude of flight. Inasmuch as aerodynamic forces are given to the center of gravity of the rocket, and the center of gravity itself shifts along the axis due to expenditure of fuel, then it turns out that magnitude c_m^{α} depends also on the internal distribution of masses in the rocket. In connection with this, as a measured static moment it is convenient to select instead of c_m^{α} another characteristic.

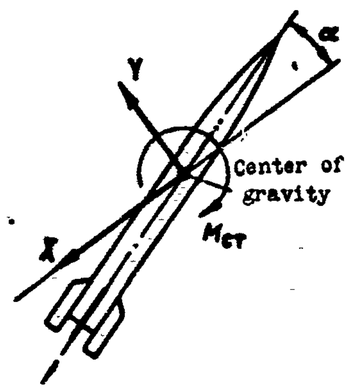

Fig. 7.42. Reduction of aerodynamic forces to the center of gravity of the rocket.

There exists such a point on the rocket axis, at which by centering the system of aerodynamic forces on it, we obtain a moment equal to zero. This point is called the center of pressure.

The center of pressure can be considered as a point of crossing of resultant aerodynamic forces with the rocket axis (Fig. 7.43). The position of the center of pressure depends exclusively on the external aerodynamics of the rocket.

Mutual location of center of pressure and center of gravity is essential for stabilization of a rocket in flight.

If the center of pressure is located behind the center of gravity, then with deflection of the rocket axis from the direction of flight the aerodynamic forces will create a moment decreasing the angle of incidence and restoring the initial direction of the axis. If the center of pressure is ahead of the center of gravity, then aerodynamic forces with deflection of the axis from the direction of flight will create a moment deflecting the axis sideways, i.e. the static moment will be destabilizing. In this case, so to speak, rocket is statically unstable. In order to displace the center of pressure back, the rocket is supplied by stabilizers. An unstabilized rocket, as a rule, is statically unstable.

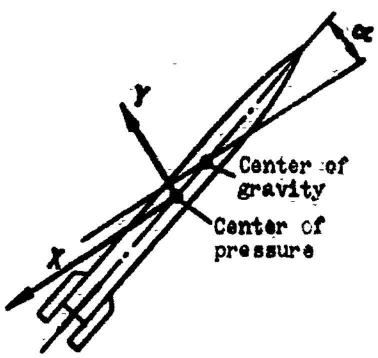

Fig. 7.43. Reduction of aerodynamic forces to the center of pressure of a rocket.

On Fig. 7.44 the dependence of the position of the center of pressure is shown for a ballistic long-range rocket on the flight speed (on the Mach number). This graph, however, is not a good indicator, since it does not allow the concept of aerodynamic stabilization of the rocket. It is more preferable to create a dependence of the position of center of gravity and center of pressure jointly in the function of the time of rocket flight. On Fig. 7.45 such a dependence for the same rocket is shown.

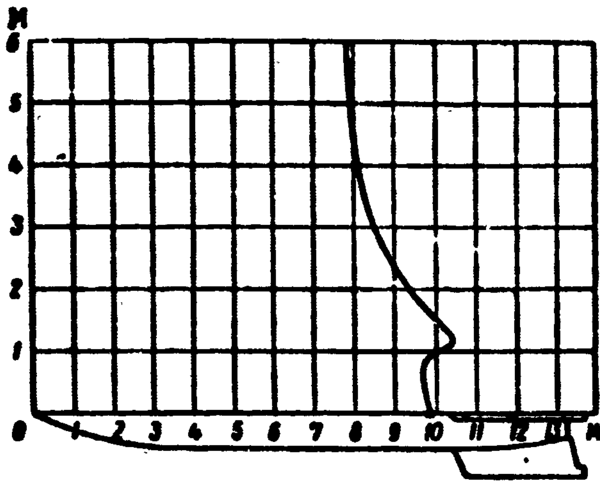

Fig. 7.44. Dependence of the position of center of pressure of the rocket on the Mach number of flight.

From the given curves it is clear that for the given rocket during all controlled flights the center of pressure remains constantly behind the center of gravity of the rocket. Thus, this rocket in all sections of controlled flight is statically stabilized.

Aerodynamic stabilization of the rocket is estimated by the degree of stabilization or, so to speak, by the reserve of static stability — by the relation of the distance between the center of pressure and center of gravity to the general rocket length. This magnitude for rockets with stabilizers oscillates from 5 to 15%.

It is necessary, however, to note that aerodynamic stabilization cannot be useful in flight outside the atmosphere. Stabilizers preserve their value only for atmospheric sections of the rocket trajectory and for rockets with a separating frontal section can be thrown-out all together. Here flight stability is ensured by the functioning of the control system.

In construction of a graph similar to the one shown on Fig. 7.45, the position of center of gravity is determined by simple calculations

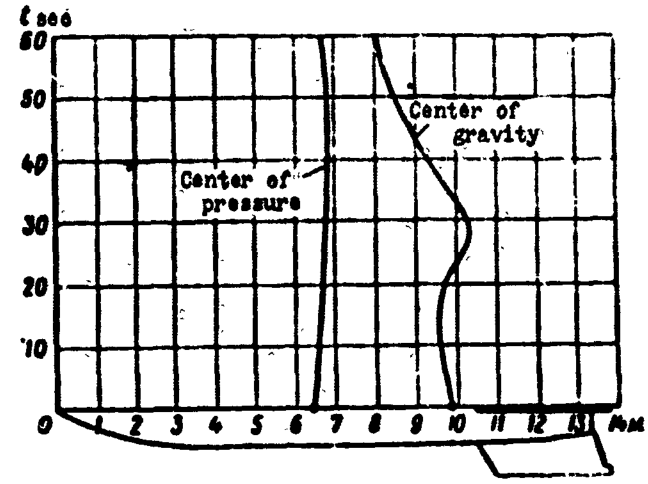

Fig. 7.45. Dependence of the position of center of gravity and center of pressure of a rocket on the time of flight.

considering the magnitude of weight and location of remaining fuel in the rocket at a given moment.

The position of center of pressure is determined by means of approximate calculations and with the help of tests on models in wind tunnels, was described above.

Damping Moment

Damping moments are those whose magnitudes depend on the angular velocity of the rocket's rotation about a certain axis. It is possible to consider damping moments about the longitudinal and the two transverse axes of the rocket. Damping moments act in a direction opposite rotation, and in first approximation are proportional to angular velocity.

The appearance and magnitude of damping moments depend on conditions of streamlining of a rocket by the external flow and on conditions of flow of liquids and gases inside the rocket and engine. In accordance with this we distinguish aerodynamic external and internal damping moments.

The external damping moment M_{da1} is the evident result of simple resistance of air to rotation of the rocket.

During rotation of the rocket relative to its transverse axis passing through the center of gravity, streaming of every element of the lateral surface by the flow will occur with a change of local angle of incidence. If rotation occurs with an angular velocity ω, then this change of angle of incidence for an element of surface located at a distance x from a certain reference point for cross sections of the rocket lengthwise to its axis,* will be (Fig. 7.46)

$$\Delta\alpha_M = \frac{\omega(x - x_c)}{v},$$

where x_c — axial coordinate of center of gravity.

As a result of such change in the angle of incidence a local additional aerodynamic moment appears, acting in a direction opposite rotation. The damping moment is determined by summation of elementary moments along the rocket surface.

An estimated expression for the damping moment can be derived by analogy with those expressions derived earlier for aerodynamic forces X and Y and moment M_{st}. The total damping moment will be considered proportional to impact pressure $\rho v^2/2$, characteristic area S and to relationship ω/v. Introducing also from the considerations the dimension of the square of the rocket length l, we get

$$M_{д1} = c_{д} \frac{\rho v^2}{2} \frac{\omega}{v} S l^2. \qquad (7.19)$$

*As the point of reference of cross sections along the longitudinal axis of rocket, it is possible to take the so-called theoretical summit — the point of crossing of the ogival outline with the rocket axis.

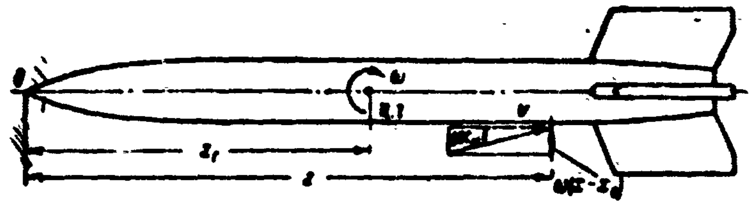

Fig. 7.46. Appearance of a local additional angle of incidence during rotation of the rocket. [Ц.Т. = Center of gravity]

The dimensionless coefficient of damping moment c_{da}, as with all aerodynamic coefficients encountered up till now at low speeds does not depend on the flight speed. At high speeds this magnitude is considered as a function of the Mach number. On Fig. 7.47 is shown an example of a graph on the dependence of coefficient c_{da} on the Mach number for a V-2 rocket during its rotation about the transverse axis.

<u>Internal damping moment</u> M_{da2} is caused by the presence of Coriolis acceleration, which appears during rotation, together with the rocket, of fluid flow along tanks and pipelines of the rocket, and flow of gases through the combustion chamber and nozzle of the engine. This moment is easy to determine if one were to consider that these flows follow the turning body of the rocket completely.

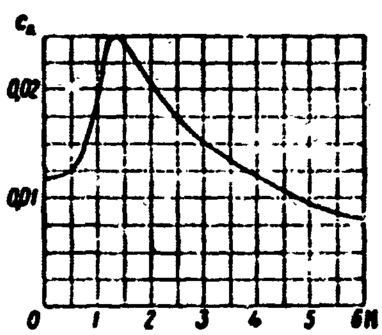

Fig. 7.47. Dependence of the coefficient of damping moment on the Mach number during rotation of the rocket about the transverse axis.

At a distance $x-x_c$ from the center of gravity of the rocket (Fig. 7.48) we will separate an element of flow with length dx and with cross section S_x and, consequently, with mass $\rho S_x dx$.

Coriolis acceleration, as it is known, is determined by the vector product

$$2[w v],$$

where w — flow speed in rocket;

ω — angular velocity of rotation of rocket about the transverse axis.

The elementary moment of the Coriolis force is

$$dM = 2w\omega\rho S_x (x - x_c)\, dx.$$

The damping moment is determined by means of integration of this expression x:

$$M'_{z2} = 2\omega \int_{x_2}^{l} \rho w S_x (x - x_c)\, dx.$$

But from the condition of constancy of flow + rate in the interval of sections x_2 and l

$$\rho w S_x = m = \text{const},$$

therefore

$$M'_{z2} = 2\omega m \int_{x_2}^{l} (x - x_c)\, dx = \omega m\, [(l - x_c)^2 - (x_2 - x_c)^2].$$

The limits of integration of this expression x_2 and l for liquid-fuel rocket have to be determined from the surface of the fuel component in the rear tank to the nozzle section (see Fig. 7.48). On this interval the flow + rate of both components in sum is equal to m. To the obtained expression it is necessary to add the same, in which the total flow + rate m is replaced by the mass flow rate of the component in the front tank m_1, and integration is conducted from x_1 to x_2 (see Fig. 7.48), i.e. the surface of the component in the front tank to the surface of the component in the rear tank. In result we get

$$M'_{z2} = \omega\, (m\, [(l - x_c)^2 - (x_2 - x_c)^2] + m_1\, [(x_2 - x_c)^2 - (x_1 - x_c)^2]).$$

Part of the mass moves in a direction to the center of gravity, and not away from it. Due to this the damping moment will be partial!.

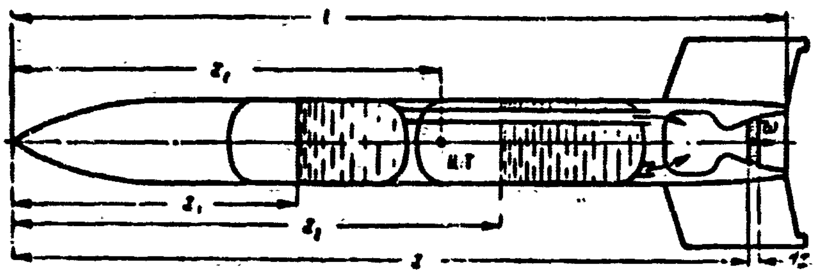

Fig. 7.48. Determination of damping moment, created by liquid flow and gases when the engine is working. [Ц.Т. = Center of gravity]

decreased.

To the obtained expression may be also added the proportional angular velocity $\omega = \dot\varphi$ moment $\dot J\dot\varphi$ separated above from inertial (see p. 291).

Speed of change of the moment of inertia of the rocket $\dot J$ is determined from simple considerations.

During the time Δt, the distribution of masses in the rocket will be changed only near the surface of the first and second components of fuel (see Fig. 7.48). Consequently,

$$\Delta J = [-m_1(x_1 - x_c)^2 - m_2(x_2 - x_c)^2]\Delta t$$

and

$$\dot J = -m_1(x_1 - x_c)^2 - m_2(x_2 - x_c)^2,$$

where m_1 and m_2 – secondal flow + rates of first and second components.

Adding $\dot J\omega$ to the expression of moment obtained above and considering that $m = m_1 + m_2$, we get

$$M_{д2} = \omega\{m_1[(l - x_c)^2 - 2(x_1 - x_c)^2)] + m_2[(l - x_c)^2 - -2(x_2 - x_c)^2]\}. \quad (7.16)$$

The presence of a jet stream passing from the engine leads not only to an internal damping moment. Due to suction and change of conditions in formation of the boundary layer, change of the external damping moment occurs. In particular, measurements show that in the launching phase, when the rocket still has a low speed, only at the expense of ejection of flow for the rocket, supplied by stabilizers, a damping moment appears exceeding the internal gas-dynamic moment by two-three times.*

Comparison of stabilizing and damping moments by numerical values has meaning only for the known relationship between the angle of incidence and the angular velocity of the rocket's rotation. Thus, for instance, for a V-2 rocket during oscillatory motion in the process of stabilization in dense layers of atmosphere the maximum value of the damping moment is approximately 10% of the biggest value of stabilizing, where of this 10% a large part is apportioned to the damping moment from external aerodynamic forces and only a small part to the internal moment, determined by expression (7.16). For an unstabilized rocket the static moment of aerodynamic forces is, as a rule, destabilizing. In this case the comparative numerical appraisal of moments will be completely different.

The internal gas-dynamic moment obtains an independent value during flight of a rocket outside the atmosphere, and also in appraisal of moments appearing during control of the rocket by turning chambers.

5. Control Forces

As the basic performing control organs we will consider at first the jet stream steering control.

Gas-dynamic forces of flow in the form of a frontal force $X_{j.s}$,

*M. K. Yesiyev, "Concerning the question of determination of the gas-dynamic damping moment." Collection of works of the Moscow Higher Technical School "Certain questions of mechanics." Defense Publishing House, 1957.

lift $Y_{j.s}$, and moment $M_{j.s}$ act on the jet stream steering control located in the stream of the engine (Fig. 7.49).

The steering control in the gas stream acts as an air vane of the aircraft in the flow of air.

Frontal and lift forces of the steering control, and also moment $M_{j.s}$ can be defined by the usual aerodynamic formulas:

$$X_{r.p}=c_{x\,r.p}\frac{\rho w^2}{2}S;$$
$$Y_{r.p}=c_{y\,r.p}\frac{\rho w^2}{2}S;$$
$$M_{r.p}=c_{m\,r.p}\frac{\rho w^2}{2}Sl,$$

[r.p = j.s = jet stream]

where w — speed of gas flow along the steering control;

ρ — density of gas in the flow;

S — characteristic area of steering control;

l — characteristic length of steering control.

Parameters of gas flow ρ and w practically remain constant in the flight trajectory of the rocket.

Coefficients $c_{x\ j.s}$, $c_{y\ j.s}$, and $c_{m\ j.s}$, as with all aerodynamic coefficinets, depend on the shape of steering controls and angle of incidence, i.e. the angle of rotation of the steering controls δ in the flow (see Fig. 7.49). It is necessary to say that this angle changes in wide enough limits. Thus, for instance, for ballistic long-range rockets, jet stream steering controls can turn in the flow within the limits of $\pm 25^{\circ}$; the operational turns of the steering control occur in an interval of $\pm 15^{\circ}$. The result is that forces acting

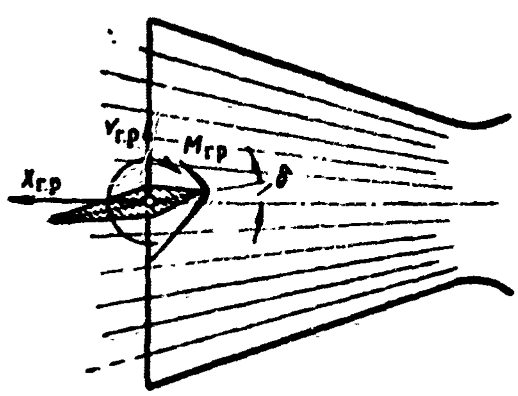

Fig. 7.49. Jet stream steering control in the stream of the engine.

on the steering control in a phase of controlled rocket flight change in rather wide limits. The typical dependence of coefficients $c_{x\ j.s}$, $c_{y\ j.s}$, and $c_{m\ j.s}$ on the angle of rotation of the steering control δ is represented on Fig. 7.50.

It is necessary to note that aerodynamic coefficients of jet stream steering mechanisms will change also because of possible burning of the steering controls in the flow of hot gases.

The magnitude of the operational angle δ in flight can be considered as consisting of two components. The first component is caused by the necessity of program rotation of the rocket on the trajectory; it is determined in dependence upon the form of the trajectory in ideal conditions of undisturbed flight. This angle of steering control displacement may be calculated beforehand. For a given trajectory the necessary control forces are calculated, and along it — also the unknown angles.

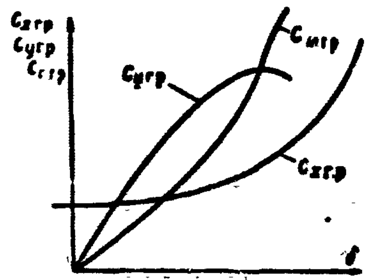

Fig. 7.50. Dependence of coefficients $c_{x\ j.s}$, $c_{y\ j.s}$, and $c_{m\ j.s}$ on the angle of rotation of the steering control δ.

The second component is an angle appearing as a reaction to stabilization, and consequently also to the steering controls on accidental perturbations. Here are included both factors constantly acting on the trajectory (such as uncoaxiality of the tractive force, or disturbance of axial symmetry of the aerodynamic form of the rocket, and so forth), and briefly acting perturbations (for instance, gusts of wind). Their magnitude cannot be determined beforehand exactly. It is determined on the basis of a system of allowances during manufacture of the rocket and also statistical data on observations of the state of the atmosphere.

As a magnitude of possible perturbations additional interval working movement of steering controls is designated beyond those determined from conditions of programmed flight.

The first of the two mentioned components of the working angle δ is basic and pertains to systematically acting control forces $X_{j.s}$ and $Y_{j.s}$ considered in ballistic calculations.

Regarding the second component of angle δ, it pertains to accidental, unsystematically acting forces, not considered in ballistic calculations.

Let us consider forces $X_{j.s}$, $Y_{j.s}$, and moment $M_{j.s}$ separately.

The total force of frontal resistance of the steering controls $X_{j.s}$ is not the directing force and is considered as a loss of thrust on the controls. This loss is quite significant. Thus, for instance, for a V-2 rocket at $\delta = 0$ it is approximately 640 kg (160 kg on each steering control) and increases due of turning of the steering controls.

Burning out of graphite steering controls during the time of work of the engine somewhat lowers magnitude $X_{j.s}$.

Moment $M_{j.s}$ is the sum of static and damping moments, acting on the steering control and proportional correspondingly with the angle of rotation of the steering control δ and angular velocity $\dot{\delta}$. If to this moment the inertial moment of the steering control $J_{j.s}\ddot{\delta}$ were also added, we get the magnitude of the hinge moment M_h (see p. 289):

$$M_h = a_0\delta + a_1\dot{\delta} + a_2\ddot{\delta}, \tag{7.17}$$

where a_0, a_1, a_2 are proportionality factors.

For graphite steering controls these coefficients during the time of the engine's operation somewhat change due to burning out of graphite from the surface. Coefficient a_0 changes the most essentially, since burning out of graphite leads to a change in position of the center of pressure. Burning out occurs basically from the front, the most intensely blown edge (Fig. 7.51a). As a result, the center of pressure for the steering control is noticeably displaced back (coefficient a_0 increases). To avoid this the graphite steering wheel is given the form shown on Fig. 7.51b. Here burning out occurs from both sides of the axis of rotation of the steering control; therefore, the center of pressure is displaced much less.

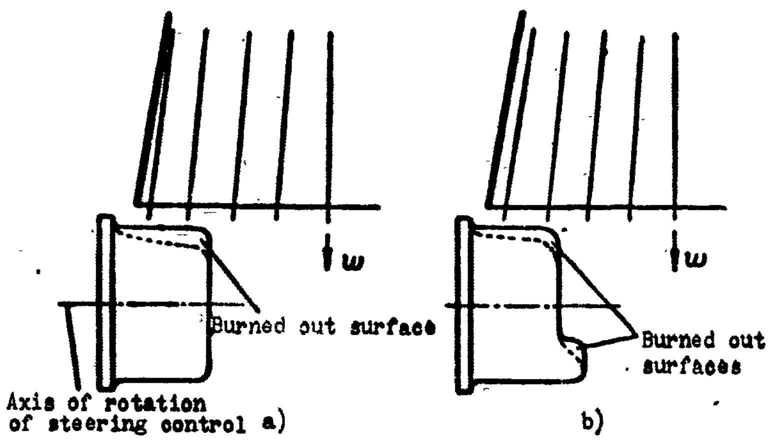

Fig. 7.51. Change of form of the graphite steering control during the engine's operation.

The hinge moment M_h entering into equation (7.4) is much smaller than magnitude $Y_{con}c$; therefore it is possible to disregard it in the motion equation. The magnitude of hinge moment is important only for appraisal of operational conditions of steering machines, turning the jet stream steering controls.

The lift force of jet stream stearing controls $Y_{j.s}$ creates a basic control moment $Y_{con}c$ turning the rocket in the needed direction [see equation (7.4)]. The lift force itself $Y_{j.s}$ in equation (7.3), utilized for ballistic calculations, is expressed usually through lift force Y from conditions of static equilibrium of aerodynamic and directing moments.

Considering that angular acceleration of a programmed turn $\ddot{\vartheta}$ is small, and disregarding magnitude M_h, from equation (7.4) we get

$$M_a + Y_{\text{упр}} c = 0.$$

Damping moments during programmed turns also can be disregarded. Then for small angles of attack, moment

$$M_a = Ye,$$

where e is the distance from the center of pressure to the center of gravity of the rocket. Therefore,

$$Y_{\text{упр}} = Y\frac{e}{c}.$$

Thus, equation (7.3) becomes

$$\dot{\theta} = \frac{1}{Mv}\left[(P - X_{\text{упр}})\alpha + Y\left(1 - \frac{e}{c}\right)\right] - \frac{g}{v}\cos\theta. \qquad (7.18)$$

Control by the rocket can be carried out not only with the help of jet stream steering controls, but also by means of turning the engine chamber (Fig. 7.52). In this case loss of thrust X_{con} will have a magnitude

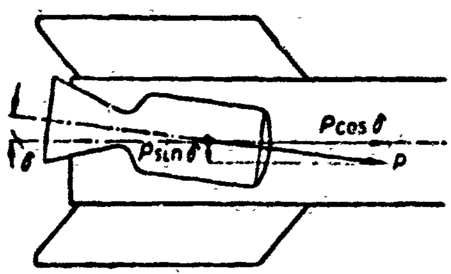

Fig. 7.52. Appearance of control force during turning of engine chamber.

$$X_{\text{упр}} = P(1 - \cos\delta),$$

where P is the tractive force of turning of the engine.

Force Y_{con} is determined by the following expression:

$$Y_{\text{упр}} = P\sin\delta.$$

In distinction from jet stream steering controls, the hinge moment for a turning chamber does not have a component proportional to the angle of rotation δ, i.e. coefficient a_0 in expression (7.17) becomes zero, and then we get

$$M_{\alpha}=a_1\dot{\delta}+a_2\ddot{\delta}.$$

Component $a_1\dot{\delta}$ is the sum of damping moments from the passing flow and ejected external flow of air (see p. 348). Coefficient a_2 is the moment of inertia of the chamber with respect to the axis of rotation.

6. Change of Tractive Force and Weight of Rocket on Trajectory

Of all forces entering into the motion equations (7.2) and (7.3), we have only two remaining unconsidered: tractive force of the engine and rocket weight.

Let us note forst of all that the tractive force during flight will change in connection with change of ambient pressure. Another cause of thrust change in flight may be flow rate change.

The dependence of tractive force on atmospheric pressure p_h and flow + rate m is determined by expression

$$P=mw_e-S_a p_h, \tag{7.19}$$

where

$$w_e=w_a+\frac{S_a p_a}{m},$$

where, as it is known, pressure p_h and flow + rate m (in the limited range of its change) do not render an influence on the magnitude of effective speed w_e (see Chapter VI).

During ascent of the rocket, atmospheric pressure changes and together with it the axial component of external static pressure, which is the component of the tractive force. Thrust due to ascent of the rocket increases in accordance with drop of atmospheric pressure, as this follows from expression (7.19).

For ballistic long-range rockets the increase of thrust with decrease of atmospheric pressure to zero can approach 20-25%.

Expression (7.19) shows also that thrust depends on flow + rate linearly in the case where deflections of flow + rate from certain nominal values are small. At sharp changes of flow + rate this occurs, for instance, during operation of the engine at different stages of thrust, no longer allowed not to be considered with the change of effective speed w_e. A significant decrease of flow + rate or, as we say, deep choking of the motor is always accompanied by a certain lowering of specific thrust (or what is the very same, effective exhaust velocity).

We know that here the temperature drops somewhat in the chamber due to great reduction of pressure and growth of losses by dissociation of combustion products (see Chapter V).

During flight of the rocket the flow rate will be continuously changed in connection with change of conditions under which operation of the propulsion system progresses in flight, and with the change of these conditions from nominal.

Thus, for solid fuel rocket engines the flow + rate depends on the burning rate of the charge and area of burning surface. In turn, the temperature of the charge and pressure in the chamber render an influence on the burning rate. Obviously, causes of the change of flow + rate in flight will be increased temperatures of the charge due to its heating, change of the burning surface and free volume of the chamber due to burning of the charge and, finally, accompanying these processes a change of chamber pressure, affecting the burning rate.

It is ncessary to note, however that for contemporary solid fuel engines all these causes do not lead to great relative changes of flow + rate in flight. Special measures are taken, ensuring stability of the burning process by means of corresponding selection both of the form of the charge and the composition of solid fuel.

For liquid-fuel rockets the cause of change of flow + rate in flight is the inconstancy of operational conditions of the supply system. Change of feed pressure occurs mainly due to variation of hydrostatic pressure in the fuel's liquid components disposed in tanks and main lines. These pressures change in sufficiently wide limits. On the one hand, due to emptying of tanks the height of the liquid column decreases creating hydrostatic pressure. On the other hand, the influence of inertial forces during progressively increasing accelerations contributes to increase of hydrostatic pressure (see Chapter VIII).

For engines with a pump supply system the pressure of fuel components on entrance into the turbopump unit is composed of feed pressure above the liquid surface in the tanks and hydrostatic pressure created by the given component. Change of inlet pressure in a turbopump unit will evoke approximately the same pressure change of the component on entrance into the combustion chamber, inasmuch as a centrifugal pump creates an approximately constant drop of pressures.

Obviously, any change in pressure conditions of tank feed will also be reflected on the magnitude of feed pressure. This totaly pertains also to engines with a displacive supply system.

Thus, engine thrust is a composite function of feed pressure for each of the components. At change of supply pressure both the total flow + rate and the relationship of fuel components change.

Change of supply conditions in flight does not have to lead to significant quantitative changes of flow + rate and the relationship of components.

For ballistic rockets the flow rate in flight usually increases somewhat. An increase of thrust here does not exceed 1-2%.

Speaking of the tractive force, it is impossible not to mention its change in transition stages at starting and turning off of the engine.

Ignition of the engine is usually produced at a greatly lowered supply of components in conditions of the so-called preliminary stage. The nominal flow + rate at starting of the engine is not established at once, but upon the expiration of a certain time. When increasing thrust is equal to the weight of the rocket starts to exceed it, the rocket leaves the launching pad. At this instant, so to speak, the rocket is launched.

At turning off of a liquid fuel engine, thrust also disappears not instantly, and the phenomenon of after-effect is observed. After turning off of the engine, due to burning of a definite quantity of fuel a certain tractive force continues to be created. The magnitude of this thrust turns out to be indefinite enough, leading to perceptible straggling by distance for ballistic long-range rockets. To decrease it, before final turning off, the engine is shifted to the final stage with a decreased fuel feed. On Fig. 7.53 a typical graph is shown of change of engine thrust of a ballistic long-range rocket.

Also the weight of rocket is changed in trajectory. This occurs due to change of mass M, also due to change of acceleration due to gravity g_h.

The rocket mass in a given moment of flight is equal to the initial mass M_0 after subtracting the mass of burned fuel:

$$M = M_0 - \int_0^t m\,dt.$$

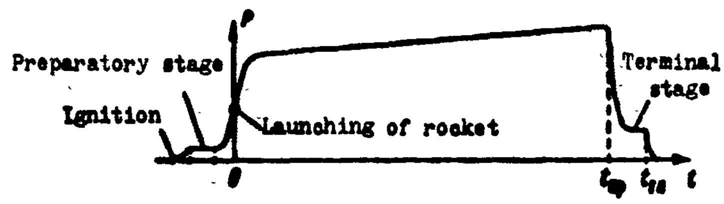

Fig. 7.53. Change of tractive force of engine depending upon time.

If the flow + rate remains constant, then

$$M = M_0 - mt.$$

Acceleration due to gravity to an altitude h above the surface of the Earth, in accordance with expression (2.2), will be

$$g_h = \frac{k^2}{(R+h)^2},$$

where k^2 — gravitation constant of Earth;

R — average radius of Earth.

The vector of acceleration due to gravity at any point in the trajectory has a direction toward the center of Earth. Therefore, for trajectories of great length it is necessary to consider the change of direction of the acceleration vector due to gravity about the origin of the coordinates system.

CHAPTER VIII

FLIGHT TRAJECTORY OF BALLISTIC ROCKETS

1. Sections of Trajectory

The rocket trajectory from the moment of launching and to the moment of drop or explosion in flight is divided usually into two sections, the section of active flight, i.e., flight with a working engine, and the section of free flight, when the engine does not work (Fig. 8.1). Thus, from the point of view of flight dynamics the difference between sections of trajectory consists in the fact that from the number of forces acting on the rocket, in the period of its free flight tractive force is excluded.

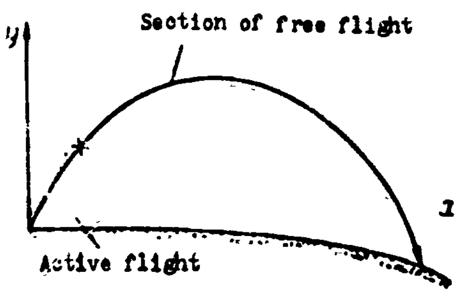

Fig. 8.1. Active section of free flight of rocket trajectory.

The powered-flight trajectory may be in turn subdivided into smaller sections.

In those cases when the rocket is started from directrixes, motion in directrixes is considered separately. This section is peculiar mainly to unguided rockets in particular, solid-propellant rockets of local action. During motion on directrixes the forces communicated by the controlling devices of the launcher are added to external forces.

For controlled zenith rockets the active section may, depending upon methods of control, be divided into sections of free uncontrolled and controlled flight, but in the case where the rocket has a homing guidance system, also into the section of homing guidance.

A ballistic long-range rocket starts from the launching pad vertically and during the first several seconds moves directly upwards. This section of flight may be called the starting section. Next, programmed turning of the rocket on the trajectory begins. The rocket deviates from the vertical and, making an arc on the section of removal, emerges into the last inclined section (section of cut-off), in which occurs the cut-off of the engine upon achievement of a speed by the rocket, necessary for reaching the given distance. After this starts the section of free flight, and the rocket flies as a freely catapulted body usually without definite orientation, as this is shown on Fig. 8.2.

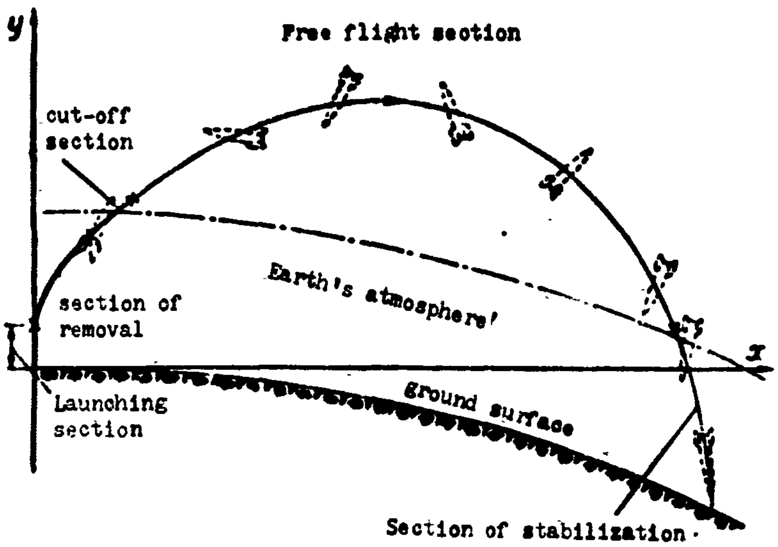

Fig. 8.2. Trajectory of ballistic long-range rocket.

The trajectory of a ballistic long-range rocket can be divided into sections of atmospheric and outer space flight. For atmospheric, flight is considered in so rarefied layers of atmosphere that the aerodynamic forces turn out to be insignificantly small as compared to the force of weight of the rocket. Since the atmosphere does not have a sharp upper boundary, then a sharp division between these two sections does not exist. Furthermore, the start of outer space flight is determined not only by altitude, but also by speed of flight, inasmuch as the magnitude of aerodynamic forces depends on the latter.

Depending upon the distance of firing, engine cut-off can occur both in the end of atmospheric and also in outer space flight. In the atmospheric section of free flight strong braking occurs in the descending part of the trajectory — the rocket speed noticeably decreases. In this section the rocket is stabilized and approaches the target with nose forward (see Fig. 8.2).

Summarizing everything said above, it is necessary to note that division of the trajectory into sections, by the criterion of presence of any forces determining rocket motion and in significant measure determining the approach to integration of differential equations of motion, is conditional.

Let us return now to motion equations (7.2), (7.3), and (7.4). We will write them considering that equation (7.3) is converted into (7.18) and that in equation (7.4) the magnitude M_h may be rejected, and moment M_a replaced by the sum $M_{st} + M_{da}$:

$$\dot{v}=\frac{P-X_{упр}-X}{M}-g\sin\theta; \tag{8.1}$$

$$\dot{\theta}=\frac{1}{Mv}\left[(P-X_{упр})\alpha+Y\left(1-\frac{\varepsilon}{\varepsilon}\right)\right]-\frac{g}{v}\cos\theta; \tag{8.2}$$

$$J\ddot{\varphi}+M_{ст}+M_{д}+Y_{упр}c=0. \tag{8.3}$$

[упр = con = control; ст = st = static;
д = da = damping]

This system of differential equations, as was already indicated, describes planar motion of the rocket. Equations (8.1) and (8.2) determine the motion of the rocket's center of gravity, and equation (8.3) gives the law of rocket rotation about its center of gravity in the plane of trajectory.

The trajectory and parameters of motion on trajectories obtained as a result of integration of motion equations, are called undisturbed.

In real conditions of flight, the rocket is subjected to the influence of accidental external forces, which cannot be considered in equations of motion (for instance, gusts of wind, misalignment of line of thrust action, and so forth). Accidental forces render a perturbing influence on rocket motion. Basically, this concerns rotation of the rocket about its center of gravity. For a guided rocket this motion has a clearly expressed oscillatory character.

It is possible to suppose that during flight the rocket participates simultaneously in two superimposed motions; in the basic undisturbed motion, accomplished under action of forces considered in equations of motion, and in oscillatory motion, caused by external accidental forces and by corresponding reactions on the part of controls.

The above is illustrated by the graph on Fig. 8.3, where the law of change of pitch angle as a function of time is represented.

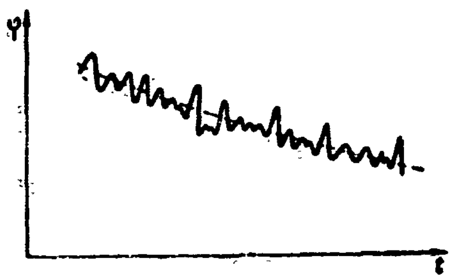

Fig. 8.3 Character of change of pitch angle in flight as a function of time.

From the given curve it is clear that a guided rocket in flight has oscillatory motion and at the same time there is a slow change of the mean value of angle φ. Analogous curves can be represented also for other angles, characterizing space orientation of the rocket.

During normal functioning of the control system, angular deflections of the rocket in perturbed motion are small and do not affect the motion of the rocket's center of gravity. Therefore, these motions can be considered independent or in any case weakly connected.

Undisturbed motion, caused by known forces is studied in ballistics. Forces and moments, entering into equations (8.1), (8.2) and (8.3) are considered here as variable with respect to predetermined beforehand and above described laws. For undisturbed rotation in ballistic calculations programmed turning of the rocket is used. Oscillatory motion under the action of accidental forces pertains to the category of phenomena studied in the theory of stability of motion. These questions will be discussed in Chapter IX. We will turn to consideration of the ballistics of long-range rockets.

2. Active Section

The trajectory of the rocket's center of gravity in the active section can be obtained as a result of joint integration of equations (8.1) and (8.2). Let us remember that the final form of equation (8.2) is obtained from the condition of static equilibrium of aerodynamic and control moments. Such an assumption does not lead practically to alteration of the motion of the center of gravity, inasmuch as during programmed turning of rocket, inertial moment $J\ddot{\varphi}$ and damping moments are very small. More exact necessary values of control forces and laws of their change in flight can be obtained as a result of integrating equation (8.3).

Analytic integration of differential equations of motion in the common form is not possible. Therefore, these equations are integrated numerically in reference to some concrete types of motion.

So that the rocket can attain the given target, at the end of the active section it is necessary to transmit a definite velocity and direction of flight to it at definite coordinates at the end of the active section. This is ensured first of all by programming of removal.

The rocket, as was said above, starts vertically. Then its axis deviates from the vertical. This deflection occurs gradually and is ceased shortly before engine cut-off. Turning of rocket is carried out by controls, to which a signal is sent from the programming device. The rocket turns about a motionless horizontal, fixed by a gyroscope, by the given law in the function of time. The pitch angle, assigned by the programming mechanism of the gyroscopic system, is called the <u>program pitch angle</u> φ_{pro}, and the law of its change in time – the <u>program</u> of rocket <u>pitch</u>:

$$\varphi_{пр} = f(t).$$

[пр = pro = program]

In the simplest case of calculation of ballistic long-range rocket trajectories, it is possible to integrate the allignment of motion without using the equation of control for the pitch angle (see p. 293). Here the error between the programmed value of the pitch angle, assigned by the gyroscopic system φ_{pro} and the real value of the pitch angle φ in a given moment of programmed turning of the rocket will not be considered. This error serves as the necessary signal for turning of controls for the purpose of realization of programmed flight.

The angle of error for ballistic rockets is small, therefore, such an assumption does not lead to a great error if motion of the center of gravity of the rocket is considered.

Allowing the absence of the angle of pitch error, we thereby consider the control system of the rocket ideal, i.e., ensuring at every moment of flight time a value of the pitch angle equal to φ_{pro} known to us even before calculation of the trajectory:

$$\varphi_{пр} = \varphi = \theta + \alpha. \qquad (8.4)$$

An example of the program of a pitch angle of ballistic long-range rocket is shown on Fig. 8.4.

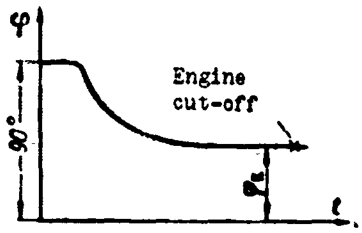

Fig. 8.4. Form of program of ballistic rocket.
[к = s]

Program angle $\varphi_{pro.s}$ at the cut-off section should be kept constant. Then engine cut-off occurs at minimum magnitudes of angular perturbations, which lowers the magnitude of dispersion. The numberical value of angle $\varphi_{pro.s}$ is changed depending upon flying range. At comparatively small distances (100-300 km) this angle is near to 45°. At great distances angle $\varphi_{pro.s}$ essentially decreases.

Selection of the program of pitch angle is one of the important problems of ballistic designing. Obviously, for every concrete construction of a rocket there exists its own extreme program, ensuring maximum range of rocket flight.

For rockets flying a great distance, questions of dispersion have special value in selection of the program of pitch angle. The degree of effect of perturbing influences on the trajectory of free flight, at given tolerances on scattering of parameters of the rocket, turns out to be essentially dependent on the type of program. Therefore, for long-range rockets the program is chosen not only by conditions of maximum range, but also primarily by the requirements of minimum dispersion. Requirements for the program also change depending upon the method of the rocket's range control.

The program renders a decisive influence on lateral loads, which the rocket body perceives in the active section of flight. It is clear that not one optimum-from the ballistic point of view-program can be used, if it causes impermissibly large transverse forces.

In selection of the program of pitch angle for ballistic rockets, a number of other requirements is also considered, concerning conditions of start, flight stability, possibility of reaching any distance in the given range at a constant program section of removal, and so forth.

Practically, the selection of the program of pitch angle is carried out simultaneously with selection of all remaining parameters for the rocket being designed. As was already said, electronic computers are widely applied in these calculations.

Let us return to motion equations (8.1) and (8.2). Let us assume that the program of change of pitch angle is given, and control by the rocket is ideal.

Let us consider how in the simplest case numerical integration of motion equations is produced. We will copy these equations in the following form:

$$\Delta v = \left(\frac{P - X_{\text{упр}} - X}{M} - g\sin\theta\right)\Delta t; \tag{8.5}$$

$$\Delta\theta = \left\{\frac{1}{Mv}\left[(P - X_{\text{упр}})\alpha + Y\left(1 - \frac{\varepsilon}{c}\right)\right] - \frac{g}{v}\cos\theta\right\}\Delta t. \tag{8.6}$$

For determination of coordinates of the moving rocket, to these equations two more evident are joined:

$$\left.\begin{aligned}\Delta x &= v\cos\theta\,\Delta t;\\ \Delta y &= v\sin\theta\,\Delta t,\end{aligned}\right\} \tag{8.7}$$

where Δx and Δy – shift of the rocket's center of gravity along axes x and y during the time Δt (Fig. 8.5). The present coordinates x and y determine the rocket position on the trajectory at any moment of time.

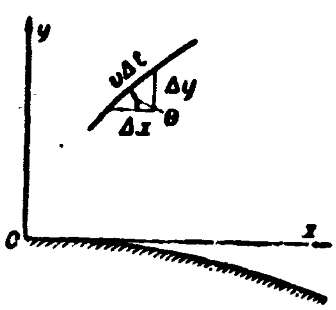

Fig. 8.5. Calculation of rocket trajectory.

Equations (8.5), (8.6) and (8.7) jointly with the program equation (8.4) allow us to find, depending upon flight time t, the value of five unknown functions: speed v, angle of inclination of speed vector θ, coordinates x and y, angle of incidence α.

In integration consequently the numerical value of the right members of equations (8.5)-(8.7) are found for a certain number of equidistant moments of time $t_{(0)}, t_{(1)}, t_{(2)}, \ldots, t_{(i)}, t_{(i+1)}$.

The difference

$$t_{(i+1)} - t_{(i)} = \Delta t$$

is called the integration increment (for instance, $\Delta t = 1$ sec). On the selection of the integration increment depends the accuracy of calculation. The smaller the increment, the greater the accuracy of integration, but even greater will be the quantity of calculations.

In the initial moment of motion $t_{(0)} = 0$ the value of parameters of motion will be the following:

$$v_{(0)} = 0,$$
$$\theta_{(0)} = 90^{\circ},$$
$$x_{(0)} = 0,$$
$$y_{(0)} = 0,$$
$$\alpha_{(0)} = 0.$$

The method of numerical integration allows us to calculate the parameters of motion in moment $t_{(i+1)}$, if in moment $t_{(i)}$ these parameters ($v_{(i)}, \theta_{(i)}, x_{(i)}, y_{(i)}, \alpha_{(i)}$) are known to us.

On the basis of equation (8.5), speed in moment $t_{(i+1)}$ will be equal to

$$v_{(i+1)} = v_{(i)} + \Delta v_{(i)} = v_{(i)} + \left(\frac{P_{(i)} - X_{ynp} - X_{(i)}}{M_{(i)}} - g_{(i)} \sin \theta_{(i)}\right) \Delta t.$$

By the table of standard atmosphere we determine air density

$\rho_{(i)}$ and pressure $p_{(i)}$, corresponding to the known altitude of flight $h_{(i)} \approx y_{(i)}$ in moment $t_{(i)}$.

Then, the tractive force at this instant is

$$P_{(i)} = m_{(i)} w_e - S_a p_{(i)},$$

and the force of frontal aerodynamic drag is

$$X_{(i)} = c_{x(i)} \frac{\rho_{(i)} v_{(i)}^2}{2} S,$$

where $c_{x(i)}$ is on the graph $c_x = f(M)$, where the $M_{(i)}$ number is determined depending upon speed v_i and altitude $h_{(i)}$ (the latter must be considered since atmospheric temperature affects the speed of sound) in a given moment $t_{(i)}$.

The rocket mass at this instant is

$$M_{(i)} = M_{(i-1)} - m_{(i-1)} \Delta t.$$

Acceleration due to gravity at a given altutide is

$$g_{(i)} = \frac{k^2}{(R + h_{(i)})^2}.$$

Thus, all magnitudes in the right member of equation (8.5), corresponding to time $t_{(i)}$ are known, as a result of which we get the value of speed $v_{(i+1)}$ in the following moment $t_{(i+1)}$:

In exactly the same way, from equation (8.6) it follows that

$$\theta_{(i+1)} = \theta_{(i)} + \Delta\theta_{(i)} = \theta_{(i)} + \left\{ \frac{1}{M_{(i)} v_{(i)}} \left[(P_{(i)} - X_{yup}) \alpha_{(i)} + \right.\right.$$
$$\left.\left. + Y_{(i)} \left(1 - \frac{\varepsilon_{(i)}}{\varepsilon_{(i)}} \right) \right] - \frac{g_{(i)}}{v_{(i)}} \cos\theta_{(i)} \right\} \Delta t.$$

Lift $Y_{(i)}$ is determined depending upon the angle of incidence $\alpha_{(i)}$ and number $M_{(i)}$ for a known value of impact pressure:

$$Y_{(i)} = c_{y(i)} \frac{\rho_{(i)} v_{(i)}^2}{2} S.$$

The position of the rocket's center of gravity in a connected system of coordinates is a known function of flight time; the position of the center of pressure is determined depending upon $M_{(i)}$ and $\alpha_{(i)}$. This means that magnitudes $e_{(i)}$ and $c_{(i)}$ in moment $t_{(i)}$ are also known to us.

Finally, from equations (8.7) we have

$$x_{(i+1)} = x_{(i)} + \Delta x_{(i)} = x_{(i)} + v_{(i)} \cos \theta_{(i)} \Delta t,$$
$$y_{(i+1)} = y_{(i)} + \Delta y_{(i)} = y_{(i)} + v_{(i)} \sin \theta_{(i)} \Delta t.$$

The new value of the angle of incidence will be determined from the program equation (8.4):

$$\alpha_{(i+1)} = \varphi_{пр(i+1)} - \theta_{(i+1)},$$

where φ_{pro} is a given function of time.

In result we get the value of motion parameters

$$v_{(i+1)};\ \theta_{(i+1)};\ x_{(i+1)};\ y_{(i+1)};\ \alpha_{(i+1)}.$$

in a moment of time $t_{(i+1)}$.

Analogously it is possible to continue calculation also for subsequent increments of integration. Results of calculations are compiled in a corresponding table.

Calculations in this sequence are conducted for the whole trajectory beginning from the moment of start. Obviously, on the vertical section of the trajectory directly after start, integration occurs without use of equation (8.6), inasmuch as the trajectory remains vertical. During transition to the section of free flight, tractive force is excluded from the number of acting forces.

The described sequence of integration of motion equations gives an idea only of the principle side of the question. In practice, ballistic calculations are conducted on the basis of a well developed system of calculations, ensuring high accuracy with a possibly smaller expenditure of labor.

As a result of ballistic calculations the dependence of motion parameters (i.e., components of acceleration of the center of gravity $\dot{v}$ and $v\dot{\theta}$, magnitude and direction of speed vectors v and θ, and also coordinates x and y) on flight time is obtained. Simultaneously, for every moment of time determining the motion of the rocket's center of gravity all external forces become known by magnitude and direction.

Figure 8.6 shows the trajectory of a V-2 rocket in the active section, and on Fig. 8.7 — dependence of speed, acceleration and drag of the same rocket on time. As can be seen from the curves, drag in the beginning increases repidly, a phenomenon caused by growth of speed of the rocket and significant increase of the drag coefficient during transition through the speed of sound. Further drag decreases due to decrease of air density ρ on the trajectory.

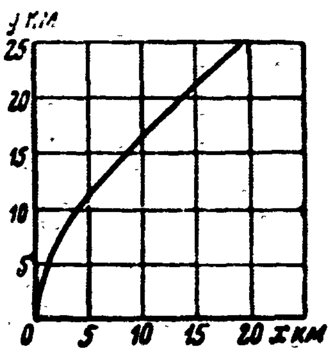

Fig. 8.6. Trajectory of a V-2 rocket in the active section.

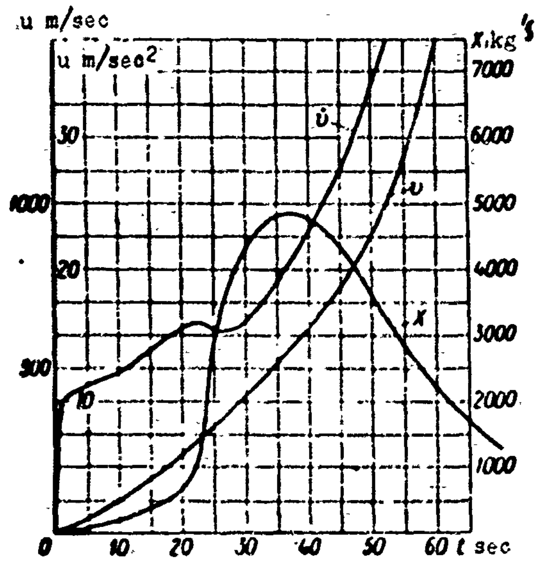

Fig. 8.7. Dependence of speed, acceleration and drag of a V-2 rocket on time on the powered-flight trajectory.

Acceleration of the rocket during flight in the powered-flight trajectory increases. This occurs because the rocket mass decreases, and tractive force is increased. Only on a small section (in a region of speeds, close to speed of sound) acceleration somewhat decreases, since here drag increases sharply.

Figure 8.8 shows the dependence of change in the incidence angle on time for a ballistic long-range rocket. This dependence is connected directly with the program of pitch angle change of the rocket.

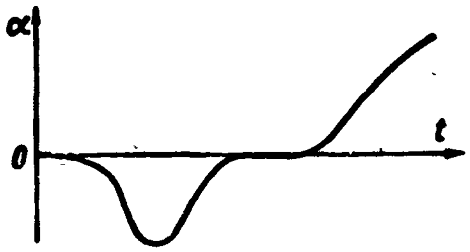

Fig. 8.8. Approximate dependence of the incidence angle of a ballistic rocket on the time of powered-flight trajectory.

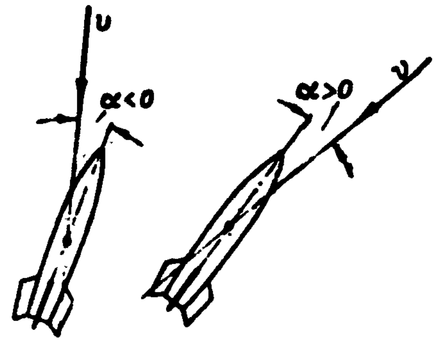

Fig. 8.9. Negative angle of incidence in the beginning and positive at the end of powered-flight trajectory.

In the beginning, while the rocket by conditions of the program moves vertically upwards, the angle of incidence is equal to zero. During deflection of the rocket from the vertical, the incidence angle due to necessity takes a negative value (Fig. 8.9). The magnitude of the incidence angle in this section depends again on the program of pitch angle. If the rocket sharply deviates from the vertical, the incidence angle will be large in absolute value. Usually the absolute value α in this section of the trajectory does not exceed 2.5-3°.

The rocket program is chosen with an intention, so that at the moment of achievement of speed of sound the incidence angle would be near to zero. It is also profitable to pass maximums of impact pressure without incidence angles; therefore, a value of α close to zero is sustained as long as the impact pressure, owing to reduction of atmospheric density, does not decrease to a sufficiently small magnitude. This is necessary for decrease of lateral aerodynamic forces and correspondingly to reduction of deflection moments acting on the rocket body.

Subsequently, the angle of incidence becomes positive, and at the end of powered flight grows rapidly, inasmuch as a constant program angle $\varphi = \theta + \alpha$ is maintained, and angle θ decreases from the action of force of gravity.

Calculation of the trajectory in powered-flight is needed not only for determination of flying range. As was already indicated, at integration of motion equations forces acting on the rocket depending upon time can be determined. This will give the possibility, on the basis of ballistic calculation, to produce also durability calculations for the rocket joints. In particular, knowing the magnitude of acceleration and distribution of axial aerodynamic resistances it is possible to, design the rocket body on compression by axial forces. After determination of the incidence angle α also the transverse aerodynamic forces can be calculated approximately, and by them we can perform calculations on durability in the section of removal, when noticeable deflection moments act on the rocket body.

The described principle of determination of motion parameters along a trajectory will apply to any types of rockets.

3. Flight Beyond the Limits of the Atmosphere in the Field of Terrestrial Gravitation

Motion Equations

A long-range rocket, such as a high-altitude meteorological rocket, for a large part of its trajectory passes through very rarefied atmospheric layers, so that in this section of flight it is possible to disregard aerodynamic forces acting on the rocket. If, moreover, the flight is accomplished without the engine working, then the equation of motion of the rocket as a material point can be integrated in final form at any distance of the rocket from the surface of the Earth.

From physics it is known that any body, cast at an angle θ_0 to the horizon with speed v_0 moves (if one were not to consider resisting forces of air) in a parabola. Actually, after time t after start of motion, the coordinates of the cast body will be

$$x = v_0 t \cos\theta_0;$$
$$y = v_0 t \sin\theta_0 - \frac{gt^2}{2}.$$

Excluding hence t, we will find an equation of a parabola shown in Fig. 8.10:

$$y = x \operatorname{tg}\theta_0 - x^2 \frac{g}{2v_0^2 \cos^2\theta_0}.$$

The obtained expression, however, is valid only in certain limits. During its derivation it was assumed that vector of acceleration due to gravity g for all points of the trajectory remains

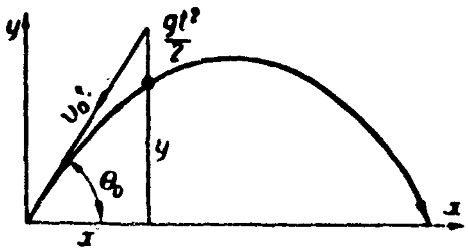

Fig. 8.10. Trajectory of a body, cast at an angle to the horizon (parabola).

parallel to axis y and is not changed by magnitude. In fact this is not so. Acceleration due to gravity g, due to ascent decreases proportionally to the square of the distance from the center of the Earth. Furthermore, the force of attraction of the Earth is directed constantly toward its center, and vectors g in different points of the trajectory will not be parallel to each other. These peculiarities of terrestrial gravitation cannot have essential value, if we consider low altitudes and small distances of casting of the body. But if great distances and altitudes are prevalent, characteristic for long-range ballistic and many meteorological rockets, the above circumstance must be taken into account.

The problem of determination of trajectory of rocket flight in such a situation coincides with the problem of determinating orbits of celestial bodies (problem of Kepler). The theory of motion of a body in these conditions is called "elliptic" in distinction from the elementary problem mentioned above on flight of a body along a parabola. The elliptic theory presents the greatest interest during solution of such basic cosmonautical problems as determination of artificial earth satellite orbits and orbits of space rockets.

Let us consider the motion of a rocket in a polar coordinate system with the pole in the center of the Earth.

Let us assume that the rocket, depicted on Fig. 8.11 by point A moves by inertia along a certain trajectory and in a given moment is at a distance r from the center of the Earth. During transition from point A to point B the kinetic energy of the rocket $Mv^2/2$ will be changed by magnitude $d(Mv^2/2)$. The potential energy will be changed by magnitude $d(Mg_r r)$. Acceleration due to gravity g_r here is not a constant.

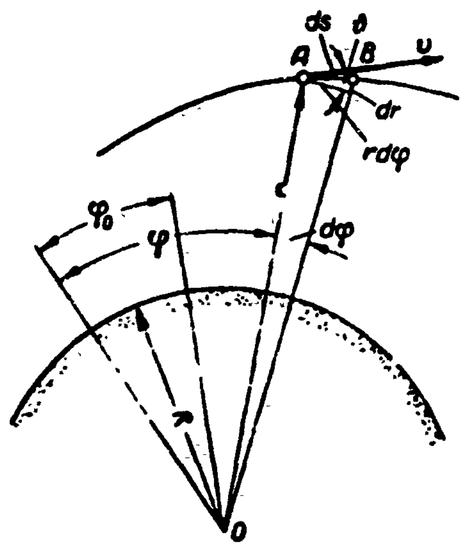

Fig. 8.11. Conclusion on equation of motion of a rocket in a polar coordinate system.

A change of kinetic energy is equal to a change of potential energy, inasmuch as the engine is not working:

$$d\left(\frac{Mv^2}{2} - Mg_r r\right) = 0.$$

Since mass M remains constant, then

$$\frac{v^2}{2} - g_r r = \text{const}.$$

Acceleration due to gravity is changed reciprocally to the square of distance to the center of the Earth:

$$g_r = \frac{k^2}{r^2},$$

where k^2 is a constant of the gravitation field [see formula (2.2)].

Thus,

$$\frac{v^2}{2} - \frac{k^2}{r} = \text{const.}$$

The obtained expression, called the energy integral, shows that the flight speed will depend only on distance r of a given point in the trajectory from the center of the Earth. The value of the constant in the right member of this expression characterizes the energy level of the trajectory and is determined by initial conditions of motion. In the initial point of the trajectory at $r = r_0$, $v = v_0$. Hence

$$\frac{v^2}{2} - \frac{k^2}{r} = \frac{v_0^2}{2} - \frac{k^2}{r_0}. \tag{8.8}$$

But speed

$$v = \frac{ds}{dt} = \frac{\sqrt{dr^2 + r^2 d\varphi^2}}{dt} = \sqrt{\dot{r}^2 + r^2\dot{\varphi}^2},$$

and then instead of expression (8.8) we get

$$\frac{1}{2}(\dot{r}^2 + r^2\dot{\varphi}^2) - \frac{k^2}{r} = \frac{v_0^2}{2} - \frac{k^2}{r_0}. \tag{8.9}$$

We will obtain the second equation from the condition that the angular momentum of the rocket about the center of the Earth is a constant:

$$Mrv\cos\vartheta = \text{const.} \tag{8.10}$$

Angle ϑ is shown on Fig. 8.11 – it is the angle of inclination of the trajectory to the local horizon; $v \cos \vartheta$ is the speed component, perpendicular to the radius.

From expression (8.10) it follows that

$$rv\cos\vartheta = r_0 v_0 \cos\vartheta_0,$$

where ϑ_0 is the angle of inclination of the trajectory at the initial point.

But from Fig. 8.11 it is clear that

$$\cos\vartheta = \frac{r\,d\varphi}{ds},$$

therefore,

$$rv\cos\vartheta = r\frac{ds}{dt}\frac{r\,d\varphi}{ds} = r^2\dot{\varphi}$$

and

$$r^2\dot{\varphi} = v_0 r_0 \cos\vartheta_0. \tag{8.11}$$

The obtained equation is called the integral of areas, inasmuch as it indicates constancy of the area, cut off by the radius-vector in a unit of time. This area, obviously, will be equal to $r^2\dot{\varphi}/2$.

If equations (8.9) and (8.11) were solved together and time t were excluded from them, then the dependence of r on φ will be obtained, i.e., the equation of the rocket trajectory.

Flight Trajectory

According to equation (8.11)

$$\dot{\varphi} = \frac{v_0 r_0 \cos\vartheta_0}{r^2}.$$

But

$$\dot{\varphi} = \frac{d\varphi}{dt} = \frac{d\varphi}{dr}\frac{dr}{dt} = \frac{d\varphi}{dr}\dot{r}.$$

Considering this, we get

$$\dot{r} = \frac{dr}{d\varphi}\frac{v_0 r_0 \cos\vartheta_0}{r^2},$$

or

$$\dot{r} = \frac{dr}{d\varphi}\frac{c}{r^2},$$

where

$$c = v_0 r_0 \cos\vartheta_0.$$

We replace this expression for $\dot{r}$ and expression for $\dot{\varphi}$ obtained from (8.11), in equation (8.9), then,

$$\left(\frac{dr}{d\varphi}\right)^2\frac{c^2}{r^4} + \frac{c^2}{r^2} - \frac{2k^2}{r} = v_0^2 - \frac{2k^2}{r_0}.$$

Hence

$$\frac{dr}{d\varphi}\frac{c}{r^2}=\pm\sqrt{v_0^2-\frac{2k^2}{r_0}-\frac{c^2}{r^2}+\frac{2k^2}{r}},$$

or

$$d\varphi=\mp\frac{d\left(\frac{c}{r}\right)}{\sqrt{v_0^2-\frac{2k^2}{r_0}-\frac{c^2}{r^2}+\frac{2k^2}{r}}}.$$

To the radicand we add and subtract constant k^4/c^2, and introduce the magnitude $-k^2/c$ under the sign of the differential in the numerator, and then we get

$$d\varphi=\mp\frac{d\left(\frac{c}{r}-\frac{k^2}{c}\right)}{\sqrt{\left(v_0^2-\frac{2k^2}{r_0}+\frac{k^4}{c^2}\right)-\left(\frac{c}{r}-\frac{k^2}{c}\right)^2}}.$$

Integration of the last expression gives

$$\varphi-\varphi_0=\arccos\frac{\frac{c}{r}-\frac{k^2}{c}}{\sqrt{v_0^2-\frac{2k^2}{r_0}+\frac{k^4}{c^2}}},$$

whence is obtained the desired equation of the trajectory

$$r=\frac{\frac{c^2}{k^2}}{1+\left(\frac{c}{k^2}\sqrt{v_0^2-\frac{2k^2}{r_0}+\frac{k^4}{c^2}}\right)\cos(\varphi-\varphi_0)}. \tag{8.12}$$

The constant of integration φ_0 depends on the origin of angle φ (see Fig. 8.11).

Let us introduce the following designations:

$$p=\frac{c^2}{k^2}=\frac{v_0^2r_0^2\cos^2\theta_0}{k^2}, \tag{8.13}$$

$$e=\frac{c}{k^2}\sqrt{v_0^2-\frac{2k^2}{r_0}+\frac{k^4}{c^2}}$$

or

$$e=\sqrt{1-\frac{2v_0^2r_0\cos^2\theta_0}{k^2}+\frac{v_0^4r_0^2\cos^2\theta_0}{k^4}}. \tag{8.14}$$

In result the equation of flight trajectory (8.12) will take the form

$$r = \frac{p}{1 + e\cos(\varphi - \varphi_0)}. \tag{8.15}$$

Let us analyze the obtained expression.

From analytic geometry it is known that equation (8.15) is an equation of a curve of the second order in a polar system of coordinates with the pole in one of the foci of the curve. Coefficient e is eccentricity of the curve. At $e < 1$ equation (8.15) is an equation of an ellipse, at $e > 1$ the equation of a hyperbola and, finally at $e = 1$ the equation of a parabola.

One should not confuse the parabola, given by equation (8.15) at $e = 1$ with the one obtained earlier for a body, freely cast and moving in a field of terrestrial gravitation, with the constant with absolute value and direction of the vector of acceleration due to gravity.

Let us first assume $e = 0$. Equation (8.15) will take the form

$$r = p = \text{const.}$$

In this case the body moves in a circle.

From (8.14) we get

$$v_0^2 = \frac{k^2}{r_0}\left(1 \pm \sqrt{1 - \frac{1}{\cos^2\vartheta_0}}\right).$$

It is not difficult to see that speed v_0 determined by the last formula has a real value only in the case when $\cos^2\vartheta_0 = 1$ and $\vartheta_0 = 0$. This means that the motion of the body in a circle is possible only under the condition that the direction of the rector of initial speed is parallel to the horizon.

For flight speed we obtain the expression

$$v_0 = \frac{k}{\sqrt{r_0}} = v_{кр}.$$

[кр = cir = circular]

This expression for first cosmic, or circular, speed is already familiar to us from Chapter II. At $r_0 \approx R$ the numerical value for the first cosmic speed

$$v_{кр} \approx 8000 \text{ m/sec}$$

At values of eccentricity

$$0<e<1,$$

as was already said, motion occurs in an ellipse. Equation (8.15) allows us to find radii of perigee r_π and apogee r_α of this ellipse:

$$r_\pi = \frac{p}{1+e}; \quad r_\alpha = \frac{p}{1-e}.$$

Hence the major semiaxis of the ellipse is

$$a = \frac{r_\pi + r_\alpha}{2} = \frac{p}{1-e^2}. \tag{8.16}$$

We will show that the energy level of an elliptic orbit determined by expression (8.8) depends only on the magnitude of the great semiaxis of the ellipse. Let us copy formula (8.14) in the form

$$e = \sqrt{1 + \frac{2r_0^2 v_0^2 \cos^2 \vartheta_0}{k^4}\left(\frac{v_0^2}{2} - \frac{k^2}{r_0}\right)}, \tag{8.17}$$

whence, considering (8.13) and (8.16) we get

$$\frac{v_0^2}{2} - \frac{k^2}{r_0} = \frac{k^2(e^2-1)}{2p} = -\frac{k^2}{2a}. \tag{8.18}$$

With an increase in a the energy level of the orbit is increased (its negative value decreases). On the other hand, lowering of the energy level of the orbit is always accompanied by a decrease of the great semiaxis of the ellipse (see Chapter II, p.69).

Let us find the magnitude of the period of conversion T for an elliptic orbit. The area which is cut off by the radius-vector during motion along the orbit in time t is, as follows from (8.11),

$$\frac{r^2\dot{\varphi}}{2}t = \frac{c}{2}t.$$

During the time of a full rotation T, the radius-vector will describe the total area of an ellipse equal to πab, where b is the small semiaxis of the ellipse:

$$b=a\sqrt{1-e^2}.$$

Thus,

$$\frac{c}{2}T=\pi a^2\sqrt{1-e^2},$$

whence the period of conversion

$$T=\frac{2\pi a^2\sqrt{1-e^2}}{c}.$$

The value of the constant of ellipse areas can be determined from relationship (8.13):

$$c=k\sqrt{p}.$$

Consequently, considering (8.16) we get finally

$$T=\frac{2\pi}{k}a^2\sqrt{\frac{1-e^2}{p}}=\frac{2\pi}{k}a^{\frac{3}{2}}.$$

We obtained formula (2.5), which we used in Chapter II for determination of the period of conversion of artificial earth satellites.

Let us now clarify, under what conditions eccentricity e will appear greater than one.

From expression (8.17) it is possible to easily establish that independently of the inclination angle of the vector of initial speed ϑ_0 at

$$v_0\geqslant\sqrt{2}\frac{k}{\sqrt{r_0}}$$

the magnitude

$$e\geqslant 1.$$

At

$$v_0=\sqrt{2}\frac{k}{\sqrt{r_0}},$$

e = 1, and motion occurs along a parabola, but at

$$v_0>\sqrt{2}\frac{k}{\sqrt{r_0}},$$

when e > 1, the trajectory of motion will be a hyperbola.

Speed

$$v_{пар} = \sqrt{2}\frac{k}{\sqrt{r_0}} = \sqrt{2}v_{кр}$$

[пар = par = parabolic]

is called parabolic speed, or second cosmic speed. This is that minimum initial speed which is necessary for surmounting the field of gravitation when a body, possessing such speed, departs to any great distance from the attracting center.

Considering $r_0 \approx R$, we obtain for the second cosmic speed the following numerical value:

$$v_{пар} \approx 11\,300 \text{ m/sec}.$$

Figure 8.12 shows the change in form of the trajectory of flight of a body depending upon speed at a constant launching angle $\vartheta_0 = 30^\circ$.

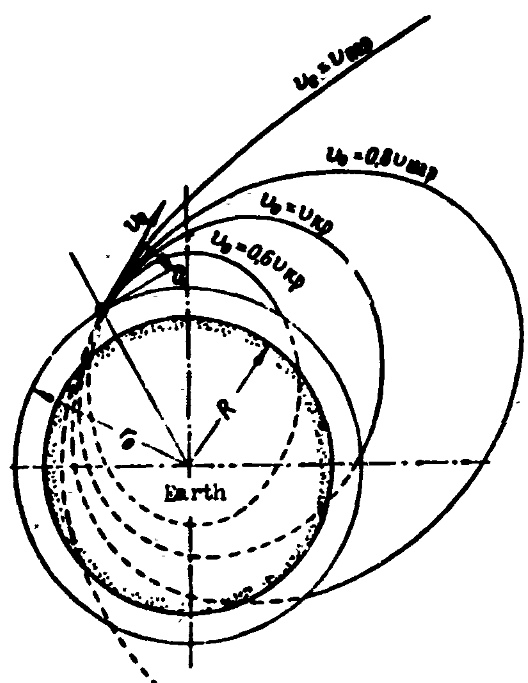

Fig. 8.12. Flight trajectory of a body, depending upon launching speed at $\vartheta_0 = 30^\circ$.

At a low launching speed of a body at an angle ϑ_0 to the horizon, describing an elliptical arc, it returns on Earth. Due to an increase of initial speed, the flying range increases. Also the arc, described by the flying body is increased.

At a speed $v_0 = v_{par}$ the trajectory becomes a parabola. A body, cast from Earth with a speed $v_0 \geqq v_{par}$, does not return to Earth.

For ballistic long-range rockets, a significant part of the trajectory of which passes beyond the borders of the atmosphere, there is the possibility of direct application of the elliptic theory to calculation of motion parameters.

To a certain altitude, while the influence of aerodynamic forces is still essential, calculation of the trajectory is conducted by the above described method of numerical integration. From the moment when it is possible to disregard aerodynamic forces, calculation is conducted by the elliptic theory formulas. Thus, there is the possibility to obtain a full picture of change in coordinates, speeds and accelerations depending upon time of flight for the whole trajectory.

Figure 8.13 shows graphs of speed and tangential acceleration of a ballistic rocket for the whole flight trajectory.

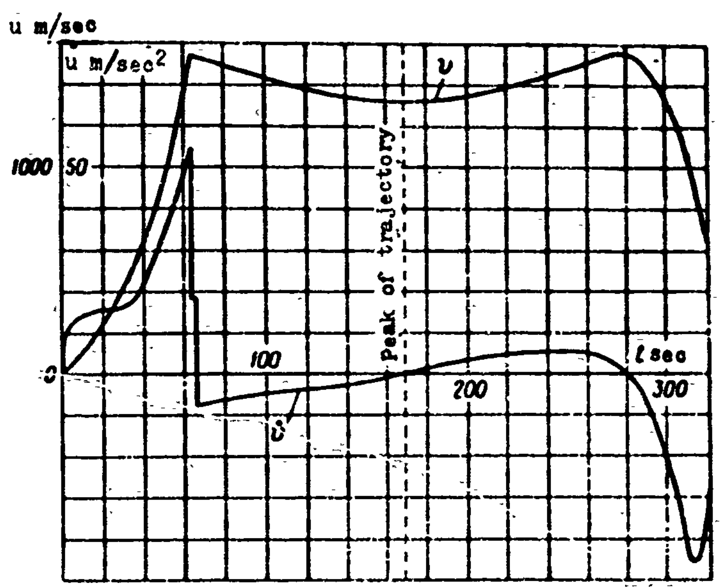

Fig. 8.13. Speed and tangential acceleration of a ballistic rocket on trajectory.

Speed increases to the moment of engine cut-off. After cut-off the speed drops, inasmuch as the rocket climbs by inertia. After the rocket reaches the peak of the trajectory, speed again starts to increase. Subsequently, under action of the resisting force of air the speed decreases.

Acceleration in the beginning grows according to the law of the active section, shown on Fig. 8.7. During transition of the engine to the final step, acceleration decreases by a jump. Then at full cut-off of the engine, acceleration becomes negative. It, obviously, is equal to $-g \sin \vartheta$. Further, tangential acceleration is changed due to change in ϑ and g. In the peak of the trajectory it becomes zero. In the descending section of the trajectory, the force of terrestrial attraction accelerates the rocket. At entrance into the upper layers of the atmosphere braking of the rocket due to resistance of the atmosphere starts. In the actual end of the trajectory, the rocket speed noticeably drops, and the drag force decreases. Acceleration in absolute magnitude also decreases.

With increase of distance of ballistic rockets, the diurnal rotation of the Earth becomes all the more essential for calculation of the trajectory. In this case in derivation of the motion equations in the starting system of coordinates it is necessary to consider Coriolis acceleration, caused by translational rotation of the Earth.

Transition to absolute motion, calculated on the basis of formulas of the elliptic theory, is carried out as a result of vector summation of relative speed, obtained as a result of numerical integration of motion equations in the starting coordinate system and translational speed in a given point of the trajectory, taken as the origin of the elliptic section. The component of translational speed directed from west to east perpendicular to the plane of the meridian, passing through a given point, is equal to

$$v_e = r_A \omega_З \cos \varphi_A,$$

[3 = E = Earth]

where r_A and φ_A — correspondingly, radius and geographic latitude of origin of elliptic section of trajectory;

$\omega_E = 7.2921 \cdot 10^{-5} \frac{1}{\text{sec}}$ — angular velocity of diurnal rotation of the Earth.

As a result, the vector of absolute velocity in the beginning of the elliptic section will be somewhat different both in magnitude and in the direction from the vector of relative speed in the starting system of coordinates. Change of speed modulus at the expense of rotation of the Earth during transition to absolute motion can be approximately estimated by the formula

$$\Delta v \approx r_A \omega_3 \cos i,$$

where i — inclination of plane of trajectory with respect to the Earth's equator.

Subsequently, after calculation of the elliptic section in an absolute system of coordinates, one should consider that during the time of flight in this section t_{el}, the Earth turns from west to east by an angle $\omega_E t_{el}$, and the final point of the elliptic section with respect to the Earth will be additionally displaced along a parallel in a western direction by the magnitude

$$r_C \omega_3 t_{эл} \cos \varphi_C.$$

[эл = el = elliptic]

where r_c and φ_c — radius and geographic latitude of final point of elliptic section.

The atmospheric section of the descending phase of the trajectory, just as the active section, is calculated by numerical integration of motion equations in a relative system of coordinates, connected with the Earth, after analogous transition here from absolute to relative motion.

Thus, during calculation of trajectories of ballistic rockets the influence of rotation of the Earth is considered.

Flying Range

At small speeds the flying range of a body cast from Earth at an angle θ_0 to the horizon is determined by the elementary formula, obtained from the expressions, shown on p. 375:

$$L = \frac{v_0^2}{g} \sin 2\theta_0.$$

The maximum distance is at $\theta_0 = 45^\circ$. This result is well-known.

The matter of launching at high speeds is more complicated.

In this case it is expedient to isolate the elliptic section of the trajectory, the beginning A and end C of which are located at one altitude (Fig. 8.14), and to characterize the distance of this section by the central angle 2β, formed by polar radii passing through points A and C. Due to symmetry of the trajectory it is sufficient to determine half of this angle — angle β (from point A to point B on the trajectory).

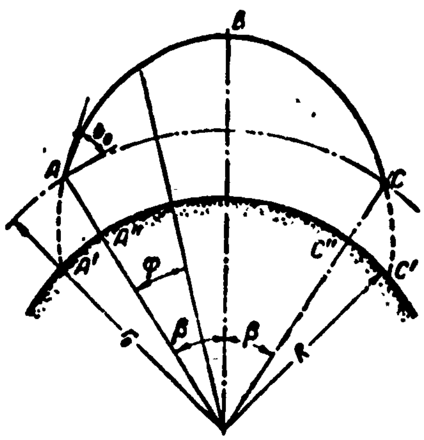

Fig. 8.14. Determination of flying range of a ballistic rocket.

Let us return to the trajectory equation (8.15):

$$r = \frac{p}{1 + e\cos(\varphi - \varphi_0)}.$$

The vectorial angle, corresponding to the maximum value of the radius r_α will be designated by φ_α. Obviously,

$$r_a = \frac{p}{1-e},$$

i.e.,

$$\varphi_a - \varphi_0 = \pi.$$

This means that

$$\varphi_0 = \varphi_a - \pi$$

and equation (8.15) can be rewritten in the form

$$r = \frac{p}{1 - e\cos(\varphi_a - \varphi)}. \tag{8.19}$$

If reading of angles φ were conducted from the radius of the origin A (see Fig. 8.14) then at $\varphi = 0$ we have

$$r = r_0,$$
$$\varphi_a = \beta$$

and equation (8.19) gives

$$\cos\beta = \frac{1}{e}\left(1 - \frac{p}{r_0}\right).$$

In order to have the possibility to judge that quarter, in which angle β is located, we find also

$$\sin\beta = \pm\sqrt{1-\cos^2\beta} = \pm\frac{1}{e}\sqrt{(e^2-1)+\frac{2p}{r_0}-\frac{p^2}{r_0^2}}.$$

On the basis of relationship (8.18)

$$e^2-1=\frac{v_0^2 p}{k^2}-\frac{2p}{r_0},$$

therefore,

$$\sin\beta=\pm\frac{\sqrt{p}}{e}\sqrt{\frac{v_0^2}{k^2}-\frac{p}{r_0^2}}.$$

Using expression (8.13), we get

$$\cos\beta=\frac{1}{e}\left(1-\frac{v_0^2 r_0}{k^2}\cos^2\vartheta_0\right),$$

$$\sin\beta=\frac{1}{e}\frac{v_0^2 r_0}{k^2}\sin\vartheta_0\cos\vartheta_0.$$

In the right part of the second relationship the plus sign is kept, inasmuch as the signs of angles ϑ_0 and β always coincide.

Thus,

$$\mathrm{tg}\,\beta=\frac{\frac{v_0^2 r_0}{k^2}\sin\vartheta_0\cos\vartheta_0}{1-\frac{v_0^2 r_0}{k^2}\cos^2\vartheta_0}$$

or

$$\mathrm{tg}\,\beta=\frac{\frac{v_0^2}{v_{cir}^2}\mathrm{tg}\,\vartheta_0}{1-\frac{v_0^2}{v_{cir}^2}+\mathrm{tg}^2\vartheta_0}, \qquad (8.20)$$

where v_{cir} – circular speed at a given initial altitude.

If this expression were differentiated by tg ϑ_0 and the derivative were equated to zero, then without special work we can find that angle of departure ϑ_0, at which magnitude β will be maximum at a given speed v_0:

$$\mathrm{tg}\,\vartheta_0=\sqrt{1-\frac{v_0^2}{v_{cir}^2}}. \qquad (8.21)$$

At a small speed v_0

$$\mathrm{tg}\,\vartheta_0 \approx 1.$$

This result was already indicated above; for obtaining of maximum distance of launching, the vector of the initial speed should be at an angle of 45° with the horizon.

At $v_0 = v_{cir}$, as was noted, $\mathrm{tg}\,\vartheta_0 = 0$ and $\vartheta_0 = 0$. This case corresponds to constant rotation of the body around the Earth with a first cosmic speed.

Figure 8.15 shows graphs of a dependence of the optimum angle of departure on relationship v_0/v_{cir}. From the graph it is clear that the optimum angle decreases with growth of launching speed. At speeds of the order 2000 m/sec, angle ϑ_0 differs little from 45° (it is equal to about 44°). At high speeds it decreases all more and more noticeably.

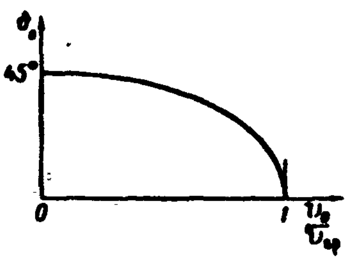

Fig. 8.15. Dependence of optimum angle of departure ϑ_0 on initial speed v_0 (v_{cir} — circular speed).

At an optimum angle ϑ_C

$$\mathrm{tg}\,\beta_{max} = \frac{\dfrac{v_0^2}{v_{кр}^2}}{2\sqrt{1-\dfrac{v_0^2}{v_{кр}^2}}}. \qquad (8.22)$$

If the square of relationship v_0/v_{cir} is small as compared to one, then we can get

$$2\beta \approx \frac{v_0^2}{v_{кр}^2}.$$

At $v_0 = 2000$ m/sec, for instance,

$$2\beta \approx \left(\frac{2000}{8000}\right)^2 = \frac{1}{16}.$$

Flying range, measured along the arc A"C" (see Fig. 8.14), is equal to

$$2\beta R=\frac{1}{16}6400=400 \text{ km}.$$

The given calculation of distance will apply for the section of the trajectory, lying higher than the dense layers of atmosphere (section ABC on Fig. 8.14). The real distance will be somewhat greater at the expense of trajectory sections A'A and CC'. The first of them A'A is determined, as was already said, by the program of change of pitch angle on the powered-flight trajectory. Section CC' can be considered by its form as a continuation of the basic elliptic curve. The form of the trajectory for a large part of this section cannot noticeably differ from an ellipse. Only the speed of rocket changes noticeably from resistance of air, and consequently also the flight time. Error in determination of distance from replacement of section CC' by the arc of the elliptic trajectory is small, because section CC' itself is small as compared to total distance. Thus, additional distance on section CC' may be approximately calculated by the formulas (8.13), (8.14) and (8.15).

The presence of section CC' affects the size of the optimum angle of departure ϑ_0. The condition of maximum distance from point A to C' will be different from condition (8.21), derived for maximum distance from A to C. Due to difference of altitudes of points A and C', optimum angle ϑ_0 in fact turns out to be somewhat less than determined by the formula (8.21). Thus, for instance, for a long-range ballistic rocket, having at the end of the active section a speed 1500 m/sec, the optimum angle of departure ϑ_0, calculated by the formula (8.21), turns out to be equal to $44^\circ30'$. If however, we consider the presence of section CC', then the optimum angle, ensuring a maximum of distance A"C', should be approximately equal to 42°.

4. Loads Acting on the Basic Elements of Construction of a Rocket in Flight

Axial Loads in the Active Section

In Chapter VII in examining of forces and moments acting on a rocket in flight, the question was not placed of what loads separate joints of the rocket perceive. Discussion centered around the rocket as a rigid integer, while resultant forces and moments were determined in such a manner so that it was possible to produce only ballistic calculations and to find the laws of rocket motion.

However, during appraisal of durability of construction it is necessary to know not only resultant external forces, but also distributive laws of loads in the structure, so that it is possible to determine internal forces and stress in the basic carrier elements of the structure.

We will stop at first on the simplest question of the appearance of axial internal forces in the rocket in the powered-flight section.

We will suppose that a ballistic long-range rocket is built with a carrier body and suspension tanks (Fig. 8.16).

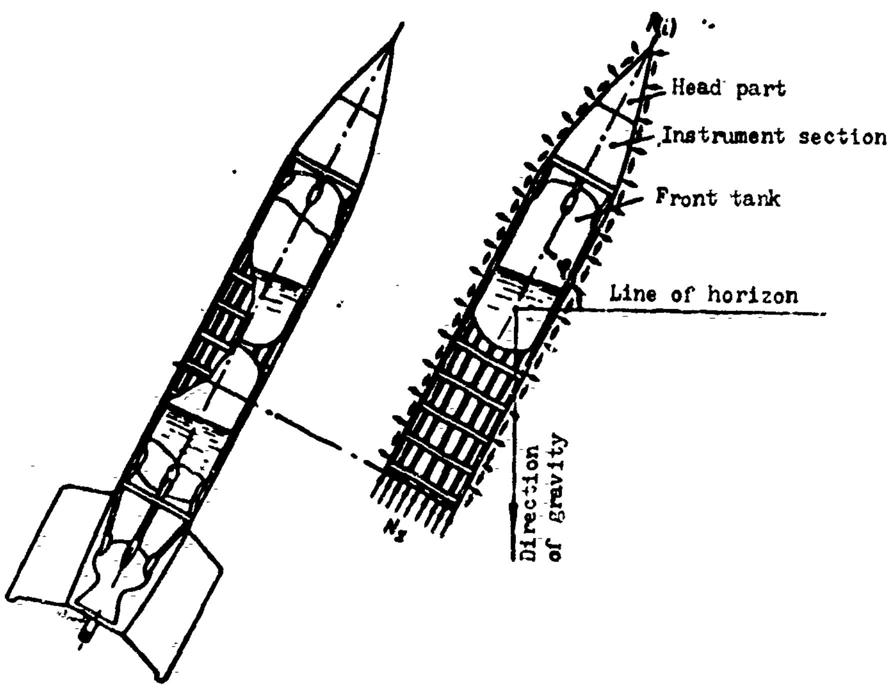

Fig. 8.16. Determination of the axial force in the body of a rocket with suspension tanks.

We will determine the axial compressing force N_x, appearing in an arbitrary transverse section of the carrier body in a certain moment of time. In examining the equilibrium conditions of the front part of the rocket, we get

$$N_x = R_i + (G'_{г.ч} + G'_{п.о} + G'_{п.б} + G'_{корп\,(i)}) \sin\varphi +$$
$$+ (M_{г.ч} + M_{п.о} + M_{п.б} + M_{корп\,(i)}) j_x.$$

[г.ч = h.p = head part; п.о = i.s = instrument section; п.б = f.t = front tank; корп = body]

where $M_{h.p}$, $M_{i.s}$, $M_{f.t}$, $M_{body\ (i)}$ – mass of head part, instrument section, front tank (together with contents) and part of body before the considered section;

$G'_{h.p}$, $G'_{i.s}$, $G'_{f.t}$, $G'_{body\ (i)}$ – weight of the same parts at a given altitude of flight;

$R_{(i)}$ – axial component of aerodynamic forces acting on the lateral surface of the considered part of the rocket;

j_x – projection of total acceleration on the longitudinal axis of the rocket:

$$j_x = \frac{P - X_{упр} - R}{M} - g_h \sin\varphi,$$

where R is the axial aerodynamic force.

In the obtained expression for N_x the weight and mass of the rear fuel tank together with its contents does not enter, inasmuch as it is braced to the rear support frame of the body.

The expression for force N_x becomes

$$N_x = R_{(i)} + (G_{г.ч} + G_{п.о} + G_{п.б} + G_{корп\,(i)}) \frac{j_x + g_h \sin\varphi}{g_0},$$

where $G_{h.p}$, $G_{i.s}$, $G_{f.t}$, and $G_{body\ (i)}$ – weight of head part, instrument section, front tank and part of body (on Earth);

g_h and g_0 – acceleration of the force of terrestrial gravitation corresponding to altitude h and at Earth level.

Magnitude

$$n_x = \frac{j_x + g_h \sin\varphi}{g_0} \qquad (8.23)$$

is called the coefficient of axial overload or simply axial overload. It shows by how many times the axial component of the apparent structural weight in a given moment is larger than the real weight under the conditions of the earth's surface. At starting, at $j_x = 0$ and $\varphi = 90^\circ$, $n_x = 1$. In conditions of free flight of a rocket with a nonworking engine, beyond the borders of the atmosphere, $j_x = -g_h \sin\varphi$ and overload $n_x = 0$.

Figure 8.17 shows the law of axial overload change in a rocket on the powered-flight trajectory.

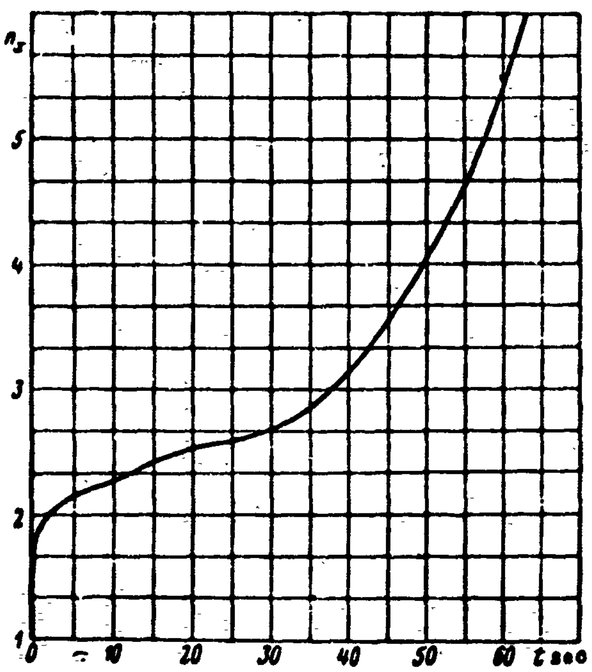

Fig. 8.17. Law of axial overload change of a V-2 rocket on the trajectory.

By analogy with expression (8.23) it is possible to write a formula also for transverse overload in the presence of a transverse component of apparent acceleration

$$n_y = \frac{j_y + g_h \cos\varphi}{g_0}. \qquad (8.24)$$

We will use this expression subsequently.

According to the accepted designation, (8.23) can be written as

$$N_x = R_{(i)} + (G_{r.ч} + G_{п.о} + G_{п.б} + G_{корп(i)})\, n_x. \qquad (8.25)$$

Force N_x is a composite function of time t and is determined by laws of change of magnitudes, entering into expression (8.25) Thus, from the preceding account it is already known that the force of frontal aerodynamic drag X in the active section first increases, and then drops practically to zero. In the same way the axial aerodynamic force $R_{(i)}$ will also be changed. Weights $G_{h.p}$, $G_{i.s}$ and $G_{body\ (i)}$ remain constant, and $G_{f.t}$ decreases in connection with flow rate of the fuel component. Overload n_x in the powered-flight section increases. Knowing the laws of change of components in expression (8.25), it is possible without special work to establish also the law of change of the axial compressing force in the trajectory. Figure 8.18 gives the dependence of force N_x for a V-2 rocket body in the front (I-I) and the rear (II-II) sections. The obtained curves differ little from each other, inasmuch as the mass of the carrier body is comparatively small. The compressing force in section II-II attains the biggest value at the end of the powered-flight section. In section I-I the maximum of force N_x has place in the 30th second of flight.

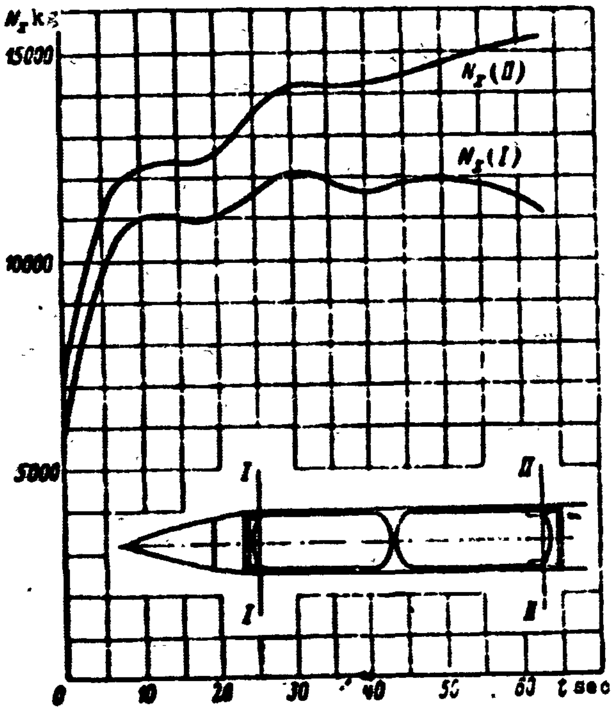

Fig. 8.18. Law of change of axial compressing force in a powered V-2 rocket body on the trajectory.

In the case when the tanks are carriers, axial force N_x in a transverse section of the rocket is determined in the exact same way. However, in distinction from the preceding, here it is necessary to take into account the unloading force of internal pressure, equal to the product of feed pressure in the tank on the area of the consider-transverse section $\frac{\pi}{4}D^2_{in}$, where D_{in} is the tank diameter on its internal surface (Fig. 8.19).

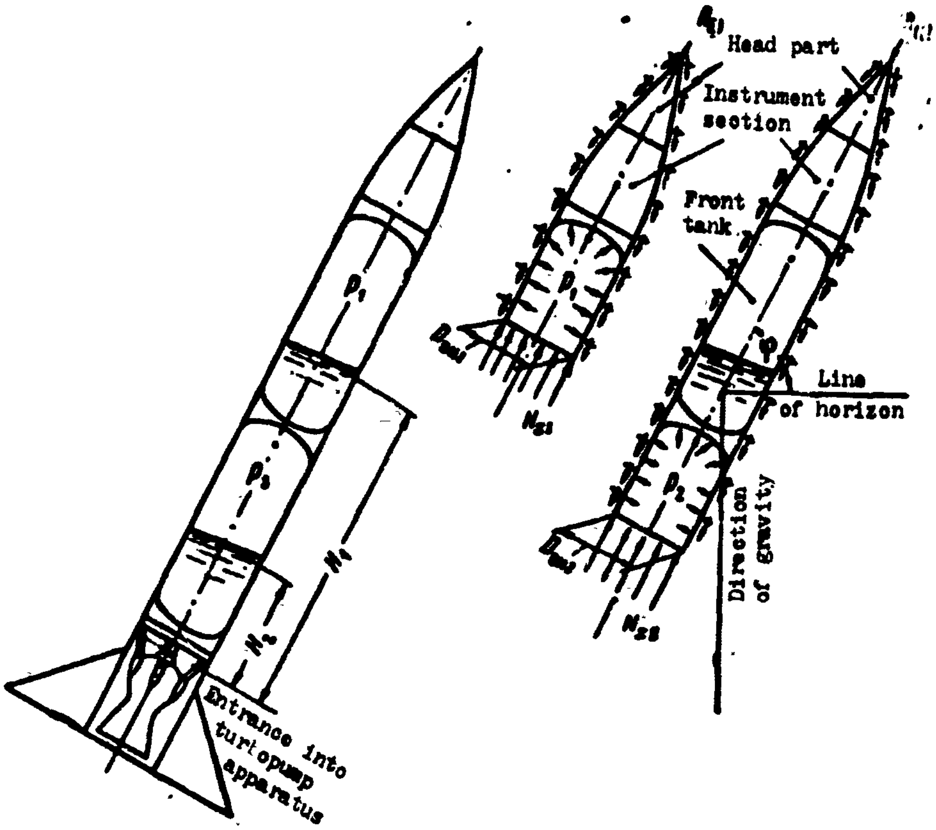

Fig. 8.19. Determination of the axial force in carrier tanks of a ballistic rocket.
[BH = in = internal]

Thus, for the front tank

$$N_{xI} = R_{(i)} - p_1 \frac{\pi}{4} D^2_{вн1} + (G_{г.ч} + G_{п.о} + G_{п.б\,(i)})\, n_x,$$

where $G_{f.t\ (i)}$ is the weight of the part of the structure of the carrier front tank in the considered section.

For the rear tank we get

$$N_{xII} = R_{(i)} - p_2 \frac{\pi}{4} D^2_{вн2} + (G_{г.ч} + G_{п.о} + G_{п.б} + G_{з.б\,(i)})\, n_x.$$

[з.б = r.t = rear tank]

where $G_{f.t}$ – weight of carrier front tank, including weight of component contained in tank in the given moment;

$G_{r.t\ (i)}$ – weight of part of structure of carrier rear tank in the considered section.

Depending upon the magnitude of feed pressure, force N_x can appear to be not compressing, as this was assumed, but stretching. For instance, in the case of displacive supply of fuel components, the pressure in the tanks is great. It, as it is known, exceeds the magnitude of operating pressure in the combustion chamber. In this case magnitudes $p_1\frac{\pi}{4}D^2_{in1}$ and $p_2\frac{\pi}{4}D^2_{in2}$ turn out to be large, and forces N_{xI} and N_{xII} knowingly will be stretching.

In a turbopump supply system the feed pressure in the tanks is small. It is obtained with such calculation, so that the pressure of components at entrance into the turbopump unit p_{out} is sufficient to guarantee uninterrupted work of centrifugal pumps. The pressure of the fuel component on entrance into the pump p_{out1} is composed of feed pressure in the tank p_1 and pressure of the liquid column at a height H_1 from the surface of liquid in the tank to the section of entrance into the pump (see Fig. 8.19). Axial overload renders direct influence on the magnitude of this pressure

$$p_{вх1}=p_1+n_x\gamma_1H_1,$$

[вх = out = outer]

where γ_1 is the specific gravity of the given component.

Depending upon the necessary magnitude of feed pressure in flight, force N_x can appear both compressing and stretching. Here cases are possible, where the sign of N_x changes on the trajectory.

Obviously, a small feed pressure in the carrier tanks turns out to be profitable for guarantee of carrier ability of the system. At large compressing forces, loss of stability of the walls of the compressed tanks is possible. Pressures p_1 and p_2 are in this sense unloading factors, and their small increase increases the reserve of stability. At further increase of p_1 and p_2 the danger of bursting of the tanks under internal pressure appears.

Axial stretching forces during flight will be experienced by the body of the tail part of the rocket. From the equilibrium condition for the tail part of such a rocket as V-2 (Fig. 8.20), we have, obviously,

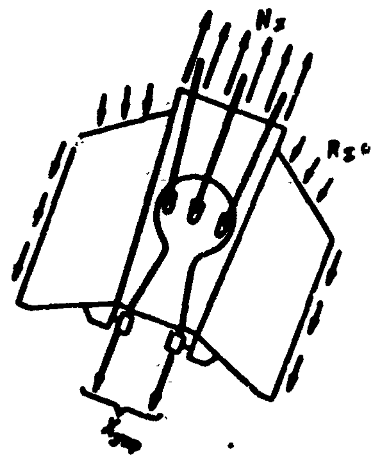

Fig. 8.20. Determination of axial force in tail cowl of rocket.

$$N_x = -(R_{x.o} + G_{x.o} n_x + X_{\text{упр}}).$$

$[x.o$ = t.c = tail compartment$]$

where $G_{t.c}$ – weight of tail cowl (without propulsion system);

$R_{t.c}$ – axial aerodynamic force, acting on tail part;

X_{con} – axial resisting force of controls situated in the gas flow of the engine. In the case where the controls are braced directly to the engine nozzle, this force in the expression for N_x would be absent.

Lateral Load in Powered-Flight Section

In the active section of controlled flight a rocket experiences not only action of axial compressing forces, but perceives lateral loads also, leading to bending of the body. This occurs on the atmospheric section of the trajectory during flight of a rocket with an angle of incidence. For a ballistic rocket the most dangerous from this point of view will be the start of the section of turning, when the rocket receives a signal by program for transition from a straight vertical ascent to slanted in the section of cut-off. For zenith rockets the greatest transverse and lateral loads appear during maneuver at high speed.

Let us consider flight of a ballistic rocket in the section of removal. We will, consider that the angular velocity of rocket turning by tangent is small and is changed very slowly. In this case control force Y_{con}, forcing the rocket to maintain a given angle of incidence, is determined from equilibrium conditions without calculation of damping moments and angular acceleration of the rocket. Equating the forces acting on the rocket's center of gravity (Fig. 8.21) to zero, we get

$$Y_{\text{упр}} = N\frac{e}{c} \approx Y\frac{e}{c}.$$

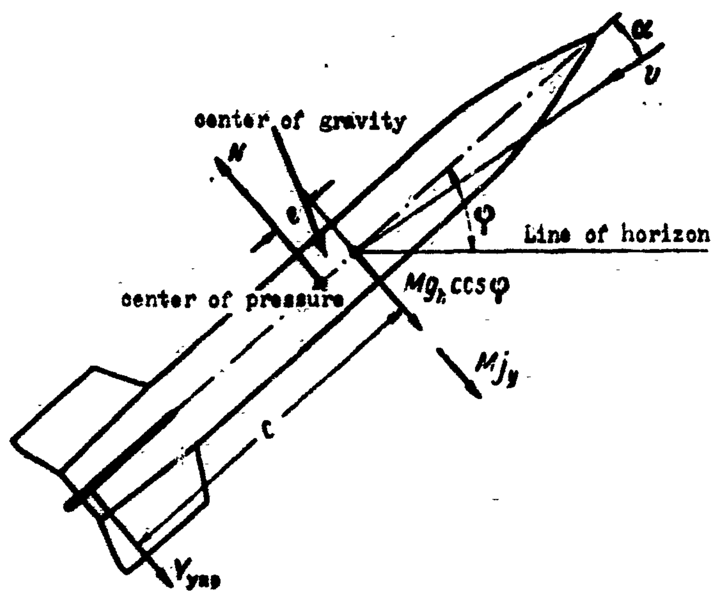

Fig. 8.21. Appearance of transverse overload in the section of removal.
[y = acc]

Figure 8.21, besides transverse aerodynamic force N, show also the transverse component force of weight $Mg_h \cos\varphi$ and transverse force of inertia Mj_{acc}, where j_{acc} is the transverse acceleration of the rocket under the action of forces N, Y_{con} and $Mg_h \cos\varphi$:

$$j_y = \frac{N - Y_{\text{упр}} - Mg_h\cos\varphi}{M} = \frac{N}{M}\left(1 - \frac{e}{c}\right) - g_h\cos\varphi. \qquad (8.20)$$

Let us consider the rocket as a beam, loaded by transverse forces.

By q_N kg/m we will designate the transverse aerodynamic force per unit of beam length. The law of change of q_N along the rocket axis is determined by results of wind tunnel tests. Magnitude q_N, as also the magnitude of the transverse aerodynamic force, depends on the angle of incidence and impact pressure. The resultant of forces q_N is equal to the transverse force N

$$N=\int_0^l q_N\,dx$$

and is applied to the center of pressure. Figure 8.22 shows the typical distributive law of forces q_N along a ballistic rocket axis, supplied with stabilizers for certain moments of flight in the active section.

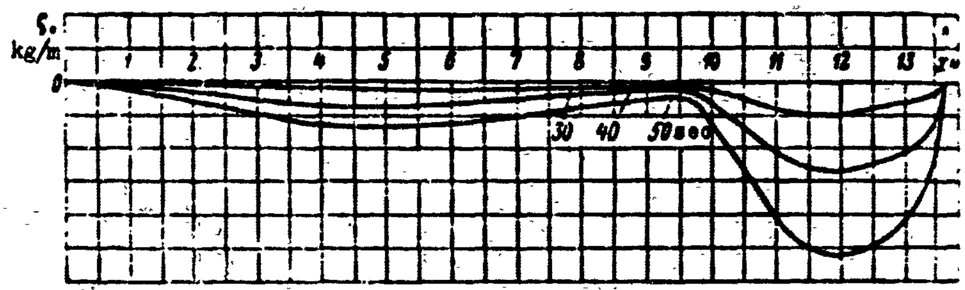

Fig. 8.22. Distribution of transverse aerodynamic load for a V-2 rocket in different moments of flight.

Let us consider further the mass forces — inertia and weight. By $\overline{G}$ kg/m we will designate the weight of a unit of rocket length. Then intensity of transverse mass forces q_M kg/m will be

$$q_M=\frac{\overline{G}}{g_0}j_y+\frac{\overline{G}}{g_0}g_h\cos\varphi=\overline{G}\,\frac{j_y+g_h\cos\varphi}{g_0}.$$

Considering designation (8.24) for transverse overload, we get

$$q_M=\overline{G}n_y.$$

Since transverse acceleration j_{acc} is determined by expression (8.26), then, obviously,

$$n_y=\frac{N}{Mg_0}\left(1-\frac{e}{c}\right)\approx\frac{Y}{Mg_0}\left(1-\frac{e}{c}\right).$$

Laws of change of magnitudes Y, M, e and c on trajectories are known. Therefore, transverse overload may be defined as a function of time for the given rocket in conditions of removal by a given program of pitch. Figure 8.23 shows a graph of the change of transverse overload for a V-2 rocket.

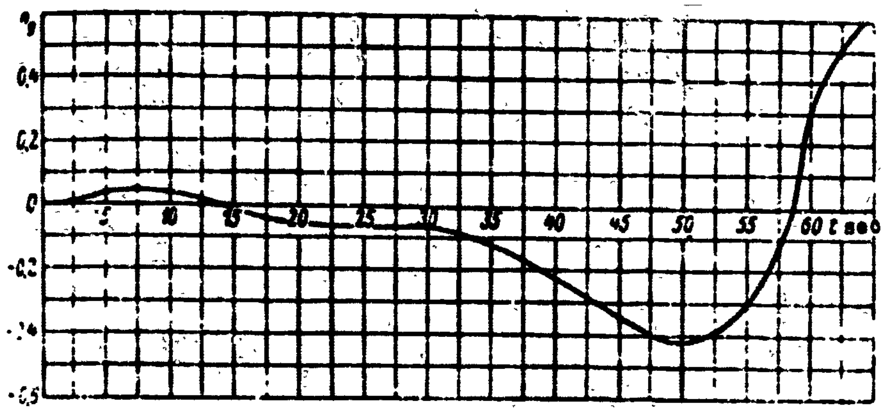

Fig. 8.23. Law of change of transverse overload of a V-2 rocket in the active section of flight.

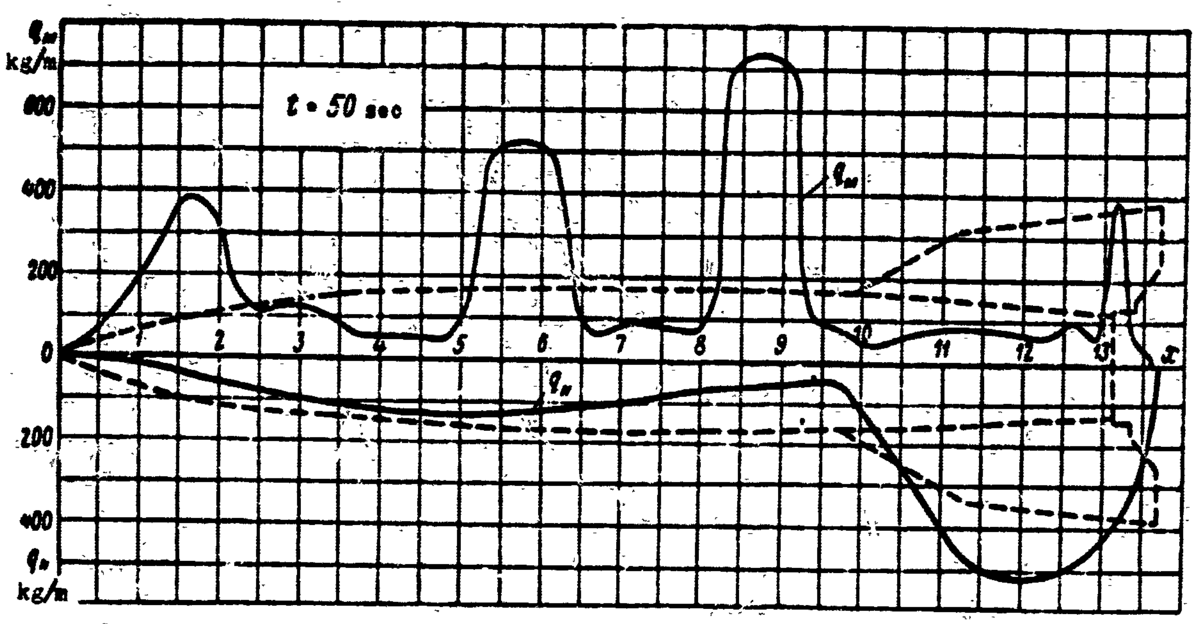

Fig. 8.24. Forces q_N and q_M, acting on the carrier elements of the V-2 rocket structure in the 50th second of flight.

In result, on the section of removal the rocket is loaded by a system of transverse forces in the form of two distributed loads of intensity q_N and q_M. Furthermore, a concentrated force Y_{con} is applied to the rocket. Figure 8.24 shows the system of distributed forces q_N and q_M for a ballistic V-2 rocket in the 50th second of the active section. It is clear that this picture changes in time, inasmuch as the angle of incidence, speed of flight, and also the magnitude of transverse overload are changed.

In any moment of time the considered force system satisfies conditions of equilibrium. Therefore, it is possible by the usual method to draw stress-strain diagrams of the bending moments along the rocket length. Determining numerically the sum of moments of forces, lying along one side of an arbitrarily taken section with coordinate x (see Fig. 8.24), we find the bending moment in this section. Thus, by using such points a stress-strain diagram of bending moments is drawn. For every moment of time there will be an individual stress-strain diagram. Figure 8.25 shows a stress-strain diagram of the bending moment for a V-2 rocket in the 50th second of the active section.

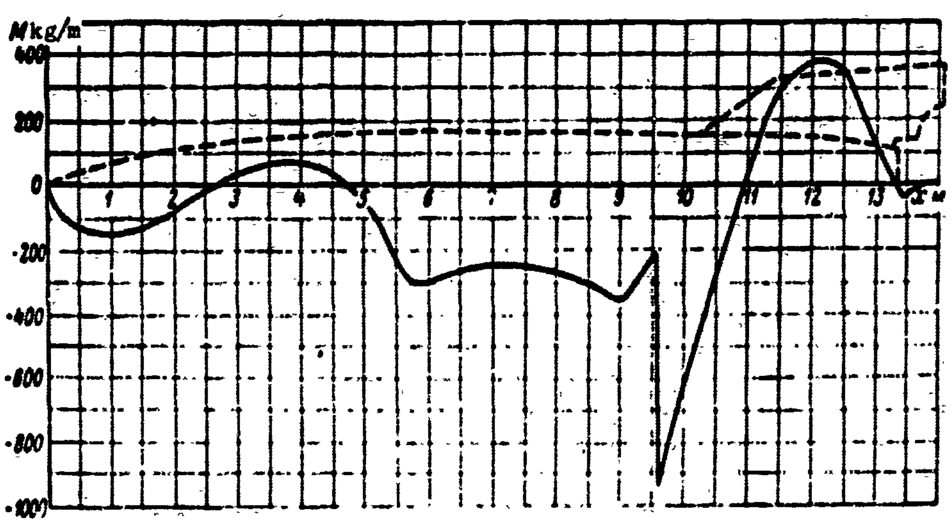

Fig. 8.25. Stress-strain diagram of bending moment along the V-2 rocket body length in the 50th second of flight.

A sharp change of bending moment at the joint of tail and fuel sections is explained by conditions of propulsion system bracing. With lateral loads, from the engine mass through its frame on the support ring a concentrated moment is transmitted, leading to a jump in the diagram of bending moments.

Lateral Loads in the Atmospheric Section of Free Flight

In the section of uncontrolled flight in a vacuum, the ballistic rocket accomplishes indefinite rotation about the center of mass, as this was shown earlier on Fig. 8.2. At entrance into dense layers of atmosphere, under action of aerodynamic forces the statically stable rocket stabilizes, taking an orientation along the flow. In the process of stabilization the rocket body experiences significant lateral loads. The problem of determination of these loads is more complicated than for the above considered case of the transverse load of a rocket in programmed flight.

In the atmospheric section of free flight the rocket is under the action of gravity and aerodynamic forces, where the latter due to great speed of flight exceeds the force of weight by a few times. Due to the fact that in the beginning the rocket does not have a definite angular orientation, the aerodynamic forces turn out to be also indefinite. Depending upon initial conditions of entrance into the atmosphere, both the magnitude of the biggest loads and the moment of time corresponding to the biggest load change.

In the upper layers of the atmosphere the rocket speed is great, and flight occurs with large angles of incidence, but due to low density of the atmosphere, transverse overloads are small. In the lower layers of the atmosphere the rocket speed is essentially less. Simultaneously the angles of incidence are minute. However, in lower layers of the atmosphere air density increases, determining the large magnitude of transverse loads. There exist, regions of altitudes, in which transverse overloads attain a maximum. For a V-2 rocket this has place at altitudes of the order of 20-30 km.

The law of rocket motion in the described conditions may be fixed by means of numerical integration of motion equations at different initial conditions of entrance into the atmosphere (rotation with a certain angular velocity, entrance into the atmospheric section at different orientations in space). As initial data for such calculation it is necessary, obviously, to know the aerodynamic coefficients of the rocket at any large angles of incidence or, so to speak, to have results of circular wind tunnel tests of a model at various flow speeds. In setting up the motion equations the rocket should be considered as a free solid body. As a result of a series of similar calculations (at different initial conditions a range of values of maximum overloads and intervals of flight time corresponding to them in the section of stabilization are established.

Analysis of conditions of ballistic long-range rocket flights and that of geophysical rockets, shows that it is most difficult to ensure durability of construction of the rocket exactly in the considered atmospheric section of free flight.

Loads druing the active section turn out to be essentially smaller. To ensure sufficient durability of the rocket in all sections of trajectory is not always possible, since this essentially shows on the total weight of the structure. Namely due to this circumstance the V-2 rocket turned out to be overloaded, since its body was designed with attention to lateral loads in the atmospheric section of free flight, where stabilization of the rocket occurs.

For rockets with separable head parts, calculation of loads in the atmospheric section of free flight is produced only for the head part. Sufficient durability of the head part is attained with a comparatively small lowering of weight indices of the rocket as a whole.

Brief Remarks on Calculation of Durability of Basic Carrier Elements in Ballistic Long-Range Rockets

Calculation for rockets on durability has that specific peculiarity that in it, it is necessary to consider not only change of loads acting on it in the trajectory, but also change of temperature of structural elements. Depending upon the flight speed the flow temperature near the surface of the rocket changes, and heating of the body can occur. The temperature of structural elements of the rocket depends on the local flow temperature and on conditions of thermal insulation, thermal radiation and heat transfer inside the body. Good thermal insulation and heightened heat transfer inside the body lower the temperature of the structure.

Calculation of the temperature of carrier joints in the rocket pertains to a number of problems, solved at present only approximately, inasmuch as reliable initial data are insufficient — condition of heat transfer, constants of radiation and heat transfer. The problem is complicated still more by the fact that heat flow is transient.

Results of theoretical calculation of temperatures of body elements are validated and supplemented by results obtained on flying tests with the help of a system of telemetric control (see Chapter X). Wire transducers, the resistance of which changes with change in temperature are established in a number of characteristic points of the rocket body. Indications of transducers are transmitted to Earth through radio channels. As a result of experimental rocket launchings, definite data is obtained, which can be used both in the process of further adjustment of the given structure and for more precise determination of initial information in designing new machines.

The temperature of structural elements of a long-range rocket attains very high values in flight. Especially high temperatures are developed in the atmospheric section of free flight.

The mechanical characteristics of the material are lowered with increase of temperature. As an example, Fig. 8.26 shows the dependenc of the yield point and ultimate strength of an aluminum-magnesium alloy on the temperature. Already at a temperature of about 400°C this alloy practically completely loses its durability properties. To a lesser degree, on temperature depend mechanical indices of the steel, especially of special alloyed steel.

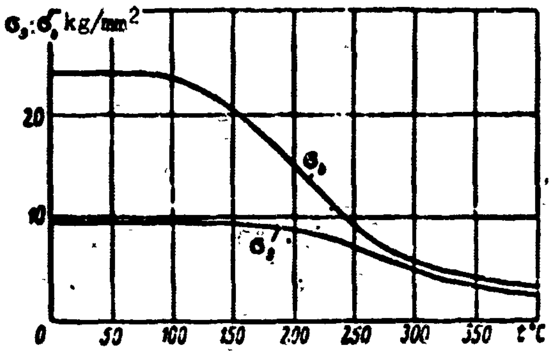

Fig. 8.26. Dependence of ultimate strength and the yield point of an aluminum-magnesium alloy on temperature.

In accounting for durability in a rocket it is necessary, obviously, to originate from the "instantaneous" mechanical characteristics of material, i.e., from characteristics corresponding to temperature in a given moment.

A graphic example illustrating the above is calculation of durability of a carrier tank (Fig. 8.27) in a meteorological rocket.

The tank is under the action of internal pressure, which changes along the generator by piecewise-linear law, shown on Fig. 8.27a. Above the liquid surface the pressure is constant and equal to feed pressure p_1. Below the liquid level (at a depth h) the internal pressure is equal to

$$p_{\text{вн}}=p_1+n_x\gamma h,$$

where γ is the specific gravity of the fuel component;

n_x is the axial overload.

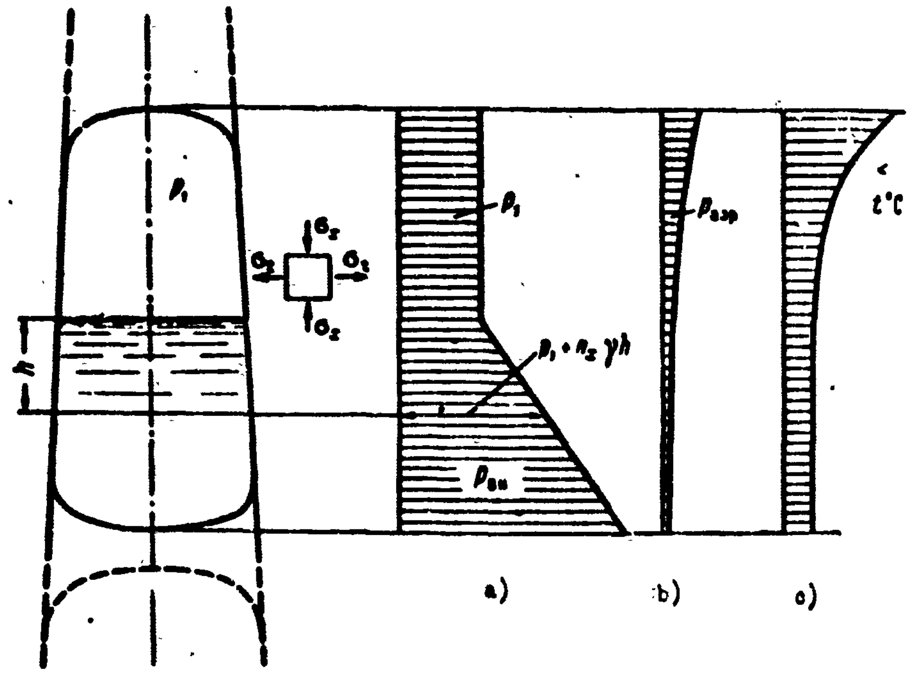

Fig. 8.27. Calculation of durability of the carrier tank in a ballistic long-range rocket. [аэр = aero]

From without, the pressure of aerodynamic forces p_{aero} acts on the tank. The approximate law of change of p_{aero} along the length of the tank is shown on Fig. 8.27b. The internal pressure is essentially greater than external. Therefore, in the peripheral direction the tank is stretched, and the appearing peripheral stress, as it is known from resistance of materials, is equal to

$$\sigma_t = \frac{(p_{вн}-p_{аэр})D}{2\delta},$$

where D is the diameter, and δ is the thickness of the tank wall. An axial compressing force N_x appears in cross sections of the tank, the dependence of which on parameters of rocket motion was considered above. The corresponding axial stress will be

$$\sigma_x = -\frac{N_x}{\pi D\delta}.$$

Thus, a biaxial state of strain appears in the tank wall, shown on Fig. 8.27.

According to the theory of durability of maximum tangential stresses

$$\sigma_{экв} = \sigma_1 - \sigma_3 = \sigma_t - \sigma_x$$

or

$$\sigma_{экв} = \frac{(p_{вн} - p_{нар})D}{2\delta} + \frac{N_x}{\pi D\delta}.$$

[экв = equ]

If it turns out that the normal force N_x will not be compressive, but stretching, then stress σ_x will change sign. In this case the greatest stress $\sigma_1 = \sigma_t$, and least $\sigma_3 = 0$. Then

$$\sigma_{экв} = \sigma_t = \frac{(p_{вн} - p_{нар})D}{2\delta}.$$

The reserve of durability is

$$n_s = \frac{\sigma_s}{\sigma_{экв}},$$

where σ_s is the yield point of the material, depending on temperature. The approximate law of change of wall temperature height of the tank for a certain moment of time is shown on Fig. 8.27c.

The reserve of durability n_s depends on the moment of time and on the location of the considered point along the height of the tank. Here it is impossible beforehand to indicate the most dangerous section. In the lower part of the tank internal pressure is the biggest. But here as a result of intense transfer of heat into the liquid, wall temperature will be smaller, practically equal to the temperature of the fuel component. Therefore, the value of σ_s remains very high, corresponding to normal temperature conditions. In the upper part of the tank the pressure is essentially lower; therefore, σ_{equ} is small. However, strong heating of the wall by aerodynamic flow and corresponding reduction of the yield point has place here.

In the process of structural calculations the most dangerous sections of the tank and the most dangerous moments of time of its load are revealed from consideration of a number of sections along the tank length in different moments of flight time. Depending upon the obtained results of calculation the thickness of the tank wall may be designated.

Problems appearing in connection with durability of rockets are often very unique and difficult to solve. Their detailed consideration goes far beyond the limits of this course.

CHAPTER IX

BASIC PRINCIPLES OF STABILIZATION, CONTROL AND GUIDING OF ROCKETS

1. Methods of Stabilization and Control of the Rocket

Stability and Stabilization

Certain motion of the rocket selected in advance under action of regular foreseen forces is called undisturbed motion of a rocket. Thus, for instance, undisturbed motion will be that of a ballistic rocket under action of tractive force and aerodynamic forces, calculated on the basis of laws known to us. Motion of a zenith rocket during guiding to a target should also be considered undisturbed, if the flight trajectory is predetermined.

Besides regular forces, accidental external influences can act on a rocket in flight, such, for instance, as gusts of wind, unforeseen changes of engine thrust and so forth. Under action of such forces the rocket will have motion, called perturbed motion.

Let us assume that the influence of accidental forces is small, and perturbed motion differs little from undisturbed. If after ceasing of action of perturbing forces, perturbed motion of the rocket becomes undisturbed, then this undisturbed motion of the rocket is called stable. If, however, after ceasing of action of external perturbation the motion of the rocket does become undisturbed, then such motion will be unstable (Fig. 9.1).* The idea of stability and instability of motion is applicable, naturally, not only to rockets, but also in general to motion of any body.

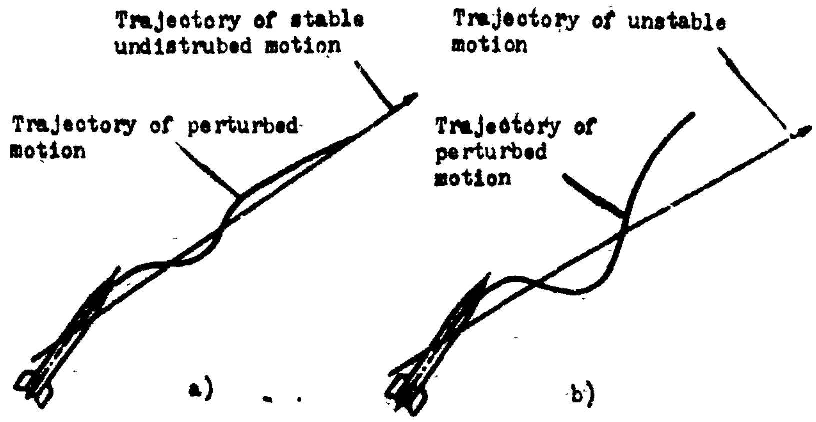

Fig. 9.1. Stable (a) and unstable (b) motion of a rocket.

It is clear that in those cases when we want to control motion, it is necessary in the first place to eliminate the undesirable influence of accidental external perturbing influences and thereby ensure stability of motion.

In real conditions of flight a rocket is subjected to the continuous influence of different accidental perturbing factors. In these conditions perturbed motion of the rocket in the whole flight time will differ from the initial undisturbed motion. Thus, in real conditions of flight, motion of the rocket only approximately corresponds to that given. The rocket continuously has perturbed motion, the parameters of which oscillate near the parameters of stable undisturbed motion. If deflections from the given motions are small,

*In the practice of investigation of rocket stability, motion conditionally is considered stable also in the case when the rocket, after ceasing of action of perturbation, returns not to undisturbed motion, but to a certain other motion differing little from it.

then practically one may assume that motion of the rocket is stable.

In connection with the above, determination of stability of motion mentioned above may be modified. Under stable motion it is possible to understand such whose parameters during action of perturbing factors on the rocket are changed in the final result within the limits of given allowance. In brief, during stable motion external accidental perturbations do not evoke great changes in parameters of motion.

The limits of allowed changes in the law of rocket motion are determined by a large number of considerations. The main ones are the requirement of accuracy of guiding to the target and the condition of preservation of controllability, inasmuch as at excessively large deflections from the given law of motion, the control system is not always able to manage its own problem.

It is necessary to say that motion, stable with respect to one parameter, can appear unstable with respect to another. Therefore, in examining of stability of motion, especially stability at allowed deflections, we must define what types of motion are being discussed. Thus, rocket motion is determined by the law of motion of its center of gravity and angular shifts of the whole rocket as rigid unit about its center of gravity.

The idea of stability and instability in the sense of angular shifts is replaced usually by the idea of angular stabilization and destabilization of the rocket. If obtaining an angular deflection, and after ceasing of the perturbing influence the rocket returns to its undisturbed position, then the rocket is called stabilized. If this does not occur, then the rocket is called destabilized.

In many cases questions of angular stabilization can be considered independently of stability of motion of the rocket's center of gravity. The rocket can appear unstabilized, and deviations in the law of motion of its center of gravity might not emerge beyond the limits of what is permissible for instance, during free flight beyond the borders of the atmosphere.

Peculiarities of Perturbed Motion and Stabilization of Unguided Rockets

In Chapter VII we already discussed stabilizing aerodynamic moment and aerodynamic stabilization. If in reduction of all aerodynamic forces to the rocket's center of gravity the static moment is directed in the direction of decrease of the incidence angle, then the rocket is called statically unstable. If, however, the moment is directed in the opposite direction, the rocket will be statically unstable.*

In the same place, in Chapter VII, the idea was introduced of center of pressure as a point of crossing of resultant aerodynamic forces with the rocket axis. In the case when the center of pressure is behind the center of gravity, the rocket is statically stable. If the center of pressure is ahead of the center of gravity, the rocket is statically unstable.

The relationship of the distance between the center of pressure and the center of gravity to the length of rocket is called the reserve of static stability.

With a positive reserve of stability, the rocket, obtaining an angular deflection, turns by aerodynamic forces into its initial position. When the angle of incidence becomes equal to zero, the restoring moment will also become zero. But since the rocket preserves angular velocity, motion will continue in the opposite direction, an

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*Frequently applied in aircraft construction, the expression "static stability" is equivalent to the idea of "aerodynamic stabilization." Static stability usually is understood as stability without interference of the control system, and dynamic — stability of motion ensured by introduction of a control system, where the type — automatic or nonautomatic — is of no matter.

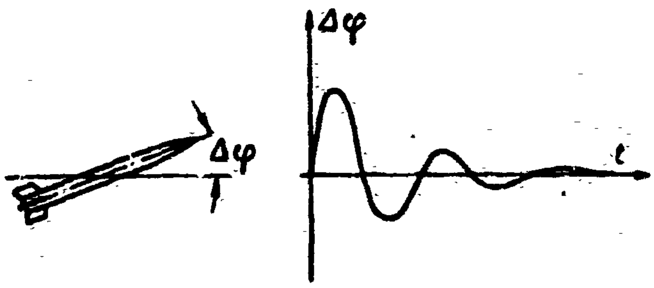

Fig. 9.2. Attenuation of short-period oscillations.

angle of incidence of the reverse sign will appear, and angular oscillations of the rocket will appear. Due to presence of damping moments, the oscillations attenuate (Fig. 9.2).

Oscillations of such kind are called short-period. Their frequency depends on the reserve of static stability and the rocket's moment of inertia. The greater the reserve of stability and the less the moment of inertia, the higher the frequency of these oscillations.

However, excessive increase in the reserve of static stability can give negative results. A guided rocket of too great reserve of stability becomes difficult to control. Unguided rockets with a large reserve of stability, as we will see below, can have much dispersion.

The described short-period angular oscillations can occur, as is easily understood, about two transverse axes.

Also peculiar to a rocket, especially winged, is the presence of so-called long-period or phugoid, oscillations. Their appearance is easiest to explain considering the horizontal uniform flight of a winged missile, the engine thrust of which on the average is balanced by the drag force at a constant angle of attack.

Let us assume that by some cause the tractive force is briefly increased. As a result, the flight speed increases, and lift, which becomes greater than the force of weight is increased. The winged missile starts to rise, and as a result loses speed. There comes a time when lift again becomes equal to the force of weight, but the aircraft, possessing a vertical velocity, continues to rise by inertia.

Further ascent will be connected with decrease of lift. The aircraft will begin to descend. So phugoid oscillation appears. The trajectory of such an aircraft is shown on Fig. 9.3.

Fig. 9.3. Phugoid, or long-period oscillation of a winged missile.

Phugoid oscillations have a large period, which can even exceed the flight time in the active section. The magnitude of lift for ballistic rockets is comparatively small; therefore, phugoid oscillations are not taken into account for the basic types of rockets.

Different oscillation modes of the rocket are not independent. During transverse angular oscillations due to periodic appearance of lift, there appears the transverse oscillations of the center of gravity near the undisturbed trajectory.

If a rocket moves with an incidence angle, then at the appearance of a yaw angle the rocket banks. This one may see well from Fig. 9.4, which shows a rocket receiving a deflection from its course — angle ψ. The right stabilizer receives an increase in speed, and the ascent force increases on it, at the same time in which it decreases on the left. A moment appears, turning the rocket along the longitudinal axis, and the rocket banks.

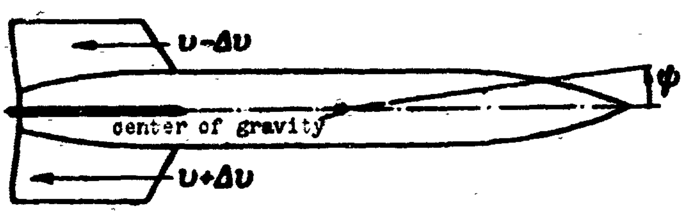

Fig. 9.4. Appearance of bank at deflection of the rocket off course.

Oscillation of the rocket about different axes can appear connected owing to gyroscopic influence of revolving masses in the rocket's steering controls, for instance the turbine of the turbopump unit.

The problem of stabilization consists of the fact that all oscillations of the described type should be limited in their amplitude and attenuating as fast as possible.

During the simplest static stabilization the influence of brief accidental perturbing influences on rocket flight is excluded, but the influence of constantly acting accidental factors remains, which is impossible to anticipate beforehand. To a number of them pertain first of all technological errors.

The rocket cannot be prepared absolutely exactly. The rocket engine always has a certain eccentricity, and the tractive force does not pass through the rocket's center of gravity. At the expense of this appears a moment, turning the rocket aside. The external outlines of the rocket also cannot be followed ideally; therefore, aerodynamic forces also possess certain eccentricity. The stabilizers themselves, intended for guarantee of stability of motion, always have errors in construction. All these factors lead to deflection of the rocket from the given trajectory.

Furthermore, aerodymamic stabilization cannot have an effect during flight beyond the borders of the atmosphere.

There is another possibility to ensure stability of motion of an unguided rocket. It consists in signaling rapid rotation about the longitudinal axis to the rocket. Such a method, as it is known, is used widely for stabilization of artillery missiles. Here the rocket preserves the direction of its axis owing to gyroscopic effect.

The method of stabilization by rotation is applied, for instance, for small solid-propellant rockets. Such rockets, as we already know, are called turboreactive missiles (abbreviated TAR). Large rockets cannot be stabilized in such a way. At fast rotation it is difficult to ensure the necessary durability of a large rocket, not even mentioning that it is difficult to control it.

A better, but, indeed, immeasurably more complicated method than the two described, is installation on the rocket of an automaton of stabilization, simultaneously ensuring motion of the rocket along a given trajectory.

Stabilization with the Help of an Automaton

At first stabilization automata were created to guarantee a constant course of motion for naval torpedoes. Subsequently with development of aviation and appearance of the necessity of automatic piloting of aircraft, automatic pilots were created, the basic principles of which during the second world war were transfered to stabilization automata of rockets and winged missiles.

The general principle of work of similar automata is easiest to comprehend by an example of a simple course automaton for an aircraft (Fig. 9.5).

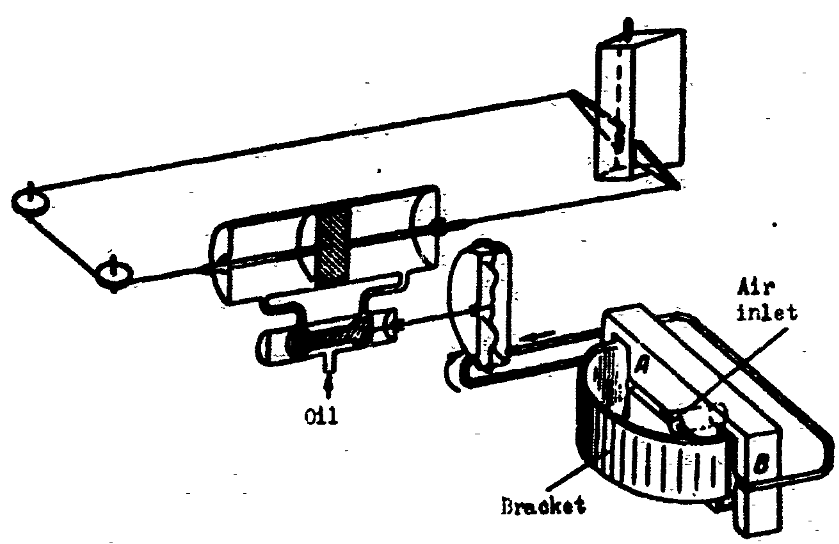

Fig. 9.5. Diagram of a simple course automaton.

The working component of the simplest course automaton is a cylinder with a mobile piston, with the help of which the steering rudder of the aircraft turns. Pressure on both sides of the piston in the cylinder is regulated by a distributor valve, which moves by a sensitive diaphragm. In order to pass oil into the right or left cavity of the cylinder, it is necessary to move the valve correspondingly to the right or to the left. A shift of the regulating valve does not require noticeable effort; therefore, a thin sensitive diaphragm turns out to be capable of controlling turns of the steering rudder.

Pressure in both cavities of the diaphragm box results from a pneumatic network of the aircraft through holes, half covered by flaps. The flaps are made of a bracket rigidly fastened to a gyroscope and therefore, remaining motionless during turning of the aircraft. During flight on course both holes are covered identically, and pressure in both cavities of the diaphragm box is equal. The steering rudder is in neutral position.

Let us assume now that the aircraft somehow started to deviate from a fixed course, for instance to the left. Hole A starts to be closed, and hole B will be opened. Pressure in the right cavity of the diaphragm box will be increased, and the rod of the valve will be displaced to the left. This will entail entering of oil into the forward cavity of the actuating mechanism (steering machine). The rudder will turn counterclockwise and the aircraft will start to turn to the right, i.e., return on course.

One would think that the described automaton completely solves the problem of course stabilization. However, this system possesses a fundamental deficiency.

Let us see what motion of the aircraft will occur further.

When the aircraft takes the needed direction, the flap, diaphragm and valve will occupy a neutral position, but the piston of the

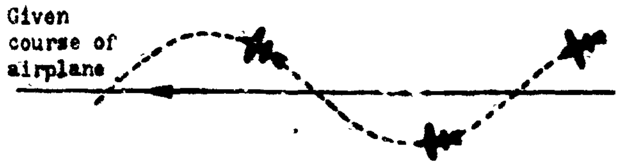

Fig. 9.6. Character of motion of an aircraft with control by a simple course automaton.

actuating mechanism and rudder will remain in the former position; and consequently, the aircraft will continue turn to the right. Only when the aircraft is turned to the right and the flap covers hole B and open hole A and the valve moves to the other side, will the steering control turn back and turn the aircraft to the left. Thus, oscillatory motion appears — roving along course. The aircraft will "zig-zag" (Fig. 9.6).

We could observe a similar picture if the driver of a motor vehicle, turning to the right, starts to recover from the turn only when the machine is completely turned to the right. If the driver repeats this error further, i.e., every time late in setting the wheels in neutral position, the machine would also zig-zag. To eliminate such motion, the driver should recover from the turn of the steering wheel still before the machine takes the needed direction.

It is necessary, obviously, that also the course automaton is controlled by motion of the aircraft, recovering the steering control from the deflected position before the moment when the aircraft returns on course. A feedback serves this purpose. As a feedback in the widest sense of the word is understood the element of a control system, connecting a subsequent link with one of those preceding in such a way so that signals passed into the control system, comform with results which this system gives.

Fig. 9.7 shows the same course automaton, but with feedback from the rudder. Here the air distributor AB is no longer rigidly connected with the aircraft and can turn about its vertical axis. This turning is connected with turning of the rudder.

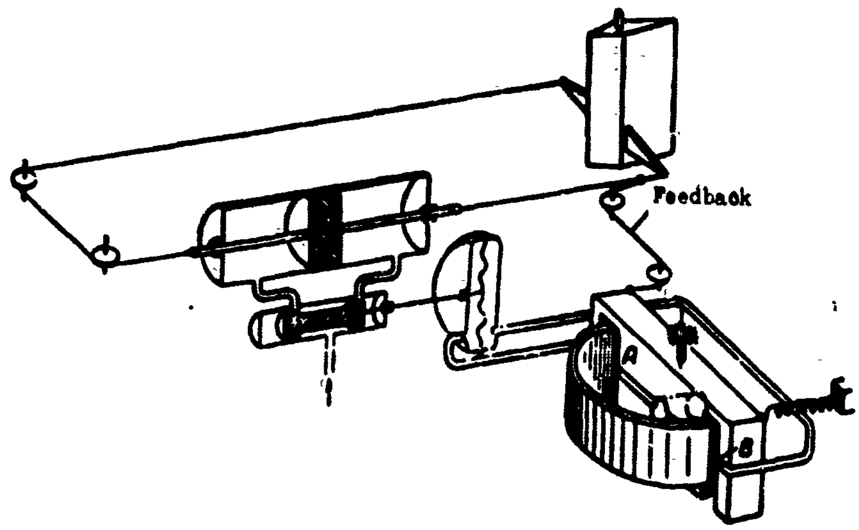

Fig. 9.7. Diagram of a course automaton with feedback from the steering control.

Let us see how return of the aircraft on course will occur now.

Again we will assume that by some cause the aircraft turned from its course to the left. Then in the above described sequence the elements of the mechanism will work and the rudder will turn counterclockwise. Simultaneously, through feedback the rudder will turn the distributor of air and set it in neutral position. The aircraft will turn to the right; as a result, hole A will start to open and hole B to close; thus the rudder will turn in the opposite direction. When the aircraft returns on course, the steering control will stand in neutral position, while together with it the distributor block of pressures with holes A and B. Return of the aircraft on course will occur without roving (Fig. 9.8).

Fig. 9.8. Character of motion of an aircraft during control by course automaton with feedback.

The essential peculiarity of such feedback, called rigid feedback, is the fact that it ensures (with a certain delay) deflection of the steering control proportional to the angle of displacement of the aircraft from the course. In the absence of rigid feedback the steering control will deviate up to a limiting possible position in the course of the total time of stay of the aircraft on an incorrect course.

As will be shown further, to ensure flight stability of ballistic rockets during flight in a vacuum is possible only in the presence of so-called flexible feedback, when to the control organs – steering machines – passes not only a signal proportional to the angle of displacement of the rocket from the direction, but also signals proportional to speeds and acceleration of this deflection. In this case at fast growing deflections of the rocket, the stabilizing organs react to deflections more effectively and limit these deflections in the total time of motion. Stabilization systems of such kind are applied on ballistic rockets and certain other pilotless aircraft.

The work of the course automaton was described above. It is clear that for a rocket control automaton should be carried out not only for course, but also for pitch and bank.

The task of the stabilization automaton on the rocket is not limited only to pure stabilization. Inasmuch as the stabilization automaton ensures motion of a rocket by a definite program, it is also the acting organ.

A rocket supplied by a stabilization automaton is called a guided rocket. If its control from the moment of launching is not connected with the earth, then such a control system is called autonomous. If, however, communication with land is continued after launching, for instance by radio channel, then this control system is called nonautonomous.

The autonomous system has that evident deficiency that after the moment of takeoff further interference of motion of the rocket is excluded. It is impossible, in particular, to correct motion of the rocket if it starts to deviate from course.

Transmission of signals by radio channel (so-called radio correction) allows introduction of correction in the rocket motion and to a significantly higher accuracy of hit. Radio control for zenith rockets allows direct guidance of the zenith rocket at an enemy aircraft and a sharp increase in the percent of hits as compared to unguided rockets.

However, radio control has an essential deficiency that the action of sufficiently powerful radio interference can bring to nought the utility of nonautonomous control.

The problem of radio control is one of the basic problems of contemporary rocket technology. Together with development of radio control methods of creating radio interference are being developed as ways of counteraction. Development of the latter will entail development of methods of creating interference resistent control instruments. All these questions are the subject of a special area of radio engineering.

2. The Gyroscope and Its Application

Properties of the Gyroscope

A massive, precisely balanced flywheel, revolving with great angular velocity, is called a gyroscope.

The unique behavior of a gyroscope is well-known; the rapidly revolving flywheel possesses the ability to preserve more or less stably the direction of its axis and manifests unexpected "irresponsiveness" when an attempt to turn this axis is made.

The gyroscope is the basic element of contemporary automatic pilots and a majority of stabilizing devices. Therefore, consideration of the work of stabilization automata should be started from properties of the gyroscope.

The theory of the gyroscope is a particular problem of the general theory of motion of a body with one rest point. For such a body a a motion equation (Euler equation) can be formed, not usable in the common form, but allowing us to give answers to certain questions of motion, including the question of motion of a symmetric rapidly rotating disk. The latter question is the content of the so-called gyroscope theory. We will not dwell on this theory in detail and will be limited by consideration of chiefly its qualitative aspect.

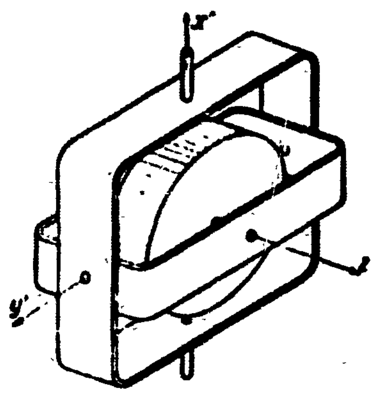

Fig. 9.9. Gyroscope in a Cardan joint suspension.

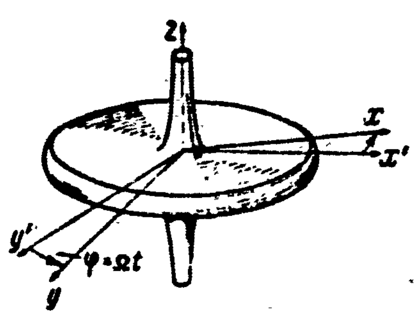

Fig. 9.10. Location of connected gyroscope axes.

We will first of all consider that a gyroscope is a symmetric body with one rest point and that this point is its center of gravity. Such a mounting can be made with the help of a Cardan suspension (Fig. 9.9). In these conditions the motion of the gyroscope may be represented as rotation with an angular velocity Ω near a certain instantaneous axis passing through the center of gravity.

We will take three mutually perpendicular axes x, y, z, rigidly joined to the gyroscope (Fig. 9.10). We will direct axis z along the axis of the gyroscope's main rotation. The two other axes will pass through the center of gravity of the gyroscope perpendicular to axis z. Axes x, y, z will be the principal axes of inertia of the gyroscope.

The vector of instantaneous angular velocity $\overline{\Omega}$ may be separated on these axes into components $\overline{\omega}_x$, $\overline{\omega}_y$, $\overline{\omega}_z$. For a rapidly rotating gyroscope the absolute value of ω_z is immeasurably larger than magnitudes ω_x and ω_y. The latter for a balanced gyroscope in general are equal to zero and appear, as we will see subsequently, only when external forces start to act on the gyroscope. Inasmuch as ω_x and ω_y are small, $\omega_z \approx \Omega$.

Vector components of the main angular momentum $\overline{N}$ will be $A\overline{\omega}_x$, $B\overline{\omega}_y$, $C\overline{\omega}_z$ where A, B, C are moments of inertia of the gyroscope relative to axes x, y, z. From conditions of symmetry, $A = B$.

One should note an evident fact: vector $\overline{N}$ in general does not coincide with vector $\overline{\Omega}$ in the direction. Only in virtue of the fact that for the gyroscope ω_x and ω_y are very small in comparison to ω_z, one may assume that vectors $\overline{N}$ and $\overline{\Omega}$ in direction are very close to one another. This significantly facilitates the problem and allows us, watching the behavior of vector $\overline{N}$, to immediately determine (as coinciding with it) the direction of vector $\overline{\Omega}$.

The basic relationship determining rotational motion of a body is the law of change of angular momentum

$$\frac{d\overline{N}}{dt}=\overline{M}, \tag{9.1}$$

where $\overline{M}$ – vector of moment acting on the body,

$\overline{N}$ – vector of angular momentum.

We will introduce a system of semiconnected central axes $x'y'$ (see Fig. 9.10). These axes are rigidly joined with axis z and turn together with it, not being connected with the gyroscope, i.e., not participating in its main rotation with speed Ω. For the gyroscope shown on Fig. 9.9, axes x' and y' coincide with the axes of the rings of the Cardan joint suspension.

Let us now see what motion the gyroscope axis z will make if we try to turn the gyroscope, applying to it a moment $\overline{M}$ with respect to an axis x' through the suspension frame (Fig. 9.11).

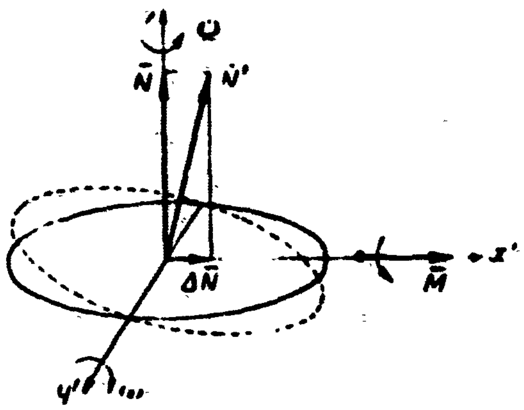

Fig. 9.11. Conclusion on the expression for angular velocity of precession.

Before moment $\overline{M}$ is applied, the gyroscope has an angular momentum

$$\overline{N}=C\overline{\Omega}.$$

According to equation (9.1), in time Δt the change in angular momentum is

$$\Delta\overline{N}=\overline{M}\Delta t.$$

Vector $\Delta\overline{N}$ coincides with the direction of vector $\overline{M}$. Adding $\overline{N}$ and $\Delta\overline{N}$, we get a new vector $\overline{N}'$ for the angular momentum. Due to the smallness of $\Delta\overline{N}$, vectors $\overline{N}$ and $\overline{N}'$ differ from each other only in direction, not differing by modulus. Thus, we see that as a result of the action of moment $\overline{M}$, vector of angular momentum $\overline{N}$, not changing its magnitude, turns at an angle $\Delta N/N$ with respect to axis y'.

It was already mentioned above that for a gyroscope the vector of angular momentum almost coincides with the instantaneous axis of rotation; consequently, we can say that under the action of moment M the axis of rotation of the gyroscope during the time Δt turns at the same angle $\Delta N/\overline{N} = M\Delta t/\overline{N}$ with respect to axis y'. Here the angular velocity of rotation of the gyroscope axis is

$$\omega=\frac{M}{N}=\frac{M}{C\Omega}. \qquad (9.2)$$

The described motion of the gyroscope with respect to an axis y' is called precessional motion, and angular velocity ω is called precession velocity.

The rule which the behavior of the gyroscope obeys at application of an external moment to it can be formulated in the following way; if to a gyroscope, having three degrees of freedom, we were to apply a moment with respect to an axis perpendicular to the principal axis of rotation, the gyroscope will start to turn so that the vector of main rotation Ω will move by the shortest way to the vector of moment M, tending to coincide with it (see Fig. 9.11).

In this law, as a matter of fact, is included all the "singularity" of gyroscopic behavior. The gyroscope axis turns not in the plane of the applied moment, but in a plane perpendicular to it. Furthermore, precessional motion in the presence of a constant moment is not accelerated. Speed ω increases only as long as the applied moment increases. At a constantly acting moment the angular velocity of precession remains constant, and after ceasing of action of the moment, precessional motion is ceased.

If the gyroscope is not given preliminarily rotation with a speed Ω, at application to it of an external moment, it would act as a usual body. From Fig. 9.11 it can be seen that in this case (at $\overline{N} = 0$) vector $\Delta\overline{N}$ would completely determine the motion of a flywheel, and under action of the moment the latter would revolve with acceleration with respect to an axis x', i.e., in the plane of action of the moment.

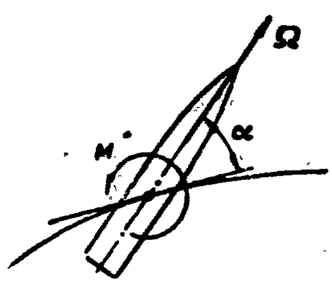

Fig. 9.12. Turbo-jet missile on trajectory.

Knowing the law of precession, it is possible to explain and to expect many phenomena connected with the gyroscopic effect.

Let us trace for instance, the behavior of a turbojet missile in flight.

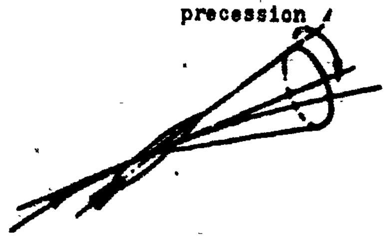

Fig. 9.13. Precession of turbojet missile.

The missile leaves the directrixes with a zero or almost zero angle of incidence. In subsequent motion, having a large angular velocity, it acts like a gyroscope and tends to preserve constant direction of the longitudinal axis. Due to bending of the trajectory an angle of incidence appears, but together with it an aerodynamic moment in a vertical plane (Fig. 9.12).

If a missile is statically unstable (as usually occurs for an unstabilized projectile), then the aerodynamic moment will be directed to an increase of the incidence angle, while the moment vector will be perpendicular to the plane of the diagram and directed at us. With axial rotation of the missile to the right, vector Ω is directed forward. Therefore, by the rule of precession, the missile will start precession to the right. With further motion the missile axis will describe a cone with motion to the right of the center of gravity near the trajectory (Fig. 9.13). For a statically stable missile, precession will be to the left.

Application of the Gyroscope

In the practice of navigational and stabilizing instruments, the gyroscope is used as a basis of support of the system of reference, i.e., as an element by which the control system determines the aircraft's angular orientation. Here the property of the gyroscope in preserving a constant direction of the axis of rotation is used first of all.

This universally recognized "property" is, however, conditional. We already know that under action of an external moment the gyroscope accomplishes precessional motion and does not maintain a constant direction of the principal axis of rotation.

On a gyroscope fixed in a Cardan joint suspension, moments of friction forces, appearing in suspension bearings are acting. These moments appear due to rotation of the gyroscope itself, and also as a result of turning of the external rings of the gyroscope about the rotor during oscillatory motion of the aircraft.

Upon the expiration of a more or less prolonged time, the axis of the free gyroscope noticeably deviates from the initial direction. Even with well made suspension bearings impermissible deflection of the axis occurs in just a few minutes of free work of the rotor. Therefore, it is clear that in the functioning of a gyroscopic instrument the position of the gyroscope axis should continuously be corrected: correction of the gyroscope, so to speak, should be introduced.

As the simplest and at the same time the most ingenious method of gyroscope correction we will consider the scheme of work of the simple gyro horizon.

The gyro horizon instrument is intended to show the pilot during blind flight of an aircraft the direction of the horizon. The sensitive element of this instrument, in this case the gyroscope, maintains constant vertical location of the rotation axis of the rotor. In pilotless flight apparatuses of such a gyroscope can serve as an indicator of the vertical.

Fig. 9.14 shows a diagram of the gyro horizon device.

The instrument consists of a gyroscope with the vertical axis fixed in a Cardan joint suspension.

The gyro rotor goes into rotation by a flow of air proceeding through the tube of axis 1 and striking the ribbed surface of the rotor. From the rotor box air emerges through four holes, 2 half covered by pendulum flaps 3, 4, 5 and 6.

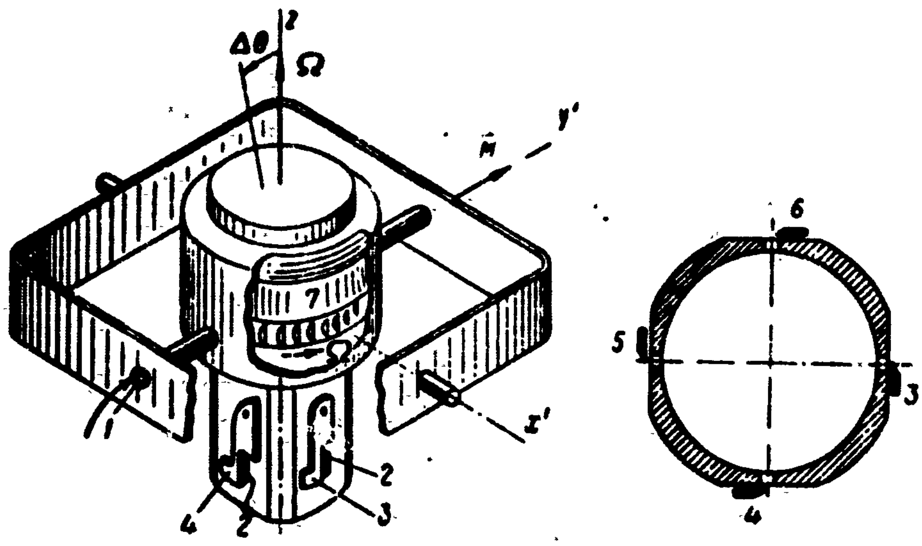

Fig. 9.14. Diagram gyro horizon device. 1 – tube, bringing in air; 2 – outlet for air; 3, 4, 5, 6 – flaps; 7 – rotor.

Let us see how correction is carried out. Let us assume that under action of friction forces in the bearings of the suspension axis y', the gyroscope axis turns with respect to axis x' at an angle $\Delta\theta$ (see Fig. 9.14). With such a turn, flap 3 will be opened, and flap 5 will be closed. The outgoing stream of air creates a reactive moment M with respect to axis y'. Here, by rule of precession the gyroscope begins to turn in such a manner so that vector Ω approaches vector $\overline{M}$ by the shortest way, and the vertical position of the gyroscope axis is restored. An analogous picture will take place if the gyroscope turns with respect to axis y'.

Besides the one described, there exist other schemes of correction.

Fig. 9.15 shows the fundamental diagram of a gyromagnetic compass, in which the gyroscope axis is oriented by a magnetic needle.

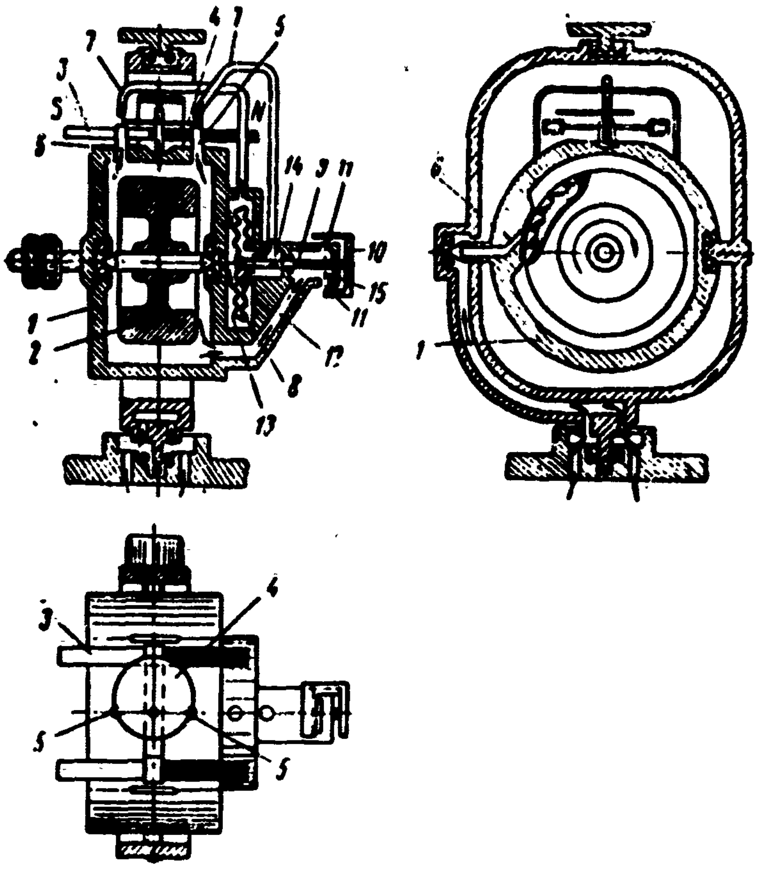

Fig. 9.15. Fundamental diagram of a gyromagnetic compass. 1 – rotor housing; 2 – rotor; 3 – turning magnetic needle; 4 – eccentric; 5 – nozzle; 6 – acceleration nozzle; 7 – receiving nozzle; 8 – diaphragm; 9 – diaphragm rod; 10 – flap; 11 – reactive nozzle; 12 – channel; 13, 14 – hermetic chambers; 15 – air chamber.

The turning magnetic needle 3 is on the general vertical axis with eccentric 4. This eccentric breaks the gas stream passing through the receiving nozzle 7. In connection with the fact that the magnetic needle of the compass has a small magnetic moment and is very sensitive to lateral loads, in the compass structure subsequent strengthening of the signal is applied. If the body, and consequently also the gyroscope axis deviated from the direction of magnetic needle, the eccentric will over one of nozzles 7 and will open the other. Then on the diaphragm 8 a difference of pressures will appear, and by means of the rod 9, the flap 10 will be moved, which in turn will cover one of the reactive nozzles 11 and will open the other. As a result a moment appears in the vertical plane, under the action of which the gyroscope will start precessional movement in the horizontal plane and will restore its orientation with respect to the magnetic needle.

In examining similar diagrams a natural question appears: is the gyroscope so necessary if it is necessary to correct it anyway with the help of a common pendulum or common magnetic needle? Is it not simpler to use a pendulum directly as an indicator of the vertical, and the magnetic needle as an indicator of course?

However, the fact is that the pendulum and needle of the compass possess very small mass, small inertia, and are strongly subject to different accidental influences. If, for instance, the pilot were to start to make a turn, the pendulum here becomes oriented in the direction of total acceleration, and an instrument deprived of a gyroscope will not show the true horizon. The gyroscope has great inertia, and a prolonged systematic influence is necessary, for instance prolonged turn, so that the gyro horizon, supplied by the gyroscope, gives noticeable error. Thus, the larger the angular momentum of the gyroscope $C\Omega$, the more inertial the gyroscope, and the more effective is its action.

Thus, the gyroscope finds application in instruments owing to its high inertia, low frequency of natural oscillations, and little susceptibility to influence of accidental perturbing forces. In distinction from the magnetic needle and pendulum, the gyroscope may be used in instruments such as the power element, inducing motion in certain mechanisms.

Till now a freely suspended gyroscope with three degrees of freedom was represented as the basic element of the system of reference of angles or, in other words, as an instrument for measurement of angular deflections of the aircraft. Inasmuch as the gyroscope maintains a constant direction, but the aircraft receives angular displacements under the action of external forces, angular error appears, the measurement of which gives the basic signal impelling the stabilization automaton into action (see, for instance, the above described diagram of a simple course automaton).

For better regulation, signals connected not only with the angular deflections but also with angular velocities are introduced into the stabilization automaton. As an instrument for measurement of angular velocity in turning of the aircraft a gyroscope can again be used, but not freely suspended and with two degrees of freedom. A diagram of installation of such a gyroscope is shown on Fig. 9.16.

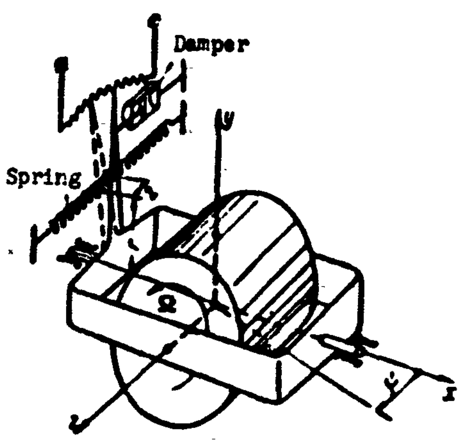

Fig. 9.16. Two-stage gyroscope.

Let us suppose that an aircraft turns in the plane x - z. But to rotation with angular velocity $\omega = \dot{\psi}$ with respect to axis y corresponds, as we know, the moment M with respect to axis x:

$$\dot{\psi} = \frac{M}{C\Omega}.$$

If the frame of the gyroscope is connected by springs, then

$$M = rKz,$$

where K — rigidity of springs;

z — displacement of the point of their bracing;

r — arm.

Excluding moment M, we get

$$z = \frac{C\Omega}{Kr}\dot{\psi}.$$

Thus, displacement of the bracing point of the springs and in general any point of the frame turns out to be proportional to the angular velocity of turning of the aircraft.

For fixing of appearing displacements either electrical transducers of resistance (see Fig. 9.16), or pneumatic gauges with jet tubes are applied, as this was shown in the diagram of a gyrocompass. Independently of applied transducers, during deflection of an aircraft from a given direction a summational signal u is given to the stabilization automaton, constituting a sum of two signals – from a three-stage gyroscope, proportional to angle ψ, and from a two-stage, proportional to angular velocity $\dot{\psi}$:

$$u = A\psi + B\dot{\psi}. \tag{9.3}$$

The relationship between coefficients A and B is determined by the structure of the stabilization apparatus and is a parameter of its adjustment.

3. Gyro Instruments and Acting Organs of a Stabilization Automaton of a Long Range Rocket

Location and Purpose of Steering Rudders is of a Long-Range Rocket

The structure and work of a stabilization automaton of a long-range rocket will be considered in an example of autonomous control of a V-2 ballistic rocket.

A rocket is supplied with four pairs of rudders. The location of the rudders is shown on Fig. 9.17. Rudders I, II, III, IV are gas current rudders; I', II', III', IV' are air rudders.

Conventional numeration of rudders is conducted clockwise if one were to look at the rocket tail. Figure I designates the lower rudder, lying in the plane of the program trajectory. Since in undisturbed flight the rocket does not revolve around its longitudinal axis, this orientation of the rudders is kept for the whole powered flight section.

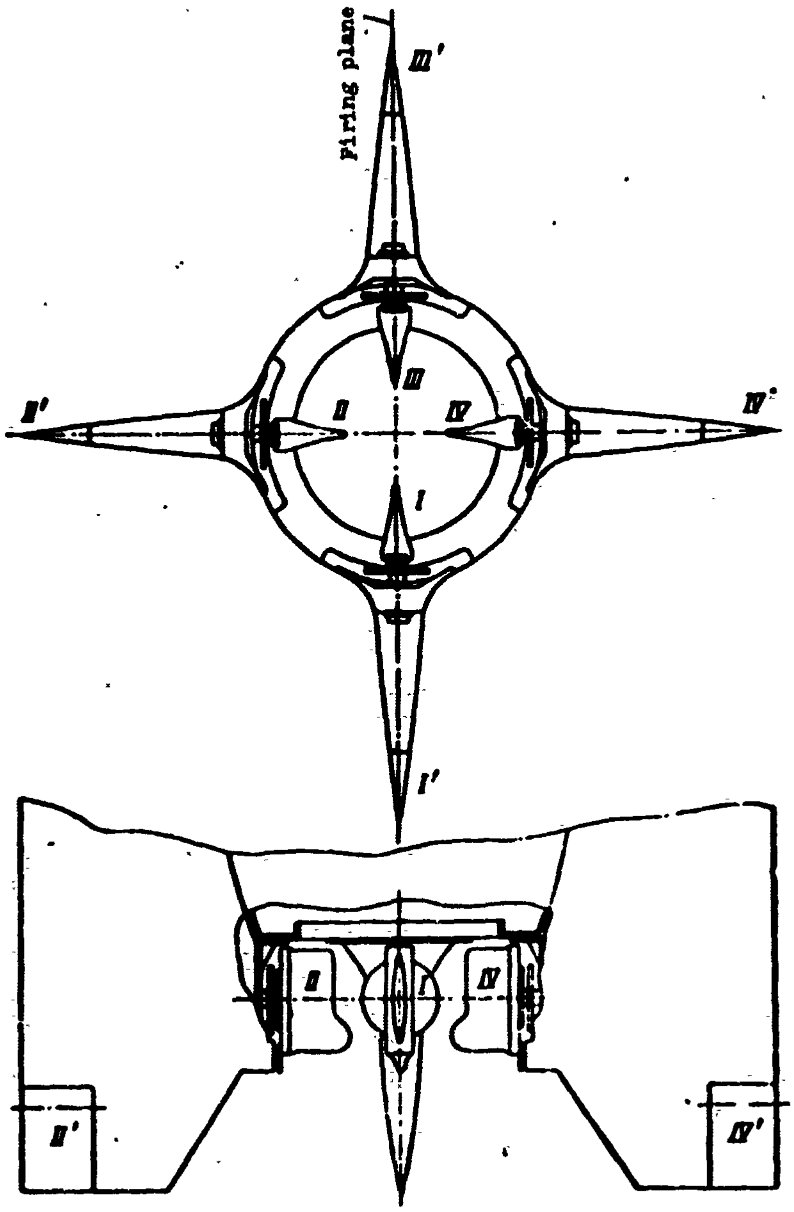

Fig. 9.17. Diagram of location of gas current and air controls for a V-2 ballistic long-range rocket.

Rudders II and IV ensure stability of rocket flight by pitch. Turning rudders II and IV synchronously by a definite law, we have the possibility to assign the rocket any program.

The task of rudders I and III consists in holding the rocket on a given course, i.e., not to allow it to deviate from the plane of firing. This is attained by a synchronous turn of rudders I and III in one direction.

Stabilization of the rocket's bank is carried out by the same rudders I and III. If the rocket gets a small bank, rudders I and III will turn in opposite directions and will thereby create a restoring moment about the longitudinal axis.

The dimensions of gas current rudders are determined by those control forces which are necessary to obtain from them under conditions of stable flight of the rocket in the powered-flight trajectory. Increase of dimensions of gas current rudders leads to noticeable increase of weight and increased losses of the engine's tractive force. A decrease is connected with a danger of loss of controllability. Introduction of air rudders allows a decrease in the dimensions of gas current rudders and thereby results in reduction of rocket weight. At the same time air rudders are able to work only at sufficiently great flight speed and in comparatively dense layers of atmosphere, i.e., at a sufficiently high speed pressure. Therefore, they have only auxiliary functions.

A consideration of work of the stabilization automaton of a ballistic long-range rocket will be started from a description of the arrangement and action of gyro instruments.

Gyro Horizon

The first gyroscopic instrument, called "Horizon," is intended for stabilization of the rocket's pitch angle. The same instrument assigns the rocket a program of pitch angle change. Consequently, signals from this gyroscope have to influence rudders II and IV.

The structure of "Horizon" is shown on Fig. 9.18.

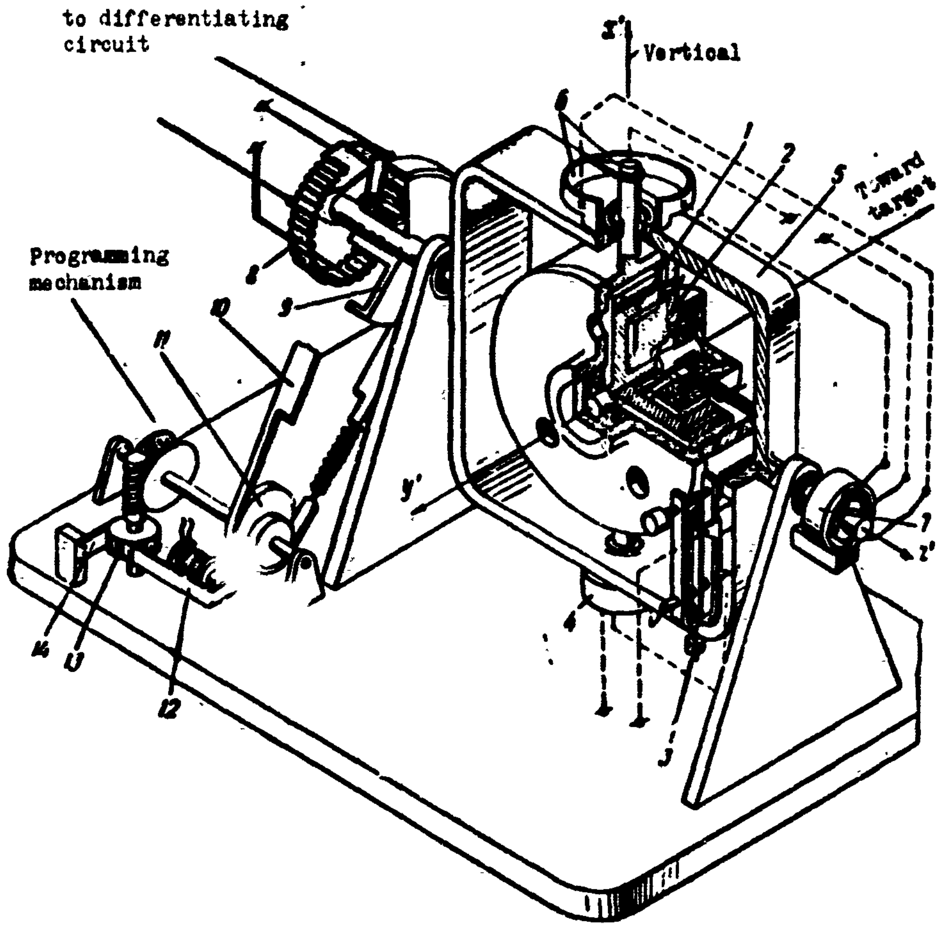

Fig. 9.18. Structure of gyro horizon. 1 — gyro rotor; 2 — winding of stator; 3 — pendulum; 4 — electromagnet; 5 — frame of gyroscope; 6 — contact device; 7 — electromagnet; 8 — potentiometer; 9 — pulley of potentiometer; 10 — belt of program mechanism; 11 — eccentric of program mechanism; 12 — bracket of step motor; 13 — ratchet wheel; 14 — stopper.

A gyroscope is placed in a Cardan joint suspension so that the axis of rotation of the rotor is horizontal and lies in the plane of the programmed trajectory.

Rotor 1 is at the same time an anchor of the asynchronous engine, the winding of the stator of which if fed by alternating current with a frequency of 500 cycles per second. The rotor is put into rotation for several minutes before start of rocket and takes the needed orientation in space automatically with the help of a pendular correcting device.

If axis x' (see Fig. 9.18) will deviate from the vertical, the pendulum 3 will make contact and a signal of the needed direction will be given to the electromagnet 4. The electromagnet will create a moment about the vertical axis x' which, as we already know, will evoke precession of the gyroscope about the horizontal axis z'. This precession will continue as long as the moment with respect to axis x', does not become zero, i.e., until the contact of the pendulum 3 is not opened.

At deflection of axis x' in the other direction, the pendulum 3 will make contact on the other side. Here the electromagnet will create a moment of opposite sign.

Orientation of the rotor with respect to the frame 5 is carried out with the help of an analogous device. If the rotor turns with respect to axis x', then one of the contacts 6 will be made and a signal will be given to the electromagnet 7. Because of the action of this electromagnet precession appears near axis x', and the needed orientation will be attained.

The correction system of a gyroscope works before the moment of start. After launching, the correction system is disconnected, and the gyroscope becomes free. During the time of controlled flight the axis of the uncorrected gyroscope does not succeed to depart noticeably from its given direction.

If the longitudinal rocket axis in flight will deviate in the plane of firing from a given direction, potentiometer 8 connected with pulley 9 and, consequently, with the rocket body, will turn together with rocket relative to the constantly oriented gyroscope and, thus, an electrical signal will be obtained, which, passing through a system of amplifier-converters will be transmitted to the acting organs – the rudders. The steering rudders II and IV will turn, and the needed position of the rocket axis will be restored.

If during flight the potentiometer were turned at a certain angle $\Delta\varphi$ about the rocket body, then, obviously, the rudders will act just as if the rocket itself deviated by this angle and will turn the rocket at an angle $\Delta\varphi$. Thus, turning potentiometer 8 by a given law, we will evoke turning of the rocket by the same law in the plane of the trajectory, i.e., we will assign the rocket a programmed change of pitch angle. Turning of the potentiometer is produced by the so-called program mechanism (see Fig. 9.18).

Potentiometer 8 and pulley 9 are a rigid unit. Onto the pulley 9 is put a thin metallic belt 10, connected on the other side with the eccentric 11, profiled depending upon the assigned program. Eccentric 11 is put into motion through screw transmission by the step motor 12. The latter is an electromagnet with an anchor. When a pulse moves to the electromagnet, the anchor is attracted to the magnet and by its edge shifts the ratchet wheel 13 by one tooth. Thus, the speed of rotation of the wheel 13 depends on the frequency of pulses passed onto the electromagnet. In flight this speed is maintained constant. Component 14, shown on Fig. 9.18, is the stopper of the ratchet wheel, not allowing its turning in the opposite direction.

Gyroverticant

The second gyroscopic instrument of the stabilization automaton is the "Verticant." This instrument is put on a rocket to guarantee stabilization on the course and bank and controls rudders I and III.

The structure of this instrument is shown on Fig. 9.19.

The rotor axis is located perpendicular to the plane of the programmed trajectory; therefore, the gyroscope is insensitive to change of pitch angle of the rocket, but reacts to bank turns and course deflections.

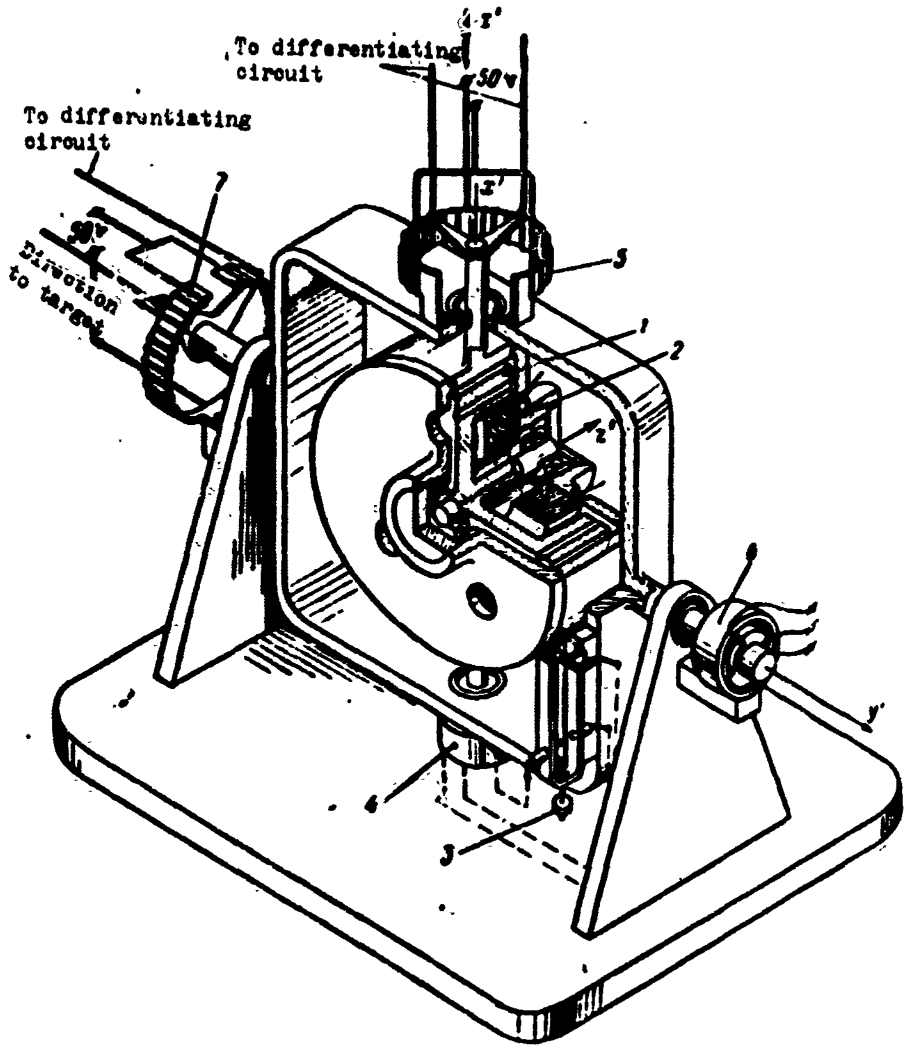

Fig. 9.19. Structure of gyroverticant. 1 — rotor of gyroscope; 2 — winding of stator; 3 — pendulum; 4 — electromagnet; 5 — potentiometer; 6 — electromagnet; 7 — potentiometer.

As for the "Horizon" for rotation of rotor 1 alternating voltage is used in the winding 2.

Correction of the "Verticant" instrument is carried out just as in the "Horizon," and thus continues only to the moment of takeoff of the rocket.

At deflection of axis x' from the vertical, the pendulum 3 makes one of the contacts, as a result of which a signal passes to electromagnet 4, and axis x' returns to the vertical position. If the gyroscope turns with respect to axis x', from potentiometer 5 a signal will be given to the electromagnet 6, and the normal position of the gyroscope also will be attained.

After takeoff, potentiometer 5 no longer governs the correcting electromagnet 6 but rudders I and III. Since axis x' coincides with the longitudinal axis of the rocket, then, obviously, potentiometer 5 in flight will react to banking of the rocket and will transmit signals on the error to rudders I and III.

From potentiometer 7 a signal on correction of the course will be removed. This signal should influence rudders I and III in such a manner so that they turn synchronously.

Steering Machines

Steering machines are the acting organs, turning the rudders to the needed direction and in the needed amount depending upon the signal obtained.

Fig. 9.20 shows a diagram, and Fig. 9.21 – the structure of an electro-hydraulic steering control.

A signal taken from the potentiometer of the gyroscope, converted and intensified in intermediate devices of the stabilization automaton is transmitted to the polarized relay 1 of the steering machine. The polarized relay, depending upon the sign of signal turns to one side or the other of the balance beam 2 and the valve distributor 3 connected with it, suspended by a laminar spring 4 replacing a mechanical hinge.

With turning of the distributor, one of the plungers (5 or 6) covers the bypass of oil supplied by a gear pump 7. As a result, pressure in one cavity of the working cylinder increases, and piston 8

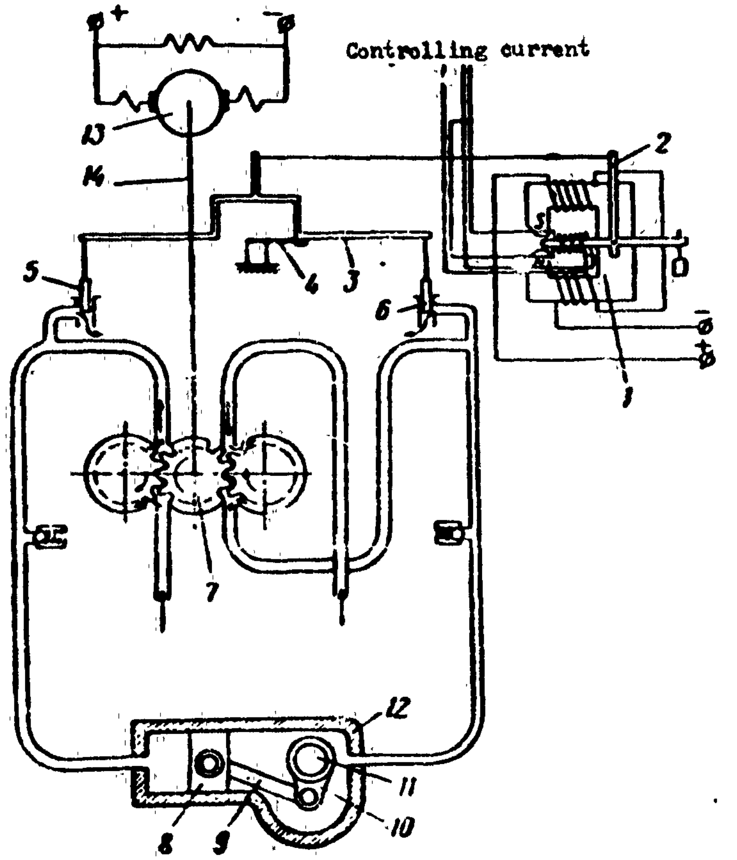

Fig. 9.20. Diagram of the steering machine of a rocket (position the same as in Fig. 9.21).

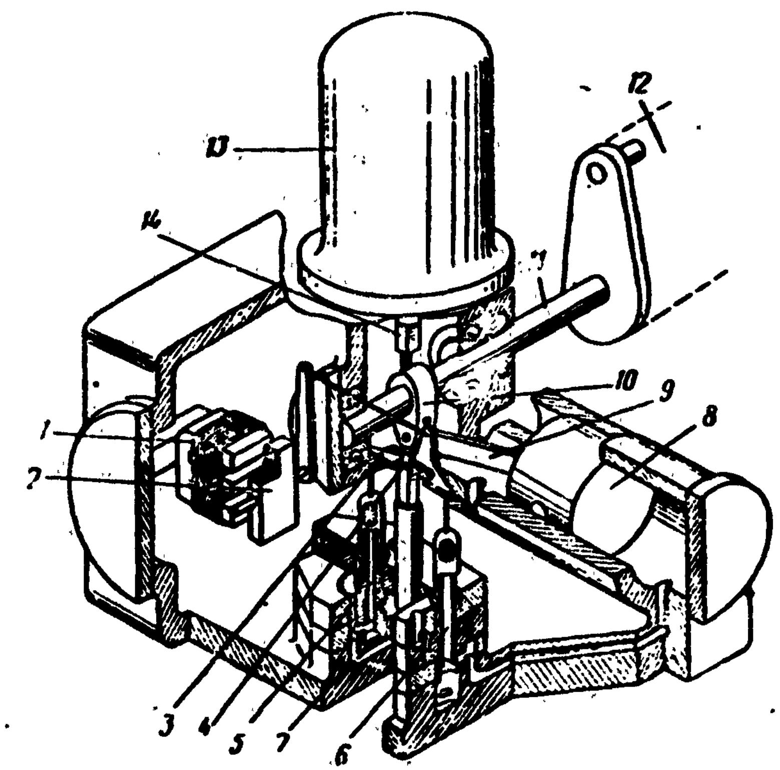

Fig. 9.21. Structure of steering machine. 1 — polarized relay; 2 — balance beam; 3 — valve distribution; 4 — laminar spring; 5, 6 — piston of valve device; 7 — gear pump; 8 — operating piston; 9 — connecting rod; 10 — crank; 11 — shaft of steering rudder; 12 — steering rudder; 13 — motor of gear pump; 14 — shaft of motor.

moves in the corresponding direction. The force through the connecting rod 9 and crank 10 will be transmitted to the shaft 11 of the rudder. Rudder 12 fastened to the shaft will turn in the needed direction. The gear pump is put into rotation by the motor 13 through the shaft 14.

The described steering machine works only on one rudder. Four such machines should be established on the rocket.

Air rudders can be put into motion or by their steering machines, or by machines of the corresponding gas current steering rudders. In the second case gas current and air rudders turn synchronously, but at different angles, which is caused by the transmission ratio of their connecting mechanism.

4. Intermediate Devices of the Stabilization Automaton of a Long-Range Rocket

General Circuit of Intermediate Devices

Intermediate devices of the stabilization automaton are used for conversion and strengthening of signals sent from potentiometers of the gyro instruments, and transmission of them to the steering machines.

A fundamental circuit of intermediate devices of a stabilization automaton is represented on Fig. 9.22.

Stabilization of pitch angle with the help of "Horizon" is carried out independently of stabilization of yaw and bank. Thus, signals passed by "Horizon" and "Verticant" have independent systems of conversion and independent supply to steering rudders.

A signal in the form of voltage sent from the potentiometer of "Horizon," proceeds first of all to a differentiatint circuit, where it will be converted to

$$u \approx C_0\Delta + C_1\dot{\Delta\varphi} + C_2\ddot{\Delta\varphi} \qquad (9.4)$$

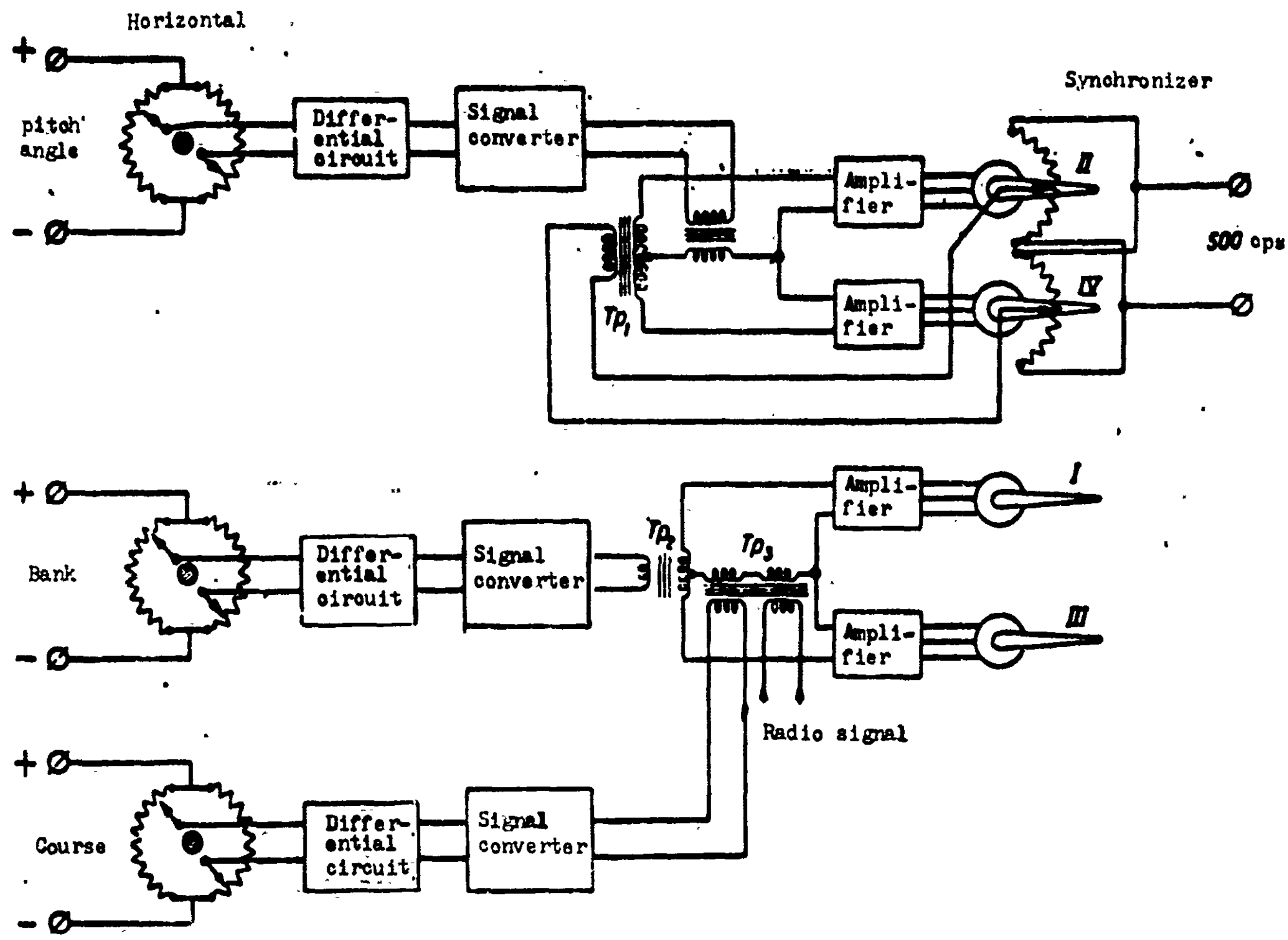

Fig. 9.22. Fundamental circuit of intermediate devices of a stabilization automaton.
[Tp = Tr = transformer].

The task of the differentiating circuit is, thus, to add to the voltage, proportional to the angle of displacement of the rocket, components proportional to angular velocities and angular acceleration of the rocket. Such conversion as will be shown lower, is necessary to guarantee stability of control.

The potentiometer is fed from a source of direct current, since to make a differentiating circuit which would work on alternating current is very difficult.

The signal from the differentiating circuit goes to the converter, where it will be converted to a signal of alternating current with a frequency of 500 cycles per second, modulated by amplitude. Such conversion is produced in order to simplify the subsequent strengthening since strengthening of a direct current signal presents great difficulties.

The converted signal through the transformer passes to amplifiers controlling steering machines of rudders II and IV.

Since rudders II and IV are not mechanically connected between themselves, a synchronizing device is introduced into the circuit of stabilization of pitch ensuring synchronousness of work of rudders II and IV and excluding, thus, the appearance of undesirable perturbations of bank. When steering rudders II and IV, and together with them the potentiometers of the synchronizer, turn at the same angle, no signal passes to the primary winding of the transformer Tr_1. With turning of the steering rudders at different angles, a signal will be given to the primary windings of transformer Tr_1, proportional to the difference of angles of turning of the rudders. This signal will proceed then to the amplifiers and through them to a relay of steering machines; as a result coordination of the rudders will be restored.

The control system of rudders I and III is analogous to the control system of rudders II and IV. Signals proceeding from two potentiometers differentiate separately, are converted and move through the transformers to amplifiers of the steering machines of rudders I and III. The signal, proceeding from the potentiometer of "Bank" forces the rudders to turn in opposite directions, but a signal from the potentiometer of "Course" is transmitted in such a manner so that

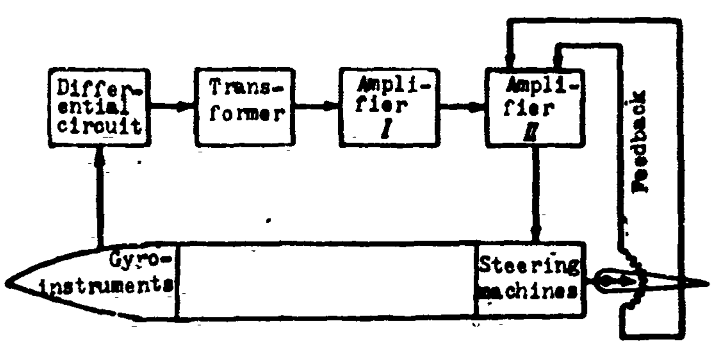

Fig. 9.23. Circuit of control system for rocket with feedback.

the rudders turn in one direction simultaneously.

The circuit of Fig. 9.22 for transformer Tr_3 shows one more free winding. Through it may be given a signal of the radio control. This signal, obviously, as also the signal from the potentiometer of "Course," will correct rocket flight by course.

For a better control system, in the circuit of the stabilization automaton also feedback from the rudders to the amplifier block can be anticipated (Fig. 9.23).

Let us consider the separate elements of the intermediate devices.

Differentiating Circuit

Voltage u, sent by the potentiometer of the gyroscope, is proportional to the angle of rocket rotation $\Delta\varphi$. Since $\Delta\varphi$ is a function of time, then u is also function of time.

Differentiation of the signal u may be carried out with the help of a simple "RC-circuit," i.e., by a series hook-up of capacitors and ohmic resistors (Fig. 9.24a).

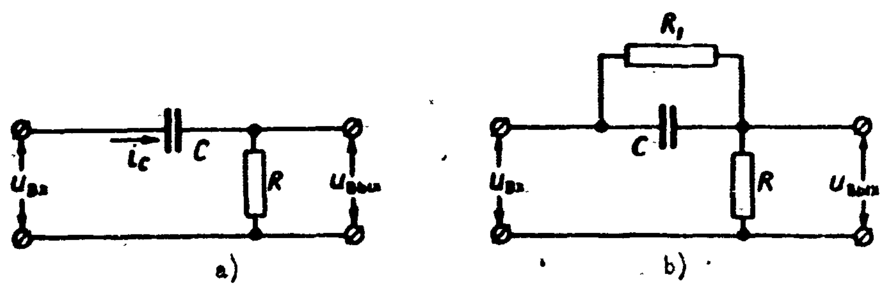

Fig. 9.24. Differentiating circuits. a) simple "RC-circuit;" b) more complex differentiating circuit.
[BX = in = inlet; BHX = out = outlet].

The differentiable signal moves to the inlet terminals, and its derivative is passed out from the outlet terminals. Here, obviously,

$$u_{вх} - u_C = u_{вых},$$

where u_c is the voltage drop on the capacitor.

The current which passes through the capacitor is

$$i_C = C\frac{du_C}{dt}.$$

If a load is not connected to the terminals, i.e., if the circuit works on the amplifier, then

$$u_{\text{вых}} = i_C R.$$

Excluding from the three obtained equations u_C and i_C, we get

$$u_{\text{вых}} + RC\frac{du_{\text{вых}}}{dt} = RC\frac{du_{\text{вх}}}{dt}.$$

If magnitude RC is sufficiently small, then the second component of the left member of this expression as compared to the first can be disregarded, and then we will have almost pure differentiation,

$$u_{\text{вых}} \approx RC\frac{du_{\text{вх}}}{dt} = RC\dot{u}_{\text{вх}},$$

i.e., at the outlet a voltage will be obtained almost proportional to the derivative of the entrance signal.

It is important to note that inasmuch as RC is small, the converted signal, as follows from the last expression is sharply weakened by the differentiating circuit. Therefore, subsequently it is necessary to introduce amplifying devices.

If we want to obtain an outlet signal depending not only on derivative $u_{in} = du_{in}/dt$ but also on the function u_{in} itself, then in the simple differentiating circuit one should include resistance R_1 in parallel to capacitor C (see Fig. 9.24b). The output signal here will have the form

$$u_{\text{вых}} \approx m_1 u_{\text{вх}} + m_2 \dot{u}_{\text{вх}},$$

where m_1 and m_2 are constants depending on parameters R, R_1 and C.

Thus, output voltage in the circuit turns out to be proportional to voltage u_{in} and its first derivative, and the cell shown on Fig. 9.24b carries out single differential conversion of the signal. For double differentiation of the signal the circuit shown on Fig. 9.25

is applied. The voltage passed from the outlet of this circuit is

$$u_{вых} \approx n_0 u_{вх} + n_1 \dot{u}_{вх} + n_2 \ddot{u}_{вх},$$

where n_0, n_1 and n_2 depend on circuit parameters.

This signal, converted and intensified, in final results reaches the polarized relay of the steering machines controlling turning of of the steering rudders. As result, a change of angles of turning of the rudders will depend not only on the angles of error on the gyro system potentiometers but also on their derivatives by time. For instance,

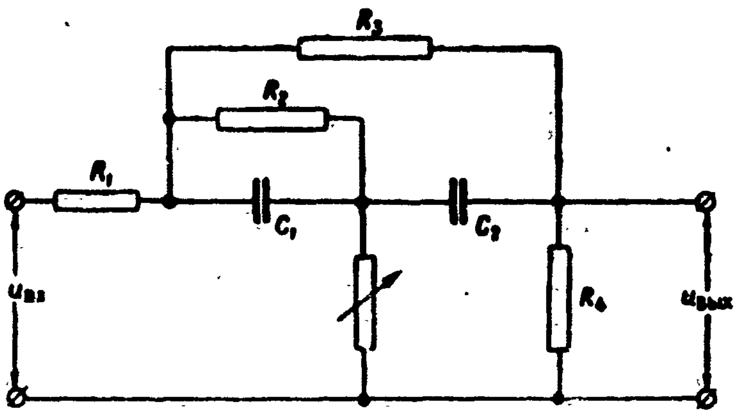

Fig. 9.25. Diagram of a differentiating circuit of a stabilization automaton.

$$\Delta\delta = b_0 \Delta\varphi + b_1 \Delta\dot{\varphi} + b_2 \Delta\ddot{\varphi}. \qquad (9.5)$$

The relationship between coefficients b_0, b_1 and b_2 depends on the relationship between coefficients n_0, n_1 and n_2. Consequently, the relationships between coefficients b_0, b_1 and b_2 necessary for control stability, are made by means of proper selection of resistors and capacitors of the differentiating circuit.

Signal Converter (Modulator)

With passage through the differentiating circuit the signal is sharply weakened, and it is necessary to strengthen it. Since strengthening of a direct current signal presents very great difficulties, then in the considered system of the stabilization automaton the signal is converted into alternating current, modulated by amplitude. For that the circuit shown on Fig. 9.26 is applied.

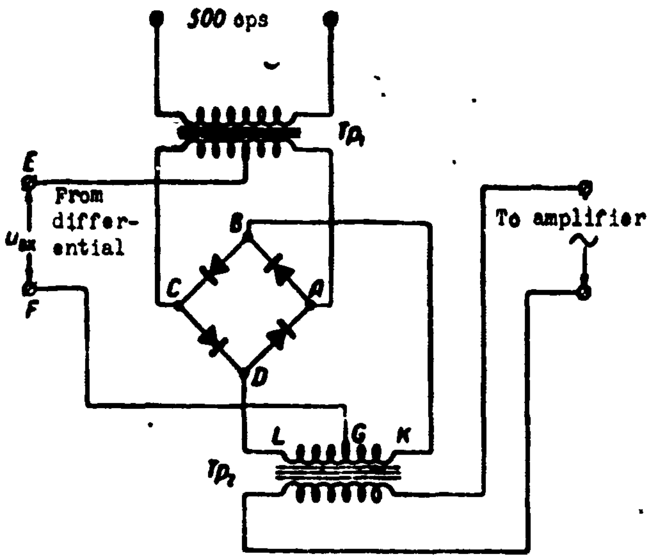

Fig. 9.26. Converter circuit.

Voltage from the differentiating circuit passes to the median points of the windings of two transformers: the first Tr_1, inlet, fed by alternating voltage of 500 cycles per second, and the second Tr_2, output, from which the converted signal is passed.

The basic element of the circuit is a bridge of four selenium rectifiers.

Selenium rectifier are a set of iron or aluminum disks covered on one side by a layer of selenium. On the selenium in turn is applied a layer of some fusible metal. The film on the boundary between selenium and the fusible metal has the property of being able to easily pass current in one direction and almost eliminate it in the other. The graph on Fig. 9.27 shows the magnitude of current intensity passing through a selenium rectifier, depending upon the applied voltage.

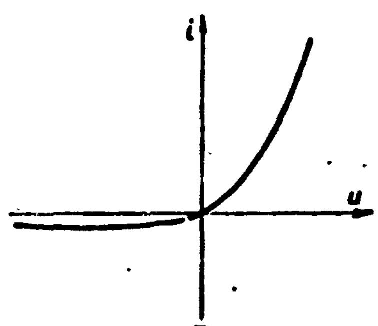

Fig. 9.27. Characteristics of a selenium rectifier.

The bridge of selenium rectifiers ABCD (see Fig. 9.26) becomes strictly symmetric. In the absence of an entrance signal on terminals E and F it is balanced. In spite of the fact that onto the circuit of bridge ABC (diagonal AC) a voltage passes from transformer Tr_1 in the second bridge circuit BAD (diagonal BD), united with the primary winding of output transformer Tr_2, the voltage is equal to zero.

During passage of a signal the symmetry of the bridge in virtue

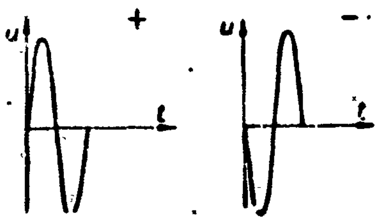

Fig. 9.28. Change of output signal in one period of alternating current depending upon polarity of signal.

of the nonlinear characteristic of the selenium rectifiers will be disturbed, and a voltage will appear on diagonal BD proportional to the strength of the entrance signal. At the ends of the secondary winding of transformer Tr_2 in the same way a voltage appears. This will be an alternating voltage, the amplitude of which is proportional to the magnitude of the entrance signal of direct current. If the direction of the current of the entrance signal is changed, then the phase of the output alternating voltage is changed by 180° (Fig. 9.28).

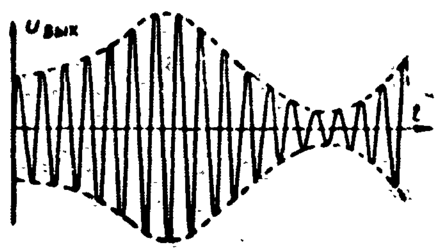

Fig. 9.29. Converted signal.

In result, in the secondary winding of output transformer we obtain an alternating voltage with a frequency of 500 cycles per second and with an amplitude changing according to the law of the signal proceeding from the differentiating circuit (Fig. 9.29). The phase of the output alternating voltage depends on polarity of the entrance signal.

Amplifier and Demodulator

A circuit of an amplifier and demodulator is shown in Fig. 9.30.

The basic element of the amplifier is an electron tube (pentode), widely applied in the most varied electronic devices and circuits. The tube is fed from a source with a voltage of 200-250 volts.

A voltage passes onto the control grid of the tube from the secondary winding of the converter output transformer. This voltage is strengthened by the tube and moves to winding A of a three-rod transformer (the windings for this transformer are superimposed on three limbs of the core).

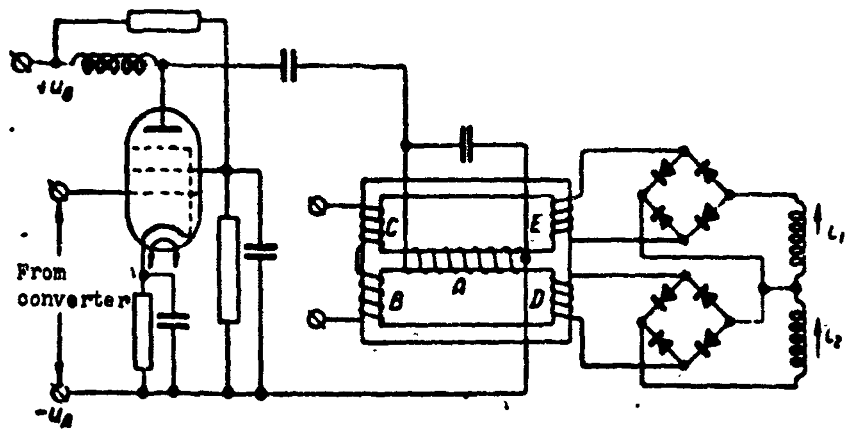

Fig. 9.30. Amplifier circuit.

The intensified voltage is a voltage of alternating current with a frequency of 500 cycles per second, the amplitude of which is proportional to the signal at the outlet of the differentiating circuit, and its phase depends on polarity of the signal.

Since in final result the signal should act by polarized direct current, it is necessary to turn the intensified signal again into a signal of direct current. This inverse, second, conversion of the signal (or demodulation) is done by a three-rod transformer and selenium rectifiers, connected together as shown on Fig. 9.30.

To the three-rod transformer, besides an intensified signal voltage (winding A), is brought an alternating voltage with a frequency of 500 cycles per second from the feeding source (windings B and C). The latter is called "support," since by superimposition with it the phase of the intensified signal is established, i.e., the sign of the initial signal passed from the potentiometer of the gyro instrument.

If the control signal on winding A is absent, then the voltages on windings E and D are equal to each other.

Voltages passed from windings E and D are straightened by selenium rectifiers and are brought into two windings of a polarized relay. Rectifiers and winding of the polarized relay are included here so that magnetic fluxes created by currents in both windings act against each other. At a zero signal both currents i_1 and i_2 (see Fig. 9.30) are equal by magnitude, and the total effect is equal to zero. Consequently, the anchor of the relay remains at rest.

The three-rod transformer works in such a way that depending upon whether the currents in windings B, C and A, coincide or are opposite in phase voltage in winding E is correspondingly increased or decreases (moreoever, in winding D the voltage correspondingly is decreased or increased).

If a signal is given, then depending upon its sign one of the currents (i_1 or i_2) is increased, and the other decreases. Here the polarized relay works in one or the other direction. In result, turning of the rudder occurs, depending on the angle, angular velocity and acceleration of turning of the rocket.

5. Certain Questions on Maintenance of Stability of Rocket Flight

Analysis of Stability of Motion in Simple Form

In the example of a simple course automaton (see p.414) we already saw that with deflection of the flying apparatus off course, simple deflection of the rudder in the needed direction is still insufficient to reach stabilization. If deflection of the rudder occurs in the needed direction, but not in the needed amount or with a delay, we always risk unstable conditions of rocket control — control with rocking or with total derivation of it from the given course.

We will clarify what conditions should satisfy the stabilization system to guarantee stability of motion. With this goal we will turn to the motion equation of a rocket (7.3) and (7.4)

$$\dot{\theta} = \frac{1}{Mv}[(P - X_{ynp})\alpha + Y + Y_{ynp}] - \frac{g}{v}\cos\theta;$$
$$J\ddot{\varphi} + M_a + Y_{ynp}c + M_{ш} = 0$$

[упр = con = control, ш = hin = hinged]

and will find the character of motion of a rocket after it experiences a certain perturbation, as in the form of an instantaneous angular turn.

We will simplify the problem, rejecting hinged moment M_{hin} in the equation as a magnitude small in comparison with $Y_{con}c$. We will furthermore, consider that the remaining parameters of motion of the rocket after a perturbation at an angle φ remain constant. Thus, for instance, the rocket speed v and the aerodynamic coefficients will be constant, preserving that magnitude which they had before the perturbation.

In first approximation for appraisal of stability such an assumption is fully permissible, inasmuch as we are interested not in the motion of the rocket itself, but only in deflection of the rocket from undisturbed motion.

With a perturbation transmitted to the rocket, variables entering into the written equations will be changed, and we get:

$$\Delta\dot{\theta} = \frac{1}{Mv}[(P - X_{ynp})\Delta\alpha + \Delta Y + \Delta Y_{ynp}] + \frac{g}{v}\sin\theta\Delta\theta;$$
$$J\Delta\ddot{\varphi} + \Delta M_a + c\Delta Y_{ynp} = 0;$$
$$\Delta\varphi = \Delta\theta + \Delta\alpha.$$

Change of lift ΔY is connected with a change in the angle of incidence $\Delta\alpha$ linearly:

$$\Delta Y = c_y^\alpha \frac{\rho v^2}{2} S \Delta\alpha.$$

Change of control force ΔY_{con} is connected with a change of the angle of rotation of the rudder $\Delta\delta$, also linearly:

$$\Delta Y_{\text{упр}} = D_1 \Delta\delta,$$

where D_1 is the porportionality factor.

Moment M_a is composed, as we know, from static and damping moments. The first of them is proportional to the angle of incidence, and the second — to the angular velocity of rotation of the rocket. Thus,

$$\Delta M_a = D_2 \Delta\alpha + D_3 \Delta\dot{\varphi}, \tag{9.6}$$

where D_2 and D_3 are proportionality factors.

Now we exclude ΔY, ΔY_{con} and ΔM_a from the obtained equations. Then we get

$$\left.\begin{aligned} -\Delta\dot{\theta} + C_1 \Delta\alpha + C_2 \Delta\theta + C_3 \Delta\delta = 0; \\ \Delta\ddot{\varphi} + m_1 \Delta\dot{\varphi} + m_2 \Delta\alpha + m_3 \Delta\delta = 0; \\ \Delta\varphi = \Delta\theta + \Delta\alpha, \end{aligned}\right\} \tag{9.7}$$

where

$$\left.\begin{aligned} C_1 &= \frac{P - X_{\text{упр}} + c_y^\alpha \frac{\rho v^2}{2} S}{Mv}; \\ C_2 &= \frac{g}{v} \sin\theta; \\ C_3 &= \frac{D_1}{Mv}; \\ m_1 &= \frac{D_3}{J}; \\ m_2 &= \frac{D_2}{J}; \\ m_3 &= \frac{cD_1}{J}. \end{aligned}\right\} \tag{9.8}$$

The desired unknowns in equations (9.7) are change of angles $\Delta\varphi$, $\Delta\theta$, and $\Delta\alpha$. Coefficients c_1, c_2, c_3, m_1, m_2 and m_3 are changed in time, but essentially slower than magnitudes $\Delta\varphi$, $\Delta\theta$ and $\Delta\alpha$.

Thus, with the help of the derived equations rapidly variable angulardiaplacements in the stabilization process are separated from those slow variations of parameters, which are considered in ballistic calculations. This separation was discussed at the start of Chapter VIII.

Coefficients C_1, C_2, C_3, m_1, m_2 and m_3 can be considered as constants calculated for a certain moment of time on the trajectory.

If from equations (9.7) angles $\Delta\alpha$ and $\Delta\theta$ are excluded we get

$$\begin{aligned}&\dddot{\Delta\varphi}+(C_1-C_2+m_1)\ddot{\Delta\varphi}+[(C_1-C_2)m_1+m_2]\dot{\Delta\varphi}-\\&-C_2m_2\Delta\varphi+m_3\dot{\Delta\delta}+[(C_1-C_2)m_3-C_2m_2]\Delta\delta=0.\end{aligned} \qquad (9.9)$$

Now the whole question consists in how the angle of rotation of the rudders $\Delta\delta$ and of perturbation of the rocket $\Delta\varphi$ are connected. If they are connected so that solution of equation (9.9) will give an attenuating, and better still a rapidly attenuating process of change of $\Delta\varphi$, control of the rocket will be stable. If, however, with solution of equation (9.9) we reveal that variable $\Delta\varphi$ in time does not attenuate, then this will signify that the stabilization automaton does not ensure stability of motion.

Let us suppose that the angle of rotation of the rudder $\Delta\delta$ is connected from angle $\Delta\varphi$, as shown earlier in relationship (9.5)

$$\Delta\delta=b_0\Delta\varphi+b_1\dot{\Delta\varphi}+b_2\ddot{\Delta\varphi}.$$

Coefficients b_0, b_1 and b_2 are constants and meanwhile indefinite, but namely they characterize the property of the system of control. The right side of this equation is called the law of adjustment. Introducing constructive changes into the control system or changing parameters of its adjustment, we can change magnitudes b_0, b_1 and b_2 and bring about a situation in which the stabilization system ensures stable motion of the rocket.

If one were to exclude angle $\Delta\delta$ from equation (9.9), then we get

$$d_0\Delta\dddot{\varphi}+d_1\Delta\ddot{\varphi}+d_2\Delta\dot{\varphi}+d_3\Delta\varphi=0, \tag{9.10}$$

where

$$\left.\begin{aligned} &d_0=1+b_2m_3;\\ &d_1=C_1-C_2+m_1+b_1m_3+b_2[(C_1-C_2)m_3-C_3m_2];\\ &d_2=(C_1-C_2)m_1+m_2+b_0m_3+b_1[(C_1-C_2)m_3-C_3m_2];\\ &d_3=-C_2m_2+b_0[(C_1-C_2)m_3-C_3m_2]. \end{aligned}\right\} \tag{9.11}$$

Thus, the law of perturbed motion of the rocket will be determined by solution of linear differential equation (9.10).

The solution of equation (9.10), as usual, is sought for in the form

$$\Delta\varphi=Ae^{kt}.$$

Placing this expression into equation (9.10), we get the characteristic equation

$$d_0k^3+d_1k^2+d_2k+d_3=0. \tag{9.12}$$

Equation (9.12) has three roots: k_1, k_2 and k_3; therefore, the solution of equation (9.10) has the form

$$\Delta\varphi=A_1e^{k_1t}+A_2e^{k_2t}+A_3e^{k_3t}. \tag{9.13}$$

Let us now see under what conditions motion will be attenuating.

The roots of equation (9.12) can be both real and complex.

With real roots k, function e^{kt} will be increasing at positive values k and attenutating at negative k.

If, however, root k is complex,

$$k=\mu+\nu i,$$

where μ and ν are real numbers, then according to the Euler identity

$$e^{kt} = e^{(\mu+\nu i)t} = e^{\mu t}(\cos \nu t + i \sin \nu t).$$

Any algebraic equation with real coefficients gives only an even number of complex roots, where these roots, as it is known, turn out to be paired.

Let us suppose, for instance, that roots k_1 and k_2 will be paired:

$$k_1 = \mu + \nu i;$$

$$k_2 = \mu - \nu i.$$

Then for the first two factors in the right member of (9.13) we find

$$A_1 e^{k_1 t} + A_2 e^{k_2 t} = A_1 e^{\mu t}(\cos \nu t + i \sin \nu t) + A_2 e^{\mu t}(\cos \nu t - i \sin \nu t),$$

or

$$A_1 e^{k_1 t} + A_2 e^{k_2 t} = (A_1 + A_2) e^{\mu t} \cos \nu t + i (A_1 - A_2) e^{\mu t} \sin \nu t.$$

Designating

$$A_1 + A_2 = B_1; \quad i(A_1 - A_2) = B_2,$$

we get

$$A_1 e^{k_1 t} + A_2 e^{k_2 t} = B_1 e^{\mu t} \cos \nu t + B_2 e^{\mu t} \sin \nu t. \tag{9.14}$$

The obtained function (9.14) will be attenuating if μ is negative. The case $\mu = 0$ signifies oscillatory undamped motion; case $\mu > 0$ signifies oscillation with build-up of amplitude.

Thus, the condition of stability of control can be formulated in the following form: for stable control it is necessary that the real parts of all roots of equation (9.12) are negative. This condition can be sustained by selecting the proper coefficients d_0, d_1, d_2 and d_3 in equations (9.12), depending on parameters of the control system.

The question of whether the system of stabilization is stable or unstable, can be answered without solving equation (9.12). Methods of higher algebra allow us to establish those relationships between coefficients d_0, d_1, d_2 and d_3 at which real parts of all the roots of the algebraic equation will be negative. For an equation of the n-th degree

$$d_0k^n+d_1k^{n-1}+\ .\ .\ .\ +d_{n-1}k+d_n=0 \tag{9.15}$$

we will apply, in particular, the criterion of stability of Gurvitz. A necessary and sufficient condition of stability of system (9.15) is the requirement that all principal minors of table

$$\left|\begin{matrix} d_1 & d_3 & d_5 & d_7 & \dots & 0 & 0 & 0 \\ d_0 & d_2 & d_4 & d_6 & \dots & 0 & 0 & 0 \\ 0 & d_1 & d_3 & d_5 & \dots & 0 & 0 & 0 \\ 0 & d_0 & d_2 & d_4 & \dots & 0 & 0 & 0 \\ \cdot & \cdot & \cdot & \cdot & \cdot & \cdot & \cdot & \cdot \\ 0 & 0 & 0 & 0 & \dots & d_{n-2} & d_n & 0 \\ 0 & 0 & 0 & 0 & \dots & d_{n-3} & d_{n-1} & 0 \\ 0 & 0 & 0 & 0 & \dots & d_{n-4} & d_{n-2} & d_n \end{matrix}\right| \tag{9.16}$$

are greater than zero, i.e.,

$$d_1>0,$$

$$\begin{vmatrix} d_1 & d_3 \\ d_0 & d_2 \end{vmatrix}>0.$$

$$\begin{vmatrix} d_1 & d_3 & d_5 \\ d_0 & d_2 & d_4 \\ 0 & d_1 & d_3 \end{vmatrix}>0$$

etc.

The table itself is constructed according to the following law.

In first line coefficients with odd indices are written, and then on the right zeroes are added in such a manner so that the number of factors in the line would be equal to n. In the second line starting from the zero factor, factors with even indices are written. The third and fourth line repeat the first and second, but shift by one member to the right, and so forth.

In the considered case of an equation of a cube (n = 3) the table takes the form

$$\begin{vmatrix} d_1 & d_3 & 0 \\ d_0 & d_2 & 0 \\ 0 & d_1 & d_3 \end{vmatrix}.$$

As a result, we get the condition:

$$d_1>0,$$

$$\begin{vmatrix} d_1 & d_3 \\ d_0 & d_2 \end{vmatrix}>0,$$

$$\begin{vmatrix} d_1 & d_3 & 0 \\ d_0 & d_2 & 0 \\ 0 & d_1 & d_3 \end{vmatrix}>0.$$

The last condition leads to the second, if $d_3 > 0$, and then the requirement of stability will take the following form:

$$d_1>0; \quad d_1d_2-d_0d_3>0 \text{ and } d_3>0. \tag{9.17}$$

A similar check of the conditions of stability should be conducted for a number of consecutive points of the trajectory, inasmuch as coefficients of the equations change on trajectory.

It is not difficult to be convinced in the fact that motion of guided rockets can appear both stable and unstable.

Let us assume that, for instance, a signal, proportional to the deflection of the rocket and not depending on speed and acceleration of turning of the rocket passes to the steering machine. Then, as this follows from expression (9.5), $b_1 = b_2 = 0$.

If one were to assume, furthermore, that the rocket left the limit of the atmosphere and the role of aerodynamic forces is insignificant, then in expression (9.6)

$$D_2=D_3=0.$$

Then from expressions (9.8) and (9.11) we get:

$$d_0=1; \quad d_1=C_1-C_2; \quad d_2=b_0m_3;$$
$$d_3=b_0m_3(C_1-C_2).$$

The obtained expressions jointly with expression (9.17) require that $C_1 > C_2$. This condition, as can be seen from expressions (9.8) may be easily consistent. However, magnitude $d_1d_2 - d_0d_3$ appears equal to zero, although conditions (9.17) in stable motion should be greater than zero. This means that the system is at the limit of stability, and the rocket will experience undamped oscillations. The cause of this was an incorrect law of adjustment, occurring in the fact that a signal moves to the rudders proportional only to angle $\Delta\varphi$.

In order to get stable motion, it is sufficient to send signals to the steering machines, proportional not only to $\Delta\varphi$, but also to $\dot{\Delta}\varphi$, and for improvement of the control system – proportional also to $\ddot{\Delta}\varphi$. Then in expression (9.5) b_1 and b_2 will not be equal to zero. With proper selection of these coefficients it is easy to ensure stability of the system of stabilization.

The stabilization system of a V-2 ballistic long-range rocket is based namely on this principle.

For appraisal of stability of the stabilization process there exist many different theoretical methods more perfect and exact than the above-stated simple method. All these methods, however, do not satisfy the demands of practice, inasmuch as the dependence of different entrance and output magnitudes in many elements of the stabilization automaton are much more complex than those used in theoretical investigation. Certain elements of the stabilization automaton have essential nonlinearity, and equations characterizing the work of these elements do not yield to sufficiently exact and simple analysis. Therefore, in practice of creation and adjustment of control systems of rockets we resort usually to the method of modeling.

Principles of Modeling of the Work of a Stabilization Automaton

Modeling is understood as the reproduction of behavior of a certain complicated system with the help of another, simpler and more accessible to observation. This simpler system is called an <u>analog computer</u>.

The stabilization automaton is designed usually on the basis of experiment and certain preliminary theoretical considerations connected with the design features of the rocket.

Created and checked by separate blocks, the stabilization automaton cannot be used in the controls of an aircraft, at least because it still needs necessary general adjustment in accordance with structural and ballistic characteristics of the rocket itself. In distinction from aircraft automatic pilot systems, the adjustment of a stabilization automaton of a pilotless aircraft cannot be modified during flight.

In virtue of the shown circumstances, the control system is preliminarily investigated by analog computers.

Practice shows that linearized equations (9.7) sufficiently accurately reflect the peculiarity of rocket behavior in the process of stabilization. They can be generalized by introduction of a perturbing transverse force N_{pert} and perturbing moment M_{pert} and will take the following form:

$$\left.\begin{aligned} -\Delta\dot{\theta}+C_1\Delta\alpha+C_2\Delta\theta+C_3\Delta\delta&=\frac{N_{\text{возм}}}{Mv};\\ \Delta\ddot{\varphi}+m_1\Delta\dot{\varphi}+m_2\Delta\alpha+m_3\Delta\delta&=\frac{M_{\text{возм}}}{J};\\ \Delta\varphi&=\Delta\theta+\Delta\alpha. \end{aligned}\right\} \qquad (9.18)$$

[ВОЗМ = pert = perturbing]

Magnitudes N_{pert} and M_{pert} are limited in size and reflect the influence of possible perturbing factors. They can be given in the form of certain functions for any moment of flight time.

The law of adjustment (9.5)

$$\Delta\delta = b_0\Delta\varphi + b_1\Delta\dot{\varphi} + b_2\Delta\ddot{\varphi}$$

in distinction from equations (9.18) cannot be used to obtain sufficiently correct judgement of the stability of the stabilization system. As experiment shows, the dependence of change in the angle of displacement of the rudder $\Delta\delta$ on the angle of displacement of the rocket axis $\Delta\varphi$ has more complicated differential relationships containing nonlinearity. Therefore, in general, the control equation has the form

$$F(\Delta\delta,\ \Delta\dot{\delta},\ \ldots,\ \Delta\varphi,\ \Delta\dot{\varphi},\ \Delta\ddot{\varphi}) = 0. \qquad (9.19)$$

Thus, for calculation of stabilization of pitch angle three equations (9.18) and equation (9.19) have to be solved jointly. Solution of equations (9.18) is derived on a modelling installation. For reproduction of dependence (9.19) in nature the stabilization automaton itself is used.

Electronic modelling installations, in which the analogs of reproduced magnitudes (angles $\Delta\varphi$, $\Delta\theta$, and $\Delta\alpha$) are voltages in different elements of electrical circuits, are the most convenient in manipulation.

For solution of equations similar to equations (9.18), it is necessary that the electronic device is able to carry out the following simple operations:

1) multiplication of voltage by a certain constant or by a slowly variable magnitude (with change of sign or without change);

2) integration of tension function;

3) differentiation of tension function.

From the elements producing the shown opeartions, circuits reproducing the solution of given equations can be built.

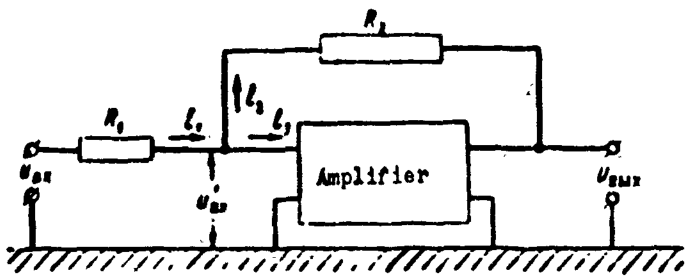

Fig. 9.31. Circuit of multiplication.

For execution of the first of the three operations a cell built by the circuit shown on Fig. 9.31 is used.

The basic element in this circuit is a multistage vacuum-tube amplifier or direct current with a large amplification factor ($K = 10^5$-10^6)

$$u_{\text{вых}} = \pm K u'_{\text{вх}}.$$

Current intensity

$$i_1 = \frac{u_{\text{вх}} - u'_{\text{вх}}}{R_1},$$

$$i_2 = \frac{u'_{\text{вх}} - u_{\text{вых}}}{R_2}.$$

Since i_3 is immeasurably less than i_1 and i_2, then

$$i_1 \approx i_2,$$

$$\frac{u_{\text{вх}} - u'_{\text{вх}}}{R_1} = \frac{u'_{\text{вх}} - u_{\text{вых}}}{R_2}.$$

In virtue of the fact that u'_{in} is K times less than u_{out}, where K is a very large number, obviously,

$$u_{\text{вых}} = -\frac{R_2}{R_1} u_{\text{вх}}.$$

Thus, at the proper relation R_2/R_1 an increase of input voltage by the needed number of times is attained. If factor R_2/R_1 must be changed with time, resistor R_1 or R_2 becomes a variable, and the cursor of the potentiometer receives a displacement by the given law.

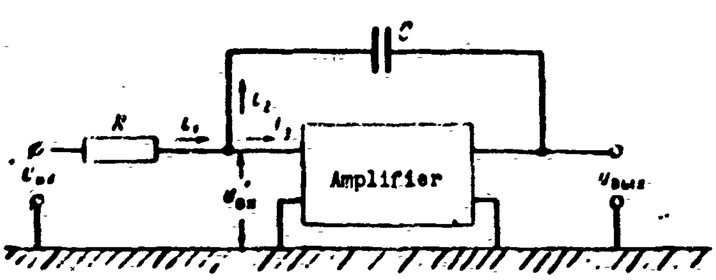

Fig. 9.32. Circuit of integration.

For the integration operation the same circuit is used but the

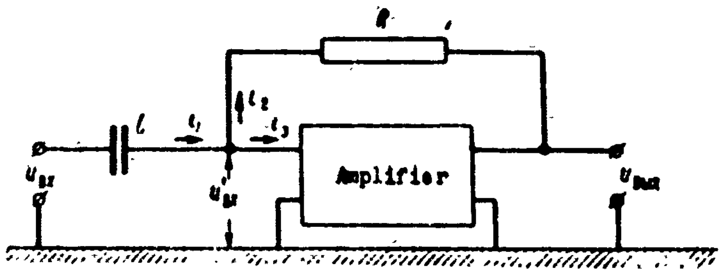

Fig. 9.33. Circuit of differentiation.

active resistor R_2 is replaced a capacitor (Fig. 9.32). In this case

$$i_1=\frac{u_{вх}-u'_{вх}}{R},\quad i_2=C\frac{d}{dt}(u'_{вх}-u_{вых}).$$

Equating the currents and considering that

$$u'_{вх}=\frac{u_{вых}}{K},$$

at a large K we get

$$u_{вых}=-\frac{1}{RC}\int u_{вх}\,dt. \qquad (9.20)$$

Fig. 9.33 shows the circuit of differentiation. Here

$$i_1=C\frac{d}{dt}(u_{вх}-u'_{вх}),\quad i_2=\frac{u'_{вх}-u_{вых}}{R}.$$

Hnece in virtue of the great size of K

$$u_{вых}=-RC\frac{du_{вх}}{dt}.$$

From similar cells producing the shown operations it is possible to make circuits for solution of many equations. Let us assume that, for instance, it is required that we make a circuit for solution of a simple equation of the first degree

$$\dot{x}+Tx=0, \qquad (9.21)$$

where T is a given slowly changing function of time. At t = 0, x = x_0.

Let us present this equation in the form

$$x\approx-T\int x\,dt.$$

The error of this equation is smaller, the slower T will change as compared to x.

Comparing this expression with expression (9.20), we notice their total similarity. Consequently, equation (9.21) is solved by the usual

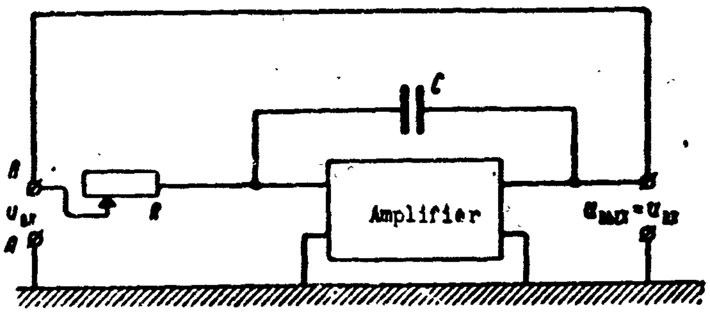

Fig. 9.34. Circuit for solution of equation $\dot{x} + Tx = 0$.

integrating cell (Fig. 9.34), the outlet of which is closed with the inlet, inasmuch as u_{in} and u_{out} are analogs of the same magnitude x. A circuit for solution of equation (9.21) is represented on Fig. 9.34.

The analog of variable coefficient T is parameter 1/RC, the magnitude of which changes by means of a shift of the potentiometer cursor by a given law.

The operation of solution of equation (9.21) appears in the following form.

To clamps A and B a recording instrument is connected, for instance an oscillograph, which will fix magnitude u_{in}, reproducing the law of change of x. To the same clamps a constant voltage u_0 is applied corresponding to the initial value x_0. In moment $t = 0$, the source of the voltage is disconnected, and the potentiometer cursor R is put into motion by a given law. In subsequent time the oscillograph will fix the desired dependence $x \approx f(t)$.

Now we will consider in broad terms how a modelling installation for solution of equations (9.18) and (9.19) should look. For reproduction of dependence (9.19) in nature the stabilization automaton (AC, Fig. 9.35). If on its entrance, i.e., to the potentiometers of the gyro instrument, a signal $\Delta\varphi$ is applied, the rudders will respond by a deflection $\Delta\delta$. In order to completely reproduce the condition of work of the rudders, the latter communicate with a load stand including mobile masses, springs, and a viscous resistance. As a result, with modeling, the rudders are under the influence of forces similar to those acting on them in a gas flow. From potentiometers fixed on the rudders an electrical signal is passed proportional to $\Delta\delta$.

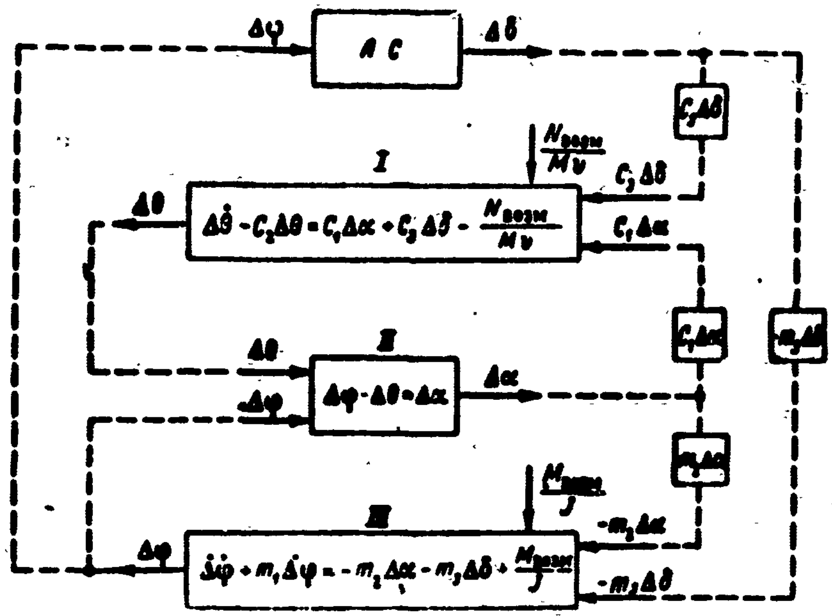

Fig. 9.35. Circuit of modeling of the stabilization process of a rocket's pitch angle.

Then from the simple cells blocks are collected producing operations of integration of equation (9.18) (see Fig. 9.35).

Block I corresponds to the first equation (9.18) and determines angle $\Delta\theta$ by magnitudes $C_1\Delta\alpha$, $C_3\Delta\alpha$. Block II produces subtraction of angles $\Delta\varphi$ and $\Delta\theta$. Finally, block III integrates twice the second equation (9.18) and issues an angle $\Delta\varphi$ found on the basis of magnitudes $m_2\Delta\delta$, $m_3\Delta\delta$, and M_{pert}/J.

If all blocks were connected, as this is shown by the dotted line on Fig. 9.35, then we obtain a general block-diagram of the modelling installation. Perturbing force N_{pert} and moment M_{pert} are introduced into the model form without in any moment of time in the required form. All coefficients of the equation are variables. The necessary

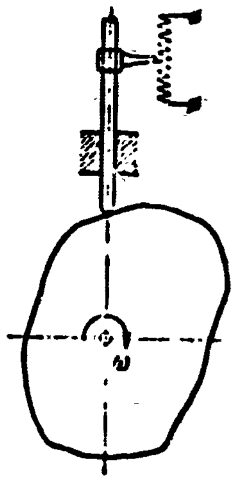

Fig. 9.36. Variation of coefficient in a modelling installation.

laws of their change are given through variables of resistance in a special device called the variator of coefficients.

The variator of coefficients is a set of specially profiled cams planted on a general axis, the angle of rotation of which is proportional the flight time of a rocket on the trajectory. With cams are connected pushers of potentiometers included in the corresponding circuits (Fig. 9.36).

Voltages in the modelling installation, corresponding to angles $\Delta\varphi$, $\Delta\theta$, $\Delta\alpha$ and $\Delta\delta$ are recorded with the help of an oscillograph for the whole time of modelled controlled flight of the rocket. Continually repeating "starting" of the rocket on the model, it is possible to establish the degree of influence of different perturbations, to determine how the stabilization automaton manages its own problems and to find parameters of its adjustment.

At present the method of modeling is the basic method utilized during creation of new systems of control.

6. Control of Range of Ballistic Rockets

Methods of Range Control

Range is one of the elements of the flight trajectory and is determined by parameters of rocket motion.

Achievement of a given target and the necessary distance may be obtained at different relationships of the parameters of flight. In other words, the rocket can be led to the given target in various ways.

Starting from that moment when rocket stops being controlled and continues motion under action of forces which cannot be modified by us, the flight trajectory no longer depends on our wishes and almost wholly is determined by those initial conditions of free flight which were obtained by rocket at the end of the active section. Therefore, for achievement of a given target by the rocket it is necessary that parameters of motion in the beginning of free flight are connected by definite relationships.

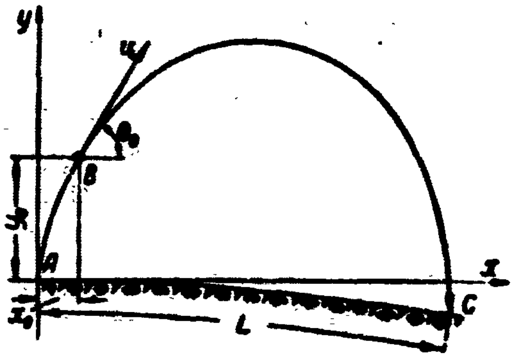

Fig. 9.37. Trajectory and distance of rocket flight.

The initial parameters of ballistic body, are, obviously, speed v_0, direction of speed vector (for planar motion of angle θ_0 or at large distances, angle ϑ_0), and the coordinates at the end of the active section x_0 and y_0 (Fig. 9.37). Furthermore, different factors can influence the trajectory of free flight, the numerical appraisal of which beforehand is unknown. Such factors are change of wind on the trajectory, deviation of the atmosphere from standard, uncertainty of stabilization of the rocket or its head part during appraoxh to the target, and so forth. However, for ballistic rockets the influence of all these factors on the flying range is relatively small.

Thus, the flying range of a rocket is a function of several magnitudes:

$$L=f(v_0, \theta_0, x_0, y_0, c), \tag{9.22}$$

where v_0, θ_0, x_0, y_0 are parameters of motion at the end of the active section, while parameter c is understood as all additional factors influencing the distance of free flight.

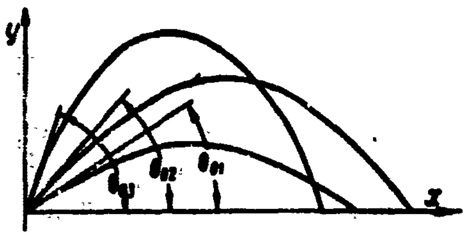

Fig. 9.38. Change of distance depending upon angle of departure θ_0.

Range control of uncontrolled solid-propellant rockets is carried out by means of simple change of the inclination angle of the directrixes at starting (Fig. 9.38).

With such a method of range control it is necessary, obviously, that the tractive force and time of work of the engine from rocket to rocket does not change; otherwise, at a constant angle θ_0 scattering by distance will appear large. In order to avoid this scattering heightened requirements of stability of dimensions and weight, and also of accuracy of manufacture of fuel chambers and nozzle holes are presented to the solid fuel charges of the engine.

However, even with complete observance of the shown conditions, scattering by distance is always prevalent, inasmuch as for different rockets differences are possible in the process of ignition, conditions of atomization, and burning of the fuel charge. For small rockets this scattering can be compensated by the quantity of rockets launched for destruction of a given area. For long-range rockets, in view of their high value, such compensation is impermissible. For these rockets it is necessary to introduce methods of range control, ensuring essentially great accuracy.

The problem of range control of a long-range combat rocket from a fundamental point of view is reduced to fixing of such a moment on time on the trajectory, when the combination of motion parameters v_0, θ_0, x_0, y_0 ensures obtaining of the necessary sighting range L (9.22). The second problem of range control consists in turning off of the engine in accordance with this moment of time.

The degree of influence on distance of each of the parameters

v_0, θ_0, x_0, and y_0 is unequal and in great measure depends on the numerical values of these parameters. For a rocket with definite structural characteristics, magnitudes v_0, θ_0, x_0 and y_0 will depend on the program of removal. In particular, it is possible to select a program of change of pitch angle on the active section in such a way that deflection of angle θ_0 will affect a change of distance in minimum degree. In this case, the basic factor determining the flying range, will be the value of speed in the moment of engine cut-off. The coordinates of the end of the active section affect the distance in a much smaller degree.

A simple, very coarse method of range control consists of turning off the engine in some moment of time assigned beforehand depending upon the required distance. Such a method of control, based on an assumption of invariability of structural parameters for a group of rockets, does not give, however, satisfactory accuracy due to significant variations of speed in the moment of engine cut-off. Scattering in the thrust of actually produced samples of one and the same motor is noticeably greater than the permissible range.

Another more usable method of range control is turning off of the engine in accordance with the speed actually obtained (and not calculated as a function of time). In this case one of the parameters of motion, affecting distance the most, is placed under control. Such a method of control gives significantly great accuracy as compared to turning off of the engine by time.

To disregard the influence of probable deviations of the finite value of angle θ_0 and coordinates x_0, y_0 is the source of basic error of this method — the so-called methodical error of range control. The most profitable way for decrease of the methodical error is turning off of the engine with the help of a radio set, using the doppler principle in its function. This device measures the difference of frequencies of radio waves sent to and reflected from the rocket. This difference of frequencies, as it is known, depends on the speed of the reflecting object, in this case a rocket. The engine is turned off in needed moment of time by a radio channel.

This method of range control, as any radio method, is subject to the influence of interferences.

Simpler and freer of the shown deficiency is the method of mechanical integration of axial overloads of the rocket on the active section. The instrument, carrying out integration of overloads and determining thus the speed of the rocket, is an integrator of overloads.

Structure and Work of a Gyroscopic Integrator of Axial Overloads

As an instrument of range control of ballistic rockets, a gyroscopic integrator shown on Fig. 9.39, can be applied.

The integrator has a gyroscopic rotor 1, put into motion by alternating current passed by the winding of the stator 2. Feeding and supply of signals from the integrator are carried out through a block 3 with slide contacts.

The rotor is placed in a housing 4 which can revolve near axis 5. The force of weight of the rotor and housing creates a moment about this axis.

Axis 5 is connected with external bracket 6 of the gyroscope, which can revolve with respect to axis x'. Through a system of toothed wheels, rotation is transmitted to disk 7, having cams 8.

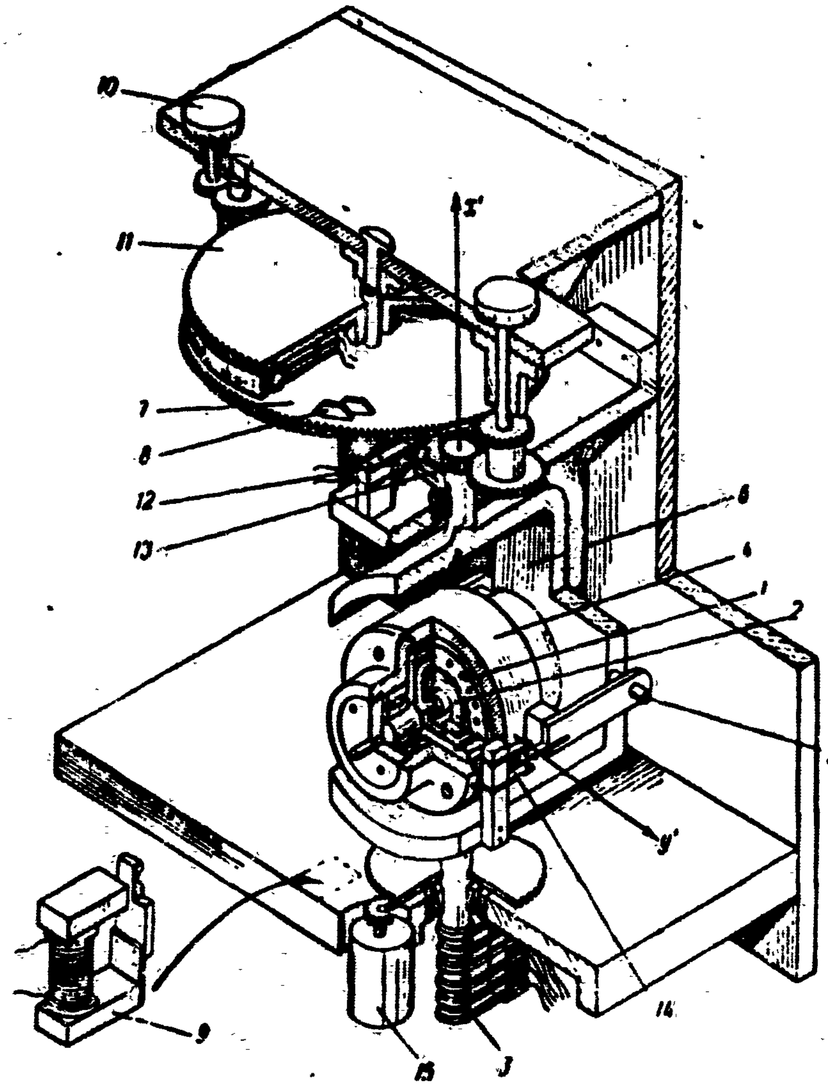

Fig. 9.39. Structure of an integrator of overloads. 1 — gyro rotor; 2 — winding of stator; 3 — block with slide contacts; 4 — housing of rotor; 5 — axis of suspension of gyroscope; 6 — external bracket; 7 — disk; 8 — cams of engine cut-off; 9 — arrester; 10 — dial for adjusting distance; 11 — disk; 12, 13 — contacts adjusting zero point; 14 — contacts of correction; 15 — motor of correction.

The integrator is established in the controls of the rocket so that its axis x' coincides with the direction of the longitudinal axis of the rocket.

Several minutes before start the gyroscope is stopped. During this time the axis of the gyroscope is secured motionlessly with the help of arrester 9. On Fig. 9.39 the arrester id displaced. It consists of an electromagnet and two gibs. During application of a signal to the electromagnet (this occurs in the moment of breakaway of the rocket from the launching pad) the gib of the arrester liberates the gyroscope axis, and the latter together with the housing hangs on axis 5.

Under action of the moment of forces of weight, precession with respect to axis x' begins.

For a motionless rocket according to equation (9.2) the angular velocity of precession is

$$\omega = \frac{G_{ч.э} a}{C\Omega},$$

[ч.э = s.e = sensitive element]

where $G_{s.e}$ — weight of sensitive element of gyroscope (weight of rotor with housing).

a — arm of action of force $G_{s.e}$ with respect to axis 5.

For a rocket moving with an acceleration j, the apparent force of weight in the direction of the rocket axis will be increased proportionally to the axial overload n_x, then

$$\omega = \frac{G_{ч.э} a}{C\Omega} n_x,$$

where according to (8.23)

$$n_x = \frac{j_x + g_h \sin \varphi}{g_0}.$$

The angle of rotation of the gyroscope with respect to axis x' during the time t is

$$\psi = \int_0^t \omega\, dt.$$

If angular momentum of the gyroscope remains constant, then

$$\psi = \frac{G_{ч.э} a}{C\Omega} \int_0^t n_x\, dt. \qquad (9.23)$$

Let us assume now for simplicity that the rocket rises vertically, i.e.,

$$j = j_x = \dot{v}\ \mathrm{and}\ n_x \approx \frac{\dot{v}}{g_0} + 1.$$

Then expression (9.23) gives

$$\phi \approx \frac{a_{x}\rho}{C\Omega g_0}(v + g_0 t),$$

whence it follows that in this case the angle of rotation of the gyroscope with respect to axis x' turns out to be linearly dependent on the speed of rocket flight in a given moment of time.

If the time of flight of the rocket t is known, then it is possible to find the rocket speed in this moment by angle ϕ.

The integrator is tuned to a determined distance by turning the dial for adjusting distance 10. Here disk 11, having a contact device, turns to needed angle ϕ about cams 8.

When the rocket attains a given speed, one of the cams 8, making contact, will transmit a signal which will lead to switching the engine to the final step, and then the second – to full cut-off.

During the work, the principal axis of the gyroscope should constantly remain perpendicular to axis x'. So that this condition is fulfilled, a correcting device is foreseen in the integrator.

If the principal axis of the gyroscope will deviate upwards or downwards, the upper or lower of contacts 14 will be made, and a signal of corresponding sign will proceed to the motor 15. Here through the toothed wheels a moment about axis x' will act on the gyroscope. Such a moment, as we already know, will not accelerate and will not delay rotation of the gyroscope with respect to axis x', i.e., the conditions of integration will not change, and will only force the gyroscope to turn with respect to axis 5 and to occupy the proper position.

In the general case of slanted flight, the integral from the

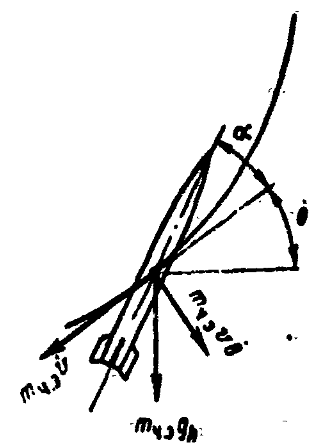

Fig. 9.40. Determination of apparent axial acceleration.

axial overload will be connected with the speed by a more complicated dependence.

As can be seen from Fig. 9.40, the apparent force of weight acting in flight on the rotor of the integrator in an axial direction, is

$$G_{s.e}n_x = m_{s.e}[\dot{v}\cos\alpha + v\dot{\theta}\sin\alpha + g_h\sin(\theta+\alpha)],$$

where $m_{s.e} = G_{s.e}/g_0$ – mass of gyro rotor with housing. Hence

$$n_x = \frac{1}{g_0}(\dot{v}\cos\alpha + v\dot{\theta}\sin\alpha + g_h\sin\varphi)$$

and

$$\int_0^t n_x dt = \frac{1}{g_0}\int_0^t (\dot{v}\cos\alpha + v\dot{\theta}\sin\alpha + g_h\sin\varphi)\, dt \approx$$
$$\approx \frac{1}{g_0}\left(v + \int_0^t g_h\sin\varphi\, dt\right). \qquad (9.24)$$

since angles of incidence on the active section are small.

The angle of rotation of the gyroscope with respect to axis x' in this case will be

$$\beta \approx \frac{G_{s.e}}{C\Omega g_0}\left(v + \int_0^t g_h\sin\varphi\, dt\right).$$

Thus, we see that the integrator of axial overloads does not give the speed of rocket flight in pure form. For its adjustment for a definite distance it is required that we know the magnitude of the integral $\int_0^t g_h \sin\varphi\, dt$, which is calculated in accordance with the given program of pitch angle and the calculated time of flight. Since in flight a program is given with certain errors, then these errors in some measure affect determination of rocket speed v by the integrator. The latter is a deficiency of integrators of axial overloads, since it leads to an additional methodical error during control of distance.

Structure and Work of an Electrolytic Integrator of Axial Overloads

Besides gyroscopic integrators, for range control of ballistic rockets electrolytic integrators of axial overloads are used.

The circuit of an electrolytic integrator is represented on Fig. 9.41.

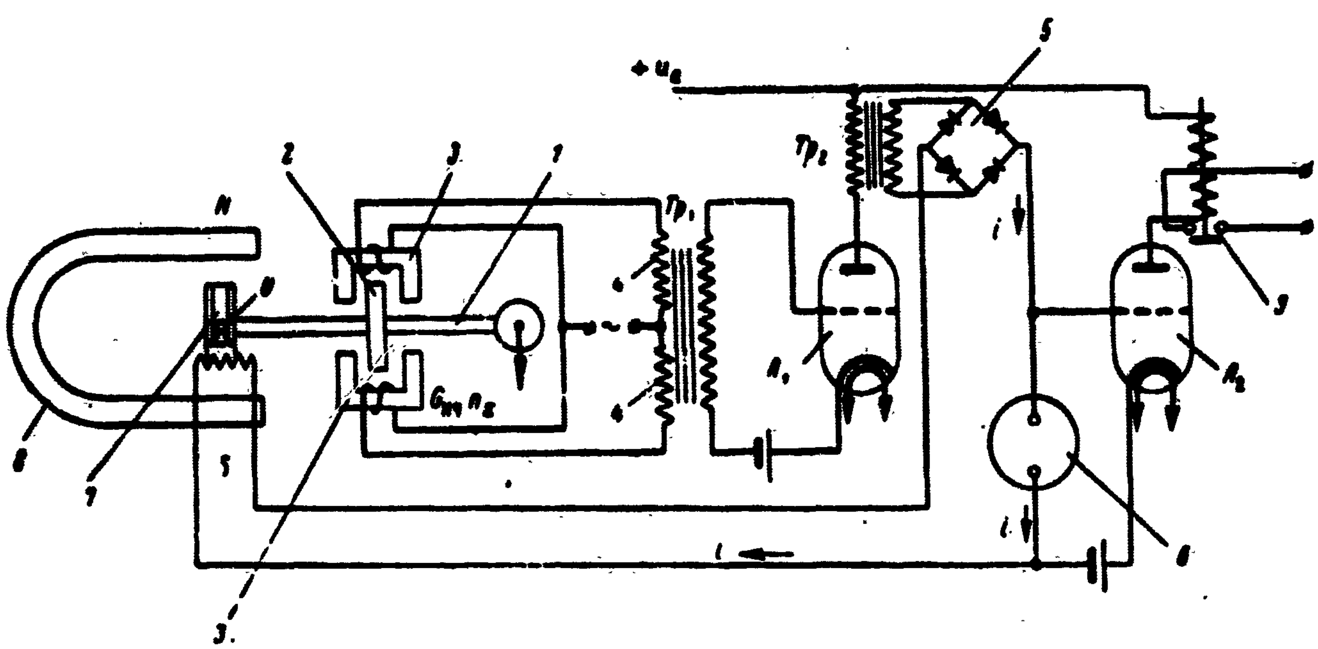

Fig. 9.41. Circuit of electrolytic integrator of overloads. 1 – lever; 2 – flag; 3 – electromagnets; 4 – winding; 5 – rectifiers; 6 – electrolytic element; 7 – balancing coil; 8 – permanent magnet; 9 – contact of relay.
[Л = T = Tube; тр = tr = transformer]

A lever 1, having a load $m_{s.e}$ on its end, is secured by a hinge at point O. On the lever a copper flag 2 is fastened in the gap between the two electromagnets 3. Windings of these electromagnets are fed by alternating current with a frequency of 500 cycles per second. The windings of the transformer Tr_1 are included in such a way at a neutral position of the lever, and consequently also of the copper flag, magnetic fluxes in the transformer from the windings 4 mutually destroy one another.

Before takeoff of the rocket, lever 1 is arrested and is in neutral position. At takeoff the lever is liberated, and under the action of inertial force $G_{s.e}n_x$, load $m_{s.e}$ drops downwards; the balance of the bridge is disturbed and in the secondary winding of transformer Tr_1 appears an alternating voltage, which is strengthened by tube T_1. The plate current of this tube induces an alternating voltage into the secondary winding of transformer Tr_2. This winding is included in one circuit with rectifier 5, electrolytic elements 6 and balancing coil 7 rigidly fastened to the lever 1. The magnetic flux, created by the current in the winding of coil 7 interacts with the field of the permanent magnet 8 and gives a moment, opposite to the moment of inertia on the lever. Thus, in flight lever 1 is in equilibrium. With an increase of overload n_x deflection of the lever from neutral position increases, and current intensity, passing through the balancing coil and electroyltic element 6, is increased correspondingly. Current intensity i turns out to be proportional to magnitude n_x:

$$i = An_x$$

Then the magnitude of the electrical charge q passed through the electrolytic element n_x is equal to

$$q=\int_0^t i\,dt=A\int_0^t n_x\,dt\approx\frac{A}{g_0}\left(v+\int_0^t g_k\sin\varphi\,dt\right).$$

The electrolytic element is used for integration by time of current intensity i and production of voltage impulses in the moment of achievement of speed by the rocket very close to that given. For this purpose silver chloride elements are applied usually, two for every integrator. The electrolytic element is a glass retort filled with an electrolyte and having two soldered silver electrodes (Fig. 9.42).

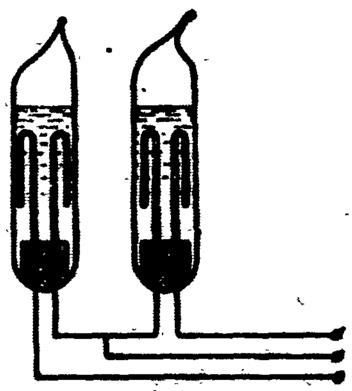

Fig. 9.42. Electrolytic elements of an integrator.

As an electrolyte a solution of table salt NaCl, sodium acetate CH_3COONa, and acetic acid CH_3COOH is used.

With connection of the electrodes to a source of direct current chlorine ions shift to the anode and, being connected with the silver electrode, will form a layer of silver chloride on the electrode:

$$Ag+Cl^-=AgCl+e^-.$$

On the other electrode at this instant evolution of hydrogen occurs.

The quantity of separating on a silver chloride electrode depends directly on the charge passed through the element. If one were now to pass a current in the opposite direction, on the first electrode a restoration of silver chloride will occur, and chlorine ions again will go into solution. On the second electrode a layer of silver chloride will be formed, as took place on the first electrode. In the moment when the reserve of silver chloride of the first electrode is completely restored, evolution of hydrogen will start and the internal resistance of the element will be changed. Voltage will be changed approximately by one volt. This jump in voltage is used as a signal indicating the fact that a definite quantity of electricity passed through the element.

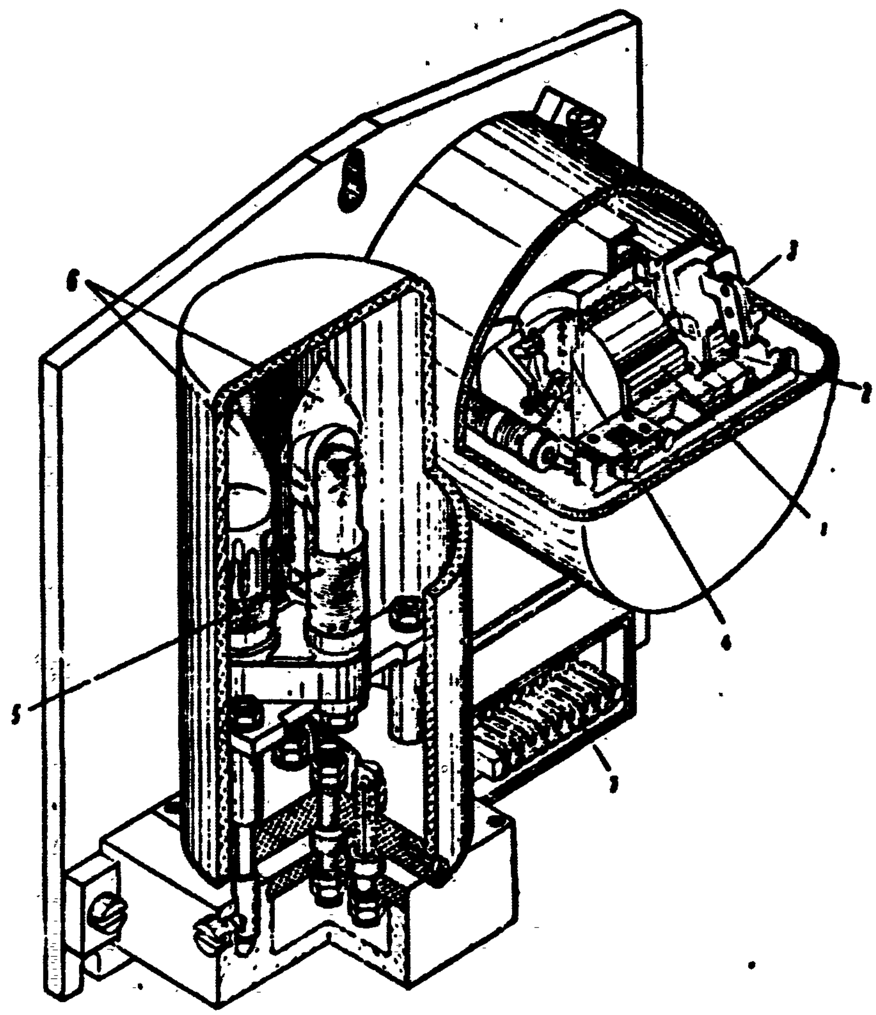

Fig. 9.43. General form of an electrolytic integrator. 1 – lever; 2 – flag; 3 – electromagnet; 4 – balancing coil; 5 – thermo-relay; 6 – electrolytic elements; 7 – plug block.

On the circuit of Fig. 9.41 it is clear that a change in voltage drop on the element is amplified by tube T. The relay 9 passes the signal for shift of the engine to the final step, i.e., to a lowered thrust. A signal from the second electrolytic element, not shown on the circuit, corresponds to complete turning off of the engine.

From the above it is obvious that adjustment of the sighting range is produced in an electrolytic integrator in accordance with the magnitude of charge preliminarily passed through the elements.

Fig. 9.43 shows the general form of an electrolytic integrator.

7. Electropneumatic Equipment and Work of the Automaton of a Ballistic Rocket During Starting and in Flight

Sources of Power Supply Aboard and Cable Network

Basic source of power supply aboard a ballistic long-range rocket are storage acid batteries with great capacities, able to discharge a current of order of several tens of amperes during a short interval of time. The most important peculiarity of these batteries are their comparatively small weight and small dimensions. The specific capacity of the batteries is about 10 ampere-hours per kilogram of weight. These qualities are attained by application of a thin plate surface and thin active layer.

Storage batteries on the V-2 rocket give a constant voltage of 27 volts. From these batteries, are fed electric motors of the steering machines, numerous relays, the program current-distributive unit and, finally, three converters, producing a variable three-phase current of a frequency of 500 cps and a voltage of 40 volts.

The converter is an electric motor working on direct current, and a generator of alternating voltages, mounted on a general shaft. As the motor a compound motor is used with the basic and additional

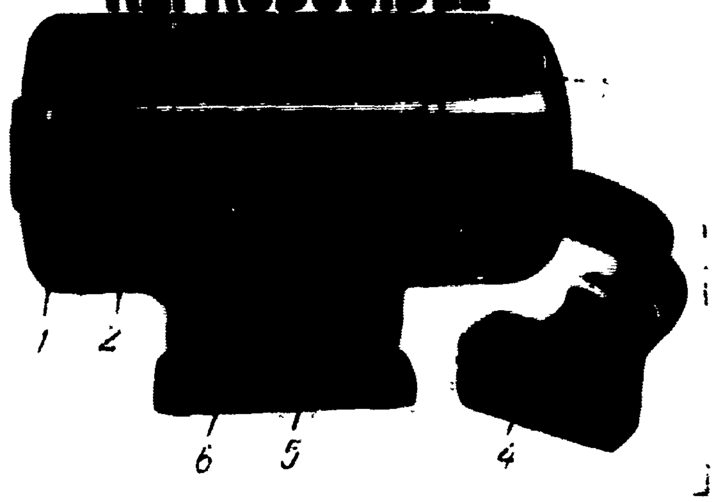

Fig. 9.44. Appearance of a converter. 1 — shield of converter on the generator part; 2 — yoke; 3 — protective cover; 4 — plug socket; 5 — plug fork; 6 — charge.

windings of excitation not connected among themselves. On the shaft of the motor there are three pairs of permanent magnets which induce an alternating voltage in the winding of the stator of the synchronous generator. To support the constancy of the number of engine revolutions, the converter is supplied by a frequency regulator, which acts on the motor through an additional winding of excitation. The appearance of a converter is shown on Fig. 9.44.

Alternating voltage is used for feeding the converter amplifier and for rotation of rotors of the gyro instruments.

Aboard the V-2 rocket there is also a source of constant voltage of 50 volts — a so-called team battery from dry cells of small capacity. From this battery only the potentiometers of gyro instruments are fed.

A skeleton diagram of the power supply of instruments aboard is represented on Fig. 9.45.

Wires from supply sources aboard are brought into different units through a main distributor located in quadrant II of the instrument section. In the external sheath of this section there are two hatches with automatically closed covers. After positioning the rocket on launching pad through these hatches two cables are connected to the main distributor, the cables connecting the rocket with ground feeding installations for pre-launching tests. Before launching, all the rocket equipment is fed from ground sources. At launching, the cables connecting the rocket with land are automatically disconnected and are ejected from the hatch.

The general plan of the cable network by quadrants and sections is represented on Fig. 9.46.

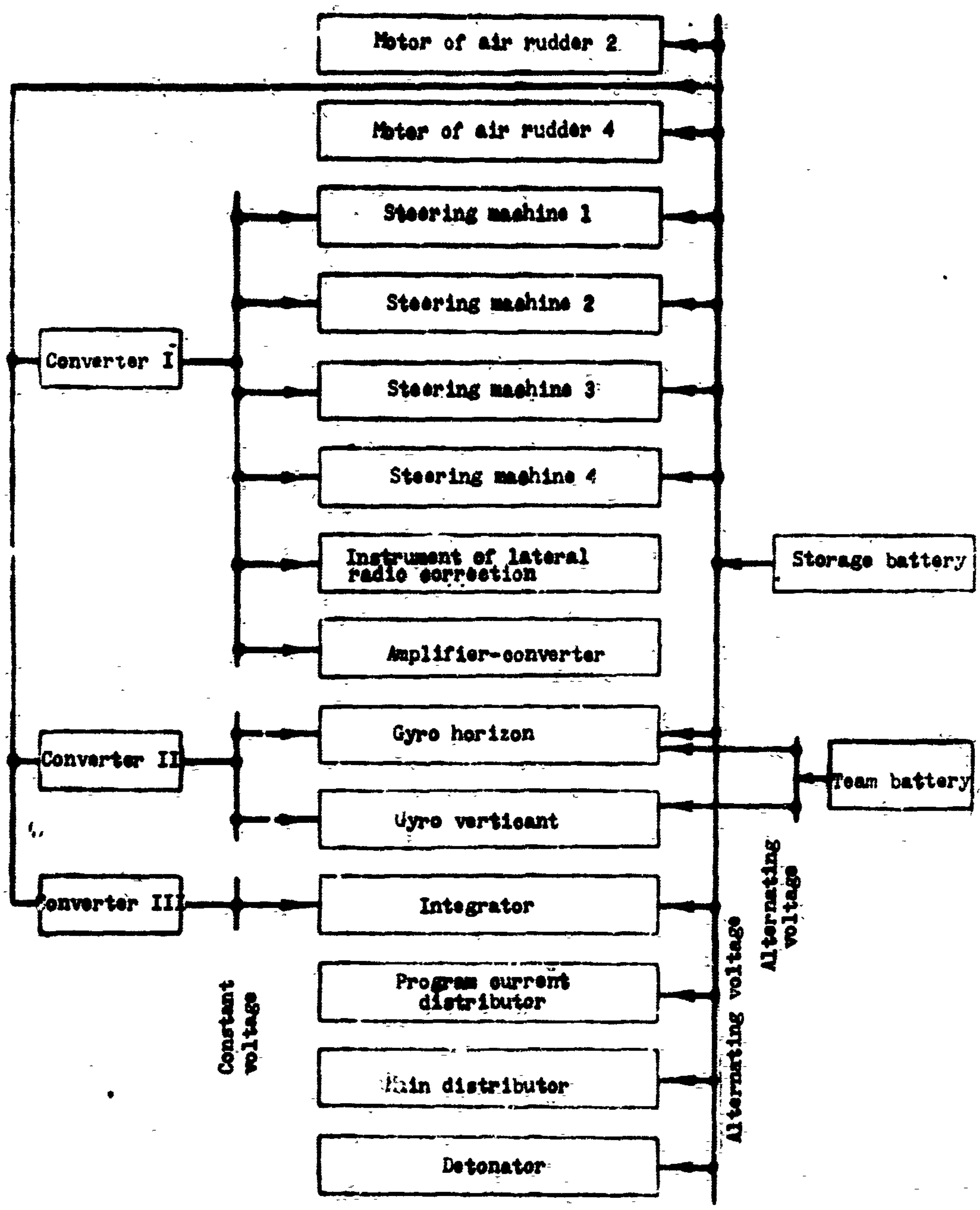

Fig. 9.45. Skeleton diagram of a V-2 rocket power supply.

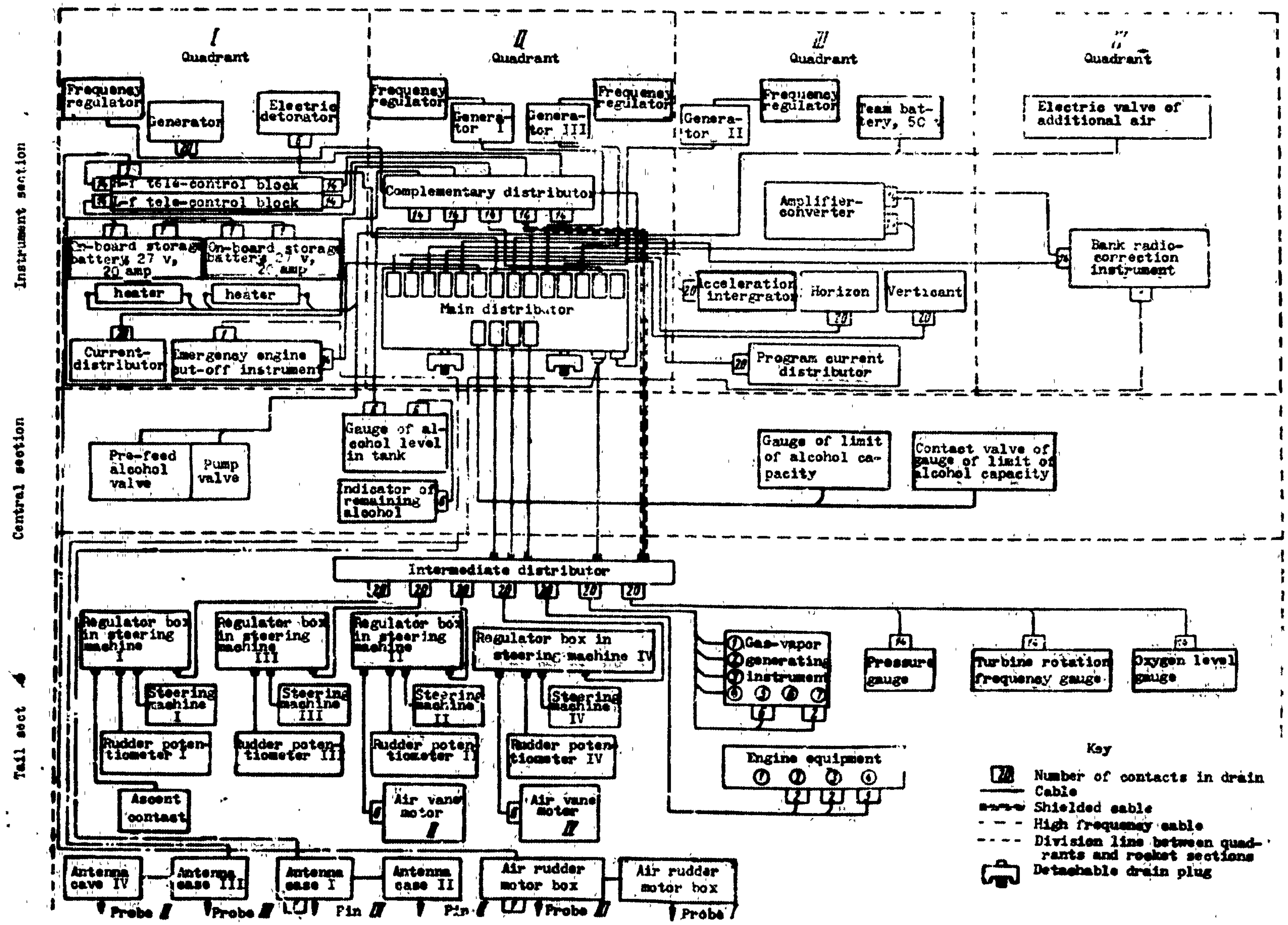

Fig. 2.46. Plan of cable network by quadrants and sections of a V-2 rocket.

The Pneumatohydraulic System in the Period of Prelaunching Preparation

A ballistic long-range rocket, along with the electrosystem, has aboard a pneumatohydraulic system. Automation of these two systems is mutual and connects together all operations, produced aboard the rocket both before start and during flight.

The source of pneumatic feeding of a V-2 ballistic rocket is seven-component battery 1 (Fig. 9.47).

To the rocket, fixed on the launching pad, through a five-branched split block 2, 3 a pneumatic shield is connected with a system of valves and manometers. Air under pressure of 200 atm (tech.) is given to the seven-component battery from the ground compressor through the main valve 4 of the pneumatic shield, the split block 2, 3, and the manual stop valve 5. Pressure is controlled by the manometer 6.

Simultaneously with filling of the seven-component battery 1, air from the main valve of the pneumatic shield 4 proceeds to the reductor of the pneumatic shield 7, where its pressure is lowered to 35 atm (tech.). Further, through the electropneumatic valve of the pneumatic shield 8 reverse valve 9, electropneumatic valves 10 and 11, of the system of pressure feed and filling of the alcohol tank and through normally open electropneumatic control valves by the main valves of the components 13, air gets to the main alcohol 14 and main oxygen 15 valves and closes them.

The three-component battery 16 is intended for creation of pressure in the alcohol tank after engine cut-off. It is filled by nitrogen before positioning of the rocket on the launching pad.

After filling of the pneumatic system by air, tests are made of the electric and pneumatic system, and then servicing of the rocket starts.

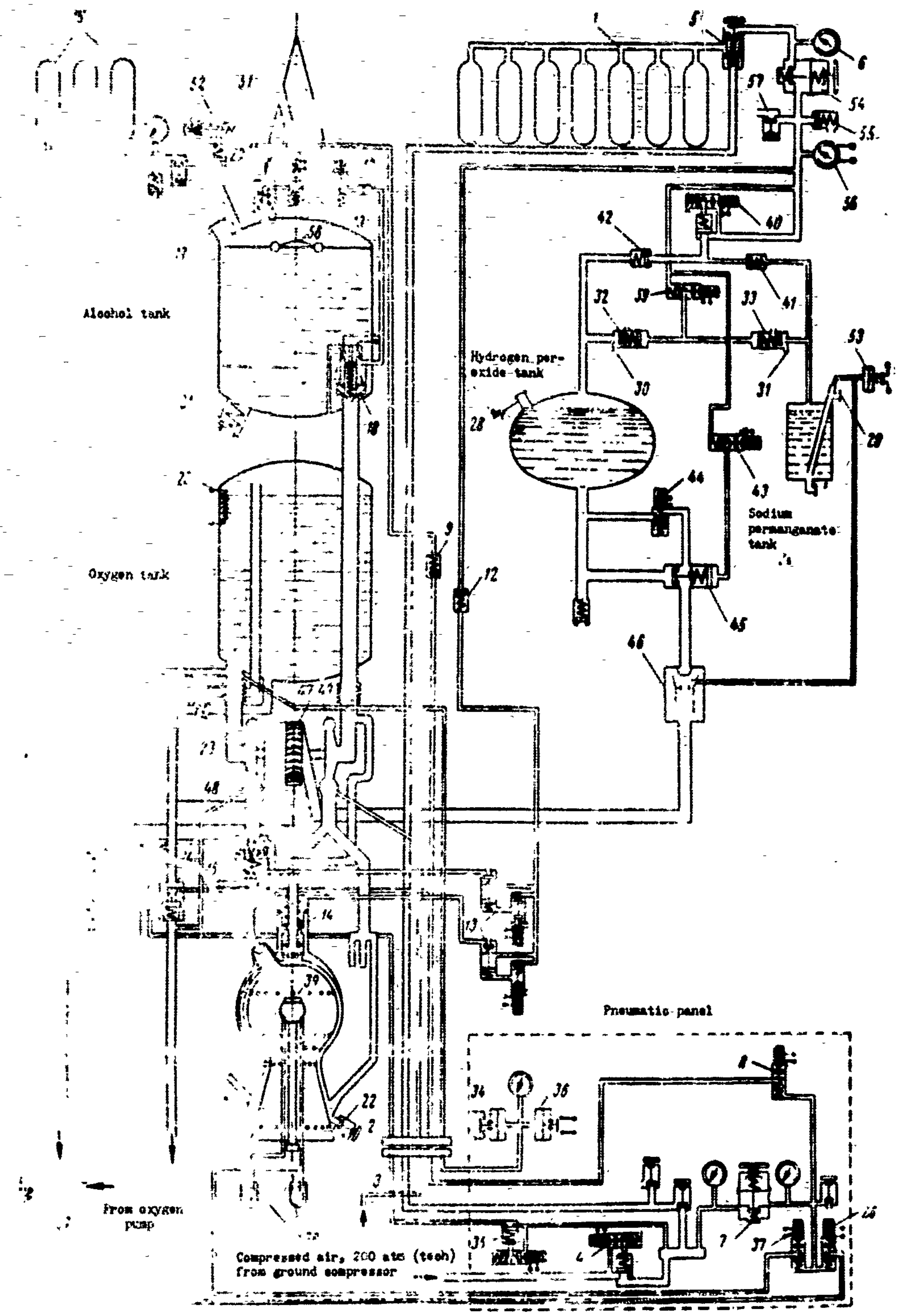

Fig. 9.47. Pneumatic-hydraulic system of a V-2 rocket. 1 - seven-component battery, 2, 3 – five-branched split block; 4 – main valve of pneumatic shield; 5 – manual closing valve; 6 – manometer; 7 – pneumatic shield reductor; 8 – electropneumatic valve of pneumatic shield; 9 – reverse valve; 10, 11 – control electropneumatic valves of the system of inflation and filling of the alcohol tank; 12 – reverse valve; 13 – electropneumatic control valves of the main valves of the components; 14 – main alcohol valve; 15 – main oxygen valve; 16 – three-component battery; 17 – servicing branch pipe of alcohol tank; 18 – valve of filling of alcohol main line; 19 – throttle; 20 – valve of pressure feed of the alcohol tank by impact pressure of atmospheric air; 21, 22 – drain valves of alcohol; 23 – valve for filling of liquid oxygen; 24 – drain valve of oxygen; 25 – transducer level of liquid oxygen; 26 – control electropneumatic valve; 27 – valve of oxygen feed; 28 – servicing branch pipe of peroxide tank; 29 – servicing branch pipe of sodium permanganate tank; 30, 31 – drain branch pipe; 32 – drain valve of peroxide tank; 33 – drain valve of sodium permangante tank; 34 – relay of pressure feed of liquid oxygen tank; 35 – electropneumatic valve; 36 – blocking relay; 37 – valve of pressure feed of tanks in the system of ignition; 38 – tanks with spontaneously inflammable components of ignition fuel; 39 – atomizer head of igniting device; 40 – main valve of Vapor-Gas Generator; 41, 42 – return valves of permanganate and peroxide, 43 – electropneumatic control valve of the main step; 44 – valve of final step; 45 – valve of main step; 46 – reactor; 47 – turbine; 48, 49 – pumps; 50 – heat exchanger; 51 – sampler for pressure feed of alcohol tank; 52 – electropneumatic valve; 53 – pressure relay in supply system of sodium permangante; 54 – reductor of Vapor-Gas Generator; 55 – valve for pressure drop; 56 – blocking manometer; 57 – valve of reductor adjustment; 58 – generator level of alcohol; [illegible] – electropneumatic control valve.

Alcohol moves into the tank through servicing branch pipe 17. When the tank is filled to 1/4 capacity, valve 18 opens, and alcohol fills the alcohol main line to the main alcohol valve 14. Valve 18 is opened under action of compressed air which is sent by the electropneumatic valve 10 through throttle 19. After filling of the alcohol main line, valve 18 is closed.

During filling of alcohol, air from the tank and filled main lines emerges through the valve of pressure feed the alcohol tank by impact pressure of atmospheric air 20. If starting does not take place, alcohol from the tank and main lines passes through drain valves 21 and 22.

Supplying of alcohol into a V-2 rocket continues for about 30 minutes. The limiting level is fixed by the gauge of the alcohol level 58.

After supplying of alcohol servicing of the rocket by liquid oxygen through servicing valve 23 starts. Servicing continues as long as oxygen does not flow through the open oxygen drain valve 24. Through the same valve evaporated oxygen leaves the tank before launching. The level of oxygen in the tank before start is maintained automatically. The gauge for the level of liquid oxygen 25 is connected with electropneumatic control valve 26 which passes air for opening or shutting of the valve of feed of liquid oxygen 27.

Tanks of permanganate peroxide refuel through the branch pipes 28 and 29. During servicing of these tanks, air emerges from them through branch pipe drains 30 and 31. Here drain valves 32 and 33 are open.

After termination of servicing the stop valve 5 is manually opened, and the reductor of the Vapor-Gas Generator 54 is tuned on a definite magnitude of feed pressure of hydrogen peroxide into the reactor.

During adjustment, the main valve of the Vapor-Gas Generator 40 is closed, and exit of air occurs through an open valve of adjustment of the reductor 57. On one of the branch pipes of this valve a precise control manometer is attached, by which is produced adjustment of the reductor of the Vapor-Gas Generator. After adjustment, valve 57 is closed, and the control manometer is removed.

Shortly before start the igniting device is established in the chamber.

With this preparation for launching is completed. Subsequent commands are transmitted remotely.

Pneumatic-hydraulic System at Launching

Several minutes before launching the gyroscope rotors are started. Then a command is transmitted to shut the drain valves; 24 — oxygen tank (see Fig. 9.47), 32 — hydrogen peroxide tank and 33 — tank of sodium permanganate. Drain valves are closed by compressed air which is sent by controlling electropneumatic valves 8 and 59. In the liquid oxygen tank a small excess pressure is established and subsequently maintained. As a transducer of pressure serves the relay of pressure feed of the liquid oxygen tank 34, controlled by the electropneumatic valve 35 connected with the ground network of compressed air.

Further, the system is blocked by relay 36 and manometef 56. In case of insufficient pressure in the oxygen tank and in the main lines of pressure feed of the Vapor-Gas Generator starting does not take place.

The following command is given for opening of the valve for filling of the alcohol main line 18. This is attained by a supply of compressed air from the controlling electropneumatic valve 10. Simultaneously, a command is transmitted for ignition. On Fig. 9.47 shows a diagram of chemical ignition. Compressed air through the valve of pressure feed of tanks 37 enters tanks 38 with spontaneously inflammable components and displaces them in the atomizer head of the igniting device 39. In the combustion chamber of the rocket engine an ignition flame jet will be formed. In the beginning of ignition a magnesium tape burns, stringed in the combustion chamber. This tape is included in the circuit controlling future starting of the propulsion system. As a result of burnout of the tape clearing of the starting system occurs.

Then the electrical current passes onto the winding of the electromagnet of one of the electropneumatic control valves 13 (on the lower circuit). This valve closes access of controlling compressed air to the main oxygen valve 15, and compressed air which earlier blocked the main oxygen valve is released into the atmosphere. Main oxygen valve 15, under the action of excess air pressure in the liquid oxygen tank and pressure of the liquid oxygen column, somewhat opens (as it is said, it is opened at a preliminary step) and oxygen by gravity (with an idle pump 48 of the turbopump unit) enters the combustion chamber. Opening of main oxygen valve 15 on the preliminary step evokes closing of the electro contact mounted in valve 15. Owing to this, the current proceeds from the electromagnet of the second (on upper circuit) electropneumatic control valve 13. This valve acts analogously to the electropneumatic control valve of the main oxygen valve. Access of control air is ceased in the main alcohol valve 14, and compressed air existing there escapes into the atmosphere. Under pressure of the alcohol column in the tank and in the main lines, the main alcohol valve is opened (it is opened on the preliminary step), and alcohol with an idle pump 49 enters the combustion chamber of the rocket engine by gravity.

Thus, at starting of the engine at first the main oxygen, and then the main alcohol valves are opened. The fuel components enter the combustion chamber by gravity with a very small flow rate and reliably ignite by an ignition flame jet created by the ignition device. Heating of the chamber and strengthening of the intensity of burning of the ignition torch occurs.

The next command is transmitted to the electromagnet of the main valve of the Vapor-Gas Generator 40, which is opened, and through reverse valves 41 and 42 passes compressed air into the tanks of hydrogen peroxide and permanganate. With an increase of pressure in the supply line permanganate (and this indicates that permanganate is already entering reactor 46) a pressure relay 53 works, which is included in the circuit of electropneumatic valves 43 and 44. Valve 43 regulates the work of the main step valve 45.

At closing of relay 53 immediately the final step valve 44 is opened, as is valve 43, the regulator of the main step valve 45, which is opened 0.4 sec after opening of the final step valve. The flow rate of hydrogen peroxide through both valves 44 and 45 corresponds to engine thrust at the main step (25 tons on earth). With a closed valve 45 the flow rate of peroxide corresponds to an engine thrust at the final step (8 tons).

Hydrogen peroxide and sodium permanganate enter the reactor 46, where formation of vapor gas occurs. The vapor gas mixture proceeds to turbine 47, from which centrifugal pumps 48 and 49 are put into motion. Pressure developed by the pumps gradually opens the main component valves 14 and 15 to total passage section. In several seconds engine thrust attains nominal. When the increasing thrust becomes equal to the weight of the rocket, its breaking away from launching pad occurs; as a result, the split block 2, 3 is disconnected.

Automation of a Rocket in Flight

At breaking away of the rocket from the launching pad, the so-called contact of rise works, fixed near one of the supporting points of the rocket, fixing the initial moment of its ascent. A signal from this contact is the first command aboard, including a number of devices.

During operation of the rise contact there occurs a disconnection of the electromagnets holding the break-away plugs that connect to the rocket through the main distributor two cables, bringing in an electrical current from ground feeding installations. Under action of the force of weight and springs compressed during installation the plugs are disconnected and ejected together with the ends of the cables through hatches outside, and the covers of the hatches slam shut. The rocket changes from ground to board feeding.

By a signal of the rise contact de-arresting occurs of the integrator of overloads, which from this moment begins functioning. Finally, by the same signal the program current distributor (PCD) starts to work.

The program current distributor is intended for supply of controlling commands in a definite time sequence to the group of instruments on board the rocket. The basic of these commands are the following:

1) Commands connected with the work of the program mechanism.

After several seconds of vertical flight PCD induces functioning of the program mechanism of the gyro horizon. For feeding of the step motor by a program mechanism, in the vibrator of PCD alternating current is produced with a frequency of 45 cps. After the program is worked out and it is required to sustain the rocket at a given angle φ_k, PCD will turn off the program mechanism.

2. Commands of control of detonator devices or program control of measuring apparatus.

3. The program current distributor transmits commands connected with changes of conditions in the pressure feed of tanks. Thus, for a V-2 rocket in the 40th second valve 20 for pressure feed of the alcohol tank will be turned off by impact pressure of atmospheric air, but after turning off of the engine, PCD gives a command to the electropneumatic valve 52, through which filling of this tank by nitrogen from the three-bottle battery 16 occurs.

Depending upon the peculiarities of the rocket and its board automation of commands, PCD can essentially be changed.

Structurally, the program current distributor is an electric motor of direct current, connected through a gear reductor with a cam shaft. During rotation of the shaft the cams in a determined sequence make contacts, thus giving the beginning of corresponding commands.

The program current distributor of a V-2 rocket with removed covers is shown on Fig. 9.48.

In flight, pressure feed of the oxygen tank of a V-2 rocket is produced by gasified oxygen, proceeding from heat exchanger 50 (see Fig. 9.47). Pressure feed of the alcohol tank before the 40th second is produced by impact pressure through intake 51. After lowering of impact pressure, valve 20 is closed.

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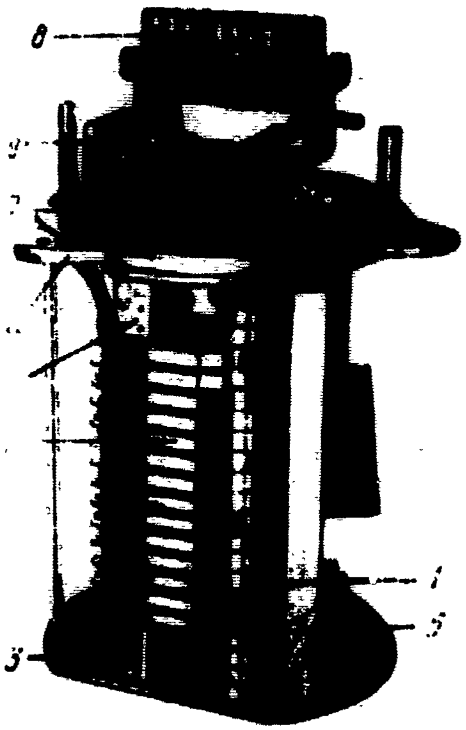


Fig. 9.48. Program current distributor. 1 – program shaft; 2 – contact group of 14 pairs of contacts; 3 – lower charge; 4 – upper charge; 5 – stand; 6 – stand for fastening current carrying components; 7 – toothed reductor; 8 – 20-poled plug; 9 – relay.

When the rocket speed approaches the given, from the integrator of overloads passes a signal to the electropneumatic control valve of the main step 43, as a result of which main step valve 45 is closed. The engine passes to an eight ton thrust.

By the main command for engine cut-off passed from the integrator when the rocket speed is equal to the given, valve 44 and the vapor gas generator are closed, and after it the engine ceases to work.

Simultaneously:

1. Windings of both electropneumatic control valves 13 are disconnected, and by compressed air the main component valves 14 and 15 are closed.

2. Electropneumatic control valve 10 is disconnected. Compressed air proceeds through the throttle 19 to the valve for filling the alcohol main line 18 and slowly closes it.

3. Electropneumatic valve 52 is opened, and compressed nitrogen from the three-bottle battery fills the empty alcohol tank in avoidance of its crumpling at entrance into dense layers of the atmosphere.

From the considered typical sequence of operations during work of the pneumatic-hydraulic system, it is clear that the equipment of a ballistic long-range rocket has a high degree of automation. Correct work of all parts of automation of the electropneumatic system is a necessary condition of normal starting and flight of the rocket.

8. Methods of Guiding Controlled Rockets at a Target

Guiding with the Help of Commands

Guiding is understood as all those actions which have to be started during rocket flight so that the rocket gets to the target. Guiding can be produced both by the gunner by remote control and automatically. If a guiding automaton is on board the rocket, then such guiding is called a homing guidance system.

Guiding is necessary for firing both at mobile and motionless targets. In both cases it allows during rocket flight elimination of those errors, which are peculiar to firing of uncontrolled missiles, and furthermore, gives the possibility to strike a fast moving target independently of maneuvers started by it. In those cases when guiding is intended for introduction of small corrections in the flight trajectory, it is called a correction system.

Of a number of cases the simplest is a guiding system with the help of commands. Here a missile moves on a trajectory prescribed to it from without, primarily depending upon motion of the target. In connection with this, the guiding system anticipates simultaneous control both of the missile and of the target.

Guiding with commands, depending upon the method of tracking, divides into systems with visual (or optical) tracking and radar.

Optical tracking is applied at comparatively small distances of firing under conditions of direct visibility of the target and missile. During rocket flight the gunner manipulates the aiming handle, with the deflection of which from neutral position the duration of neighboring transmitted impulses is changed. The more the handle deviates in one or another plane, the bigger this difference, and the signal for turning of rudders in the corresponding plane becomes all the more strong. The degree of necessary deflection of the handle is determined by observation of motion of the missile. During aiming, the missile and target are combined in the field of sight of the gunner. As a result, the missile turns out to be constantly on the line connecting the gunner with the target (Fig. 9.49), and moves by a so-called three-point curve (gunner – missile – target).

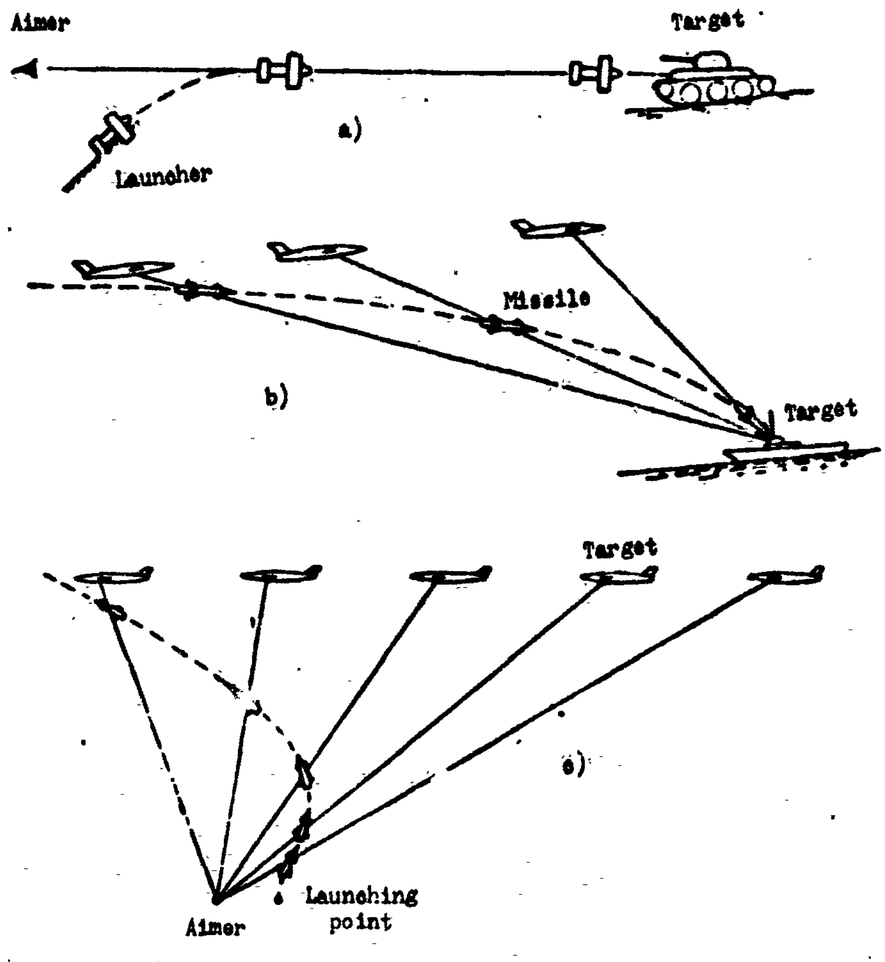


Fig. 9.49. Diagram of command guidance of missiles to a target by a three-point curve.

The control automaton of a projectile controlled with commands is supplied by an antenna for reception of aiming signals. These signals proceed to the operating organs (to steering machines) in parallel with signals produced by sensitive elements of the stabilization automaton. Control commands are sent to the missile either by wires or through a radio channel.

The system with radar tracking is more complicated. In it are used usually two locators: one is continuously trained on the target, and the other – on the missile. Tracking is automatic. Fast rotation is given for the radiating vibrator of the locator with respect to an axis formed by a small angle with the axis of radiation direction (Fig. 9.50).

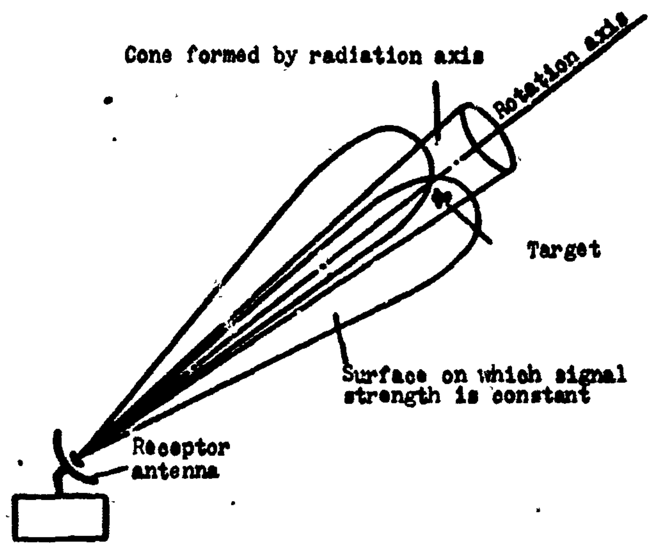


Fig. 9.50. Diagram of tracking by a target by a locator.

If the target is not on the vibrator axis there appears, so to speak, an error signal, and impulses reflected from the target become modulated by amplitude with the frequency of rotation (Fig. 9.51). Modulation percentage increases with deflection of the target from the rotation axis. This signal is strengthened, straightened, and its phase is compared with the phases of angular deflections of the vibrator in two planes. By the difference of phases a command is given to the servo-actuators, turning the locator antenna at an angle to elevation and azimuth in such a manner that the error signal decreases to zero and the axis is constantly

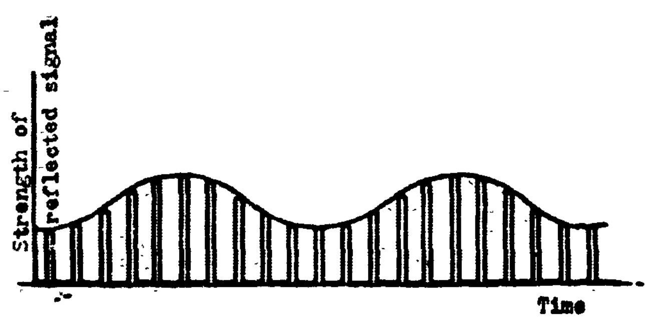


Fig. 9.51. Modulation of reflected impulses due to signal error.

directed at the target. In exactly the same way tracking after the missile is carried out.

A general diagram of guiding with two watching locators is shown on Fig. 9.52. Angles of the position of the target and missile, derivatives of these angles by time, and also the distance to the target and missile are transmitted from the locators to a computer, in which calculation of the necessary control commands is made. Through the control transmitter these commands pass to the missile.

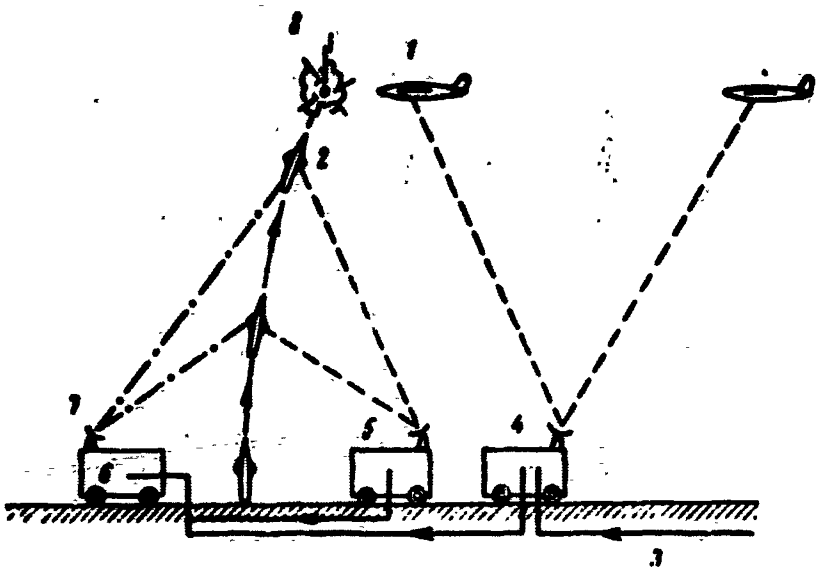


Fig. 9.52. Diagram of guiding with two locators. 1 – target; 2 – missile; 3 – communication with locator of early announcement; 4 – locator for tracking target; 5 – locator for tracking missile; 6 – computer; 7 – transmitter of control signal to rocket; 8 – point of projectile explosion.

In the described method of guiding, approach of the missile to the target is produced usually by the method of the anticipated point. A computer determines where the target will be at the moment when the rocket reaches it on the assumption that parameters of motion of the target measured in the given moment will not be changed before encounter. To this point (anticipation point) the missile is directed. Since during flight of the missile the target can maneuver, calculations are made continuously, and the position of the anticipation point changes, which is considered by introduction of corresponding commands.

Guiding with commands in the case of visual and optical tracking is the simplest. The deficiency of this system is its low effectiveness: during a more or less prolonged time the system is working for only one target and only one missile. Until tracking of the first missile is finished, it is impossible to release a second missile. In this sense direction by beam-rider guidance is better.

Beam-Rider Guidance

In a beam-rider guidance system one locator is used, continuously following the target. The missile is introduced into the following beam and subsequently before target impact is held in it independently. Motion of the missile, thus, occurs, as during visual guiding by a three-point curve (Fig. 9.53).

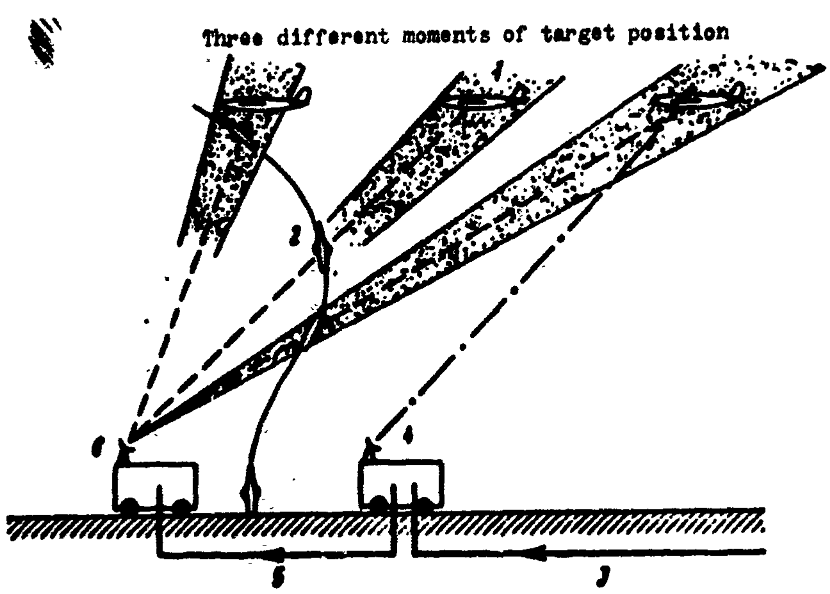


Fig. 9.53. Diagram of beam-rider guidance. 1 – target; 2 – missile; 3 – communication with locator of early announcement; 4 – locator for tracking target; 5 – synchronous connection; 6 – guiding locator.

The locator follows the target automatically. For that the principle mentioned above of creation of an error signal is used.

The missile moving along the beam has a receiver working on the carrier frequency of the locator. When the missile departs from the beam axis, then due to rotation of the radiating dipoles, the signal of the locator becomes modulated by amplitude as shown on Fig. 9.51. The intensified and straightened signal is compared in the rocket devices with reference to voltage, the phase of which is fixed through a special radio channel with the phase positions of the ground radiating dipole in two planes. Depending upon modulation percentage and phase shifts in the rocket devices, corresponding commands are made to the rudders of pitch and rove with such account so that the missile returns to the axis of the aiming beam. Stabilization of bank is carried out usually with the help of an autonomous system.

The described system of aiming has comparatively great effecitveness, inasmuch as several missiles can be sent by one beam at relatively small intervals. At the same time there exists a whole number of factors lowering the exploitational qualities of this system.

Due to imperfection of equipment angular oscillations of the radio beam inevitably appear, creating difficulty in retention of the missile on the guiding line. These difficulties are added to also by the fact that for obtaining necessary aiming accuracy the radio beam should be sufficiently narrow. In connection with the latter circumstance, the starting missile cannot be introduced immediately and directly into the narrow zone of coverage of control signals, and it is necessary to use additional means. As one of the possible measures for solution of this problem it is possible to use a second starting locator with a large beam angle.

The accuracy of guiding by radio beam is lowered with an increase of distance from the transmitter, and in the moment of nearing of the missile to the target becomes worse, when namely at this instant the highest requirements are presented to the accuracy of guiding. Therefore, the beam rider guidance system is usually combined in the last stage with a homing guidance system.

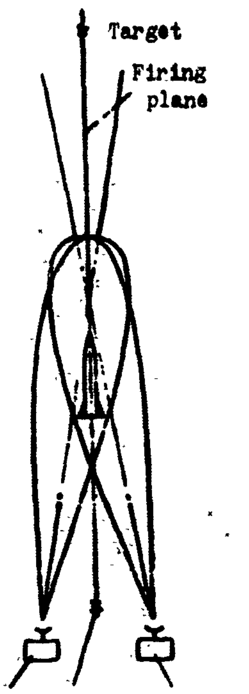


Fig. 9.54. Diagram of lateral correction of a ballistic long-range rocket.

One of varieties of beam-rider guidance is lateral radio correction, designed in our time by the Germans for ballistic rockets for the purpose of limiting rocket drift from the programmed firing plane. In order to exactly fix the firing plane in the ctive section of flight, two locators are used, working at various frequencies but giving synchronous impulses. The locators are positioned in such a manner so that the plane of equal signal strength coincides with the firing plane (Fig. 9.54).

The Semiactive Homing Guidance System

The logical development of the beam rider guidance system is the semiactive homing guidance system.

Inasmuch as the target is constantly illuminated by the locator, it is the source of reflected beams by which the missile is oriented (Fig. 9.55). In this case due to nearing the source of radiation, i.e., the target, aiming accuracy is increased.

The receiving antenna of a self-guided missile has a narrow zone

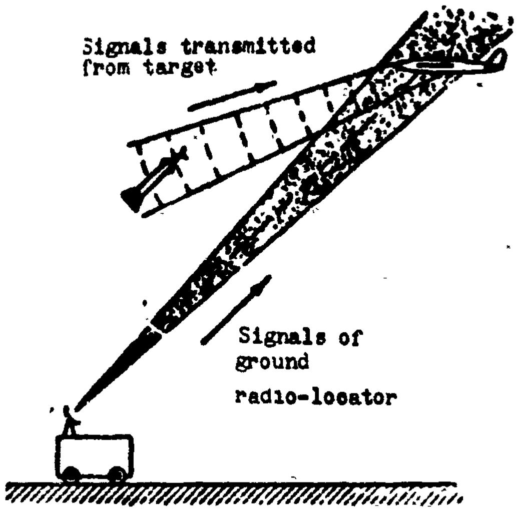


Fig. 9.55. Diagram of a semiactive homing guidance system.

of directed reception and is attached to the nose part of the missile. The antenna has freedom of angular shift and at transition to the homing guidance system carries out search and lock-on. Subsequently the antenna is continuously directed towards the target. When the target is "captured," the system of control produces turning of the missile in such a manner so that its axis coincides with the direction of the antenna axis. From this moment the missile moves on a pursuit curve, i.e., on such a curve, the tangent to which is constantly directed towards the target.

During maneuvers of the target the self-guided missile receives noticeable lateral overloads, the magnitude of which depends essentially on the method of approach to the target. From this point of view not straight guiding to the target, but following with a small lead angle is more favorable (Fig. 9.56).

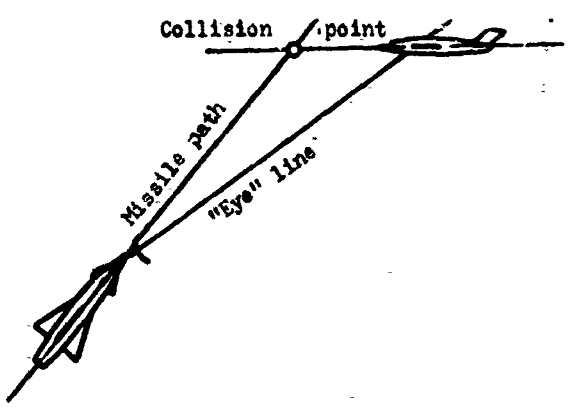


Fig. 9.56. Diagram of homing guidance system with an anticipation angle.

The semiactive homing guidance system, just a beam-rider guidance, has that deficiency that it is realizable only under condition of direct "visibility" of the target by the locator from land.

Active Homing

The system of active homing of interceptor missiles is the most complicated and most difficultly controlled of all other homing

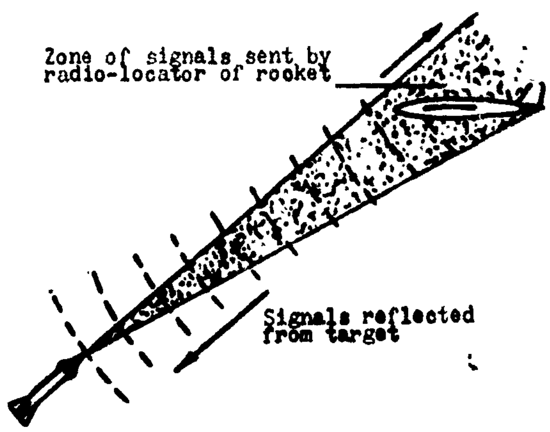


Fig. 9.57. Diagram of active homing.

guidance systems. In this system the missile itself irradiates the target and then is oriented by signals reflected from target without any sort of external help (Fig. 9.57).

In distinction from semiactive guiding, in the considered system simultaneous guiding of several missiles at several targets is possible. However, with active homing the missile should carry a transmitter which could send radio signals by an antenna directed forward. Furthermore, on board the missile there should be receiver and a control system which would direct the missile in the direction of the source of reflected signals. Obviously, the missile should carry all equipment of sighting and accompaniment of the target in an on board locator. If guiding is considered toward an anticipated point, the missile should also carry a computer. All this makes the missile very heavy and complicated. Therefore, at all other equal conditions, missiles with active homing have a relatively small range. The limitation of distance is connected also with the fact that the on board transmitter has comparatively low power, and the system is more susceptible to interferences.

Passive Homing Guidance System

The passive homing guidance system is such a system in which a receiver fixed on the missile uses energy radiated by the target itself (Fig. 9.58). Here guiding is possible by sound, light, by change of magnetic field, and so forth. The most important and most widely applied method is guiding by thermal (infrared) radiation.

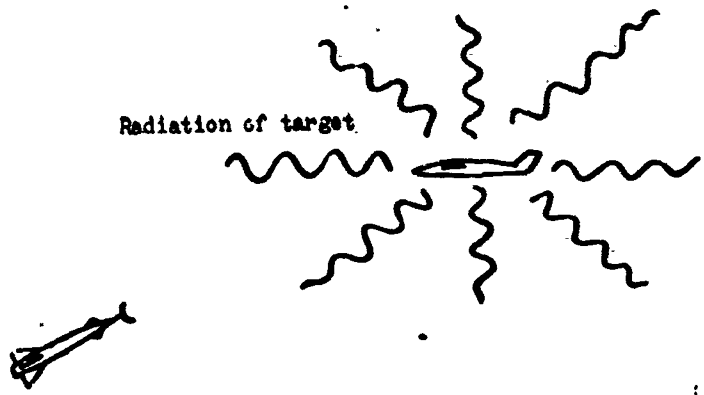


Fig. 9.58. Diagram of passive homing guidance system.

Infrared radiation is a band of electromagnetic waves or long light waves from 0.5 to 300 μ, lying beyond the limits of visibility of the human eye. Their source is primarily the exhaust gases of the engine by which the missile is oriented under a homing guidance system.

Infrared guiding heads are made usually for a definite, comparatively narrow, range of wave lengths. Thus, for instance, the temperature of exhaust gases of a turbojet engine is about $750^{\circ}C$. With this temperature thermal beams with wave length of 3 μ have the most energy.

With an increase of temperature of the radiating body, the intensity of radiation sharply increases, and the length of waves with the greatest energy somewhat decreases.

As indicators ("eye") reacting to infrared radiation, so-called thermal indicators are used. Their action is based on the fact that during illumination by thermal beams the electrical resistance of the indicator's sensitive layer noticeably changes. Usually two sensitive elements included in a bridge circuit are used in thermal indicators. Infrared radiation is directed to one element. The second ensures temperature compensation in case of change of ambient temperature. The material of the sensitive layer can be sulfurous lead, tellurous lead, selenious lead, selenious indium and certain other compounds. The guiding system of a missile works on the principle of retention of current in the sensitive element at a maximum.

The advantage of an infrared guiding system as compared to radar is lower susceptibility to artificially created interferences. Till now methods of concealment or camouflage of infrared radiation of aircraft are unknown. A homing device, working on the principle of infrared radiation, appears, as experiment shows, simpler and lighter than locating heads. Finally, the guiding system by infrared radiation has higher resolving power, i.e., ability to separate from a group of targets an individual target.

Infrared homing guidance systems also have a number of deficiencies. The basic deficiency creating great difficulties in designing and controlling equipment is the omnipresentness of infrared radiation and presence of a more or less intense infrared background. Thus, for instance, infrared radiation of the Sun, reflected from a rapidly variable overcast, can distract the missile from a target. Infrared radiation is absorbed in the atmosphere significantly more intensely than radio waves. Therefore, in heavy rain or fog the range of the infrared homing guidance system can sharply decrease.

In spite of these organic deficiencies, infrared technology at present is experiencing even greater application.

Finishing the short survey of guiding systems and homing guidance systems of rockets, one should note that in the actual structures all enumerated methods of guiding are continuously changed and perfected; they are the subject of a new, greatly developed region of technology. The conventional short classification of types of guiding given here is by far not exhausting and is able to give the reader only the most preliminary introduction to this question.

9. Dispersion in Firing of Rockets

Measure of Dispersion

For appraisal of the degree of accuracy in firing of rockets the very same indices can be used as in artillery practice.

Let us assume that firing is conducted at a certain target with an infinitely large number of identical rockets at constant conditions of starting (Fig. 9.59). Here, when we talk of "identical rockets" and "invariability," we understand by this only "similarity" and "invariability" only in those limits which can be ensured by technological considerations and by practical conditions of firing.

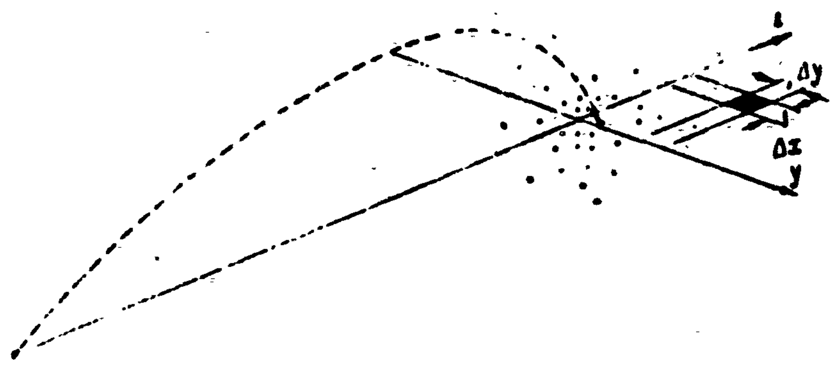


Fig. 9.59. Dispersion during firing of rockets.

In fact, geometric and weight characteristics of rockets will somewhat change from missile to missile, remaining within the limits of allowances. Conditions of burning of the charge in one rocket will be, although insignificantly, different from conditions of burning in another. The actual conditions of starting also will not remain constant. Accidental, let us assume that even absolutely inconspicuous, displacements of the launcher from shot to shot, an accidental gust of wind during flight of the rocket, in some measure will change the conditions of starting.

However, speaking of identical rockets and constant conditions, we at the same time consider that from technology of manufacture and equipment of rockets and conditions of starting, systematic errors and systematic deflections from certain mean values are excluded. For instance, inaccuracy of the controlling measuring tool is ignored, which could lead to systematic deflections of the geometric dimensions of the rocket or its charge from the norm. Wind speed is considered constant, affecting distance and so forth.

Thus, during firing of identical rockets at constant conditions of starting, deflection of rockets from the target will have an accidental character, and hits shown on Fig. 9.59, will be in a larger or smaller measure scattered about the target. With a large number of shots, probable deviations from the target and more exact distribution of missiles on the plane will obey a Gaussian distribution for random variables.

If on plane x - y (see Fig. 9.59) a rectangle with sides Δx and Δy, were isolated, then the quantity of missiles ΔN hitting this rectangle at a large number of shots will be proportional to the area $\Delta x \Delta y$:

$$\Delta N = f \Delta x \Delta y.$$

Proportionality factor f is a function of coordinates x and y:

$$f = f(x, y).$$

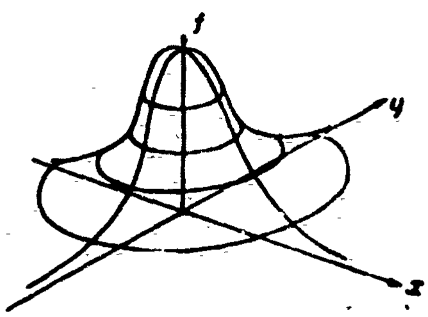


Fig. 9.60. Surface described by distribution function $f(x,y)$.

This is the distribution function. According to Gauss's law

$$f = Ae^{-k^2x^2 - h^2y^2}, \qquad (9.25)$$

where A, k and h are certain constants, where magnitudes k and h determine the degree of dispersion corresponding to axes x and y.

The surface described by equation (9.25) is shown on Fig. 9.60.

Function f has a maximum at the origin of the coordinates at $x = y = 0$ and becomes zero at $x = \pm\infty$ or $y = \pm\infty$.

Let us determine the locus of points of equal distribution. Considering $f = C = \text{const}$, from (9.25) we find

$$k^2x^2+h^2y^2=\ln\frac{A}{C},$$

i.e., the equation of an ellipse with axes coinciding with the coordinate axes. The dimensions of the semiaxes of the ellipse depend on magnitude of C. Thus, the family of "horizontals" of surface $f(x, y)$ is a family of ellipses of dispersion (Fig. 9.61).

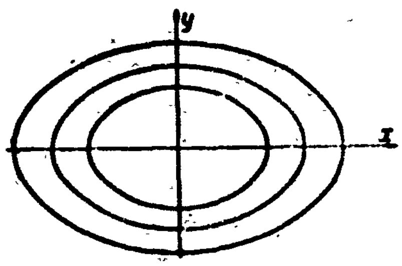


Fig. 9.61. Family of ellipses of dispersion.

For measure of dispersion of missiles during firing, the dimensions of the axes of one of these ellipses can be used, for instance the ellipse inside which at a large number of shots half of the missiles land, i.e., the ellipse of the best half.

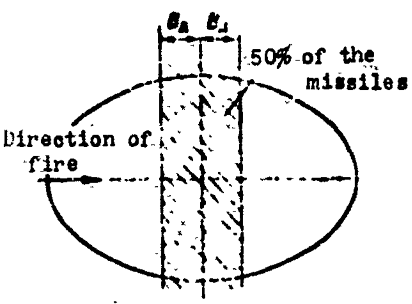


Fig. 9.62. Determination of B_{1e}. [Д = le = length; б = la = lateral].

In artillery practice, as a measure of dispersion not the dimensions of an ellipse, but a bandwidth is used, within the limits of which at large number of shots half of the missiles land. The dimensions of such a band in the direction of firing (by length) are designated by $2B_{1e}$ (Fig. 9.62), and in the lateral direction —through $2B_{1a}$.

If during firing systematic errors are not eliminated, the center of the dispersion ellipses will not coincide with the target. Deflection of the center of ellipses from the target characterizes the accuracy of firing.

Dispersion of Unguided Rockets

At present, dispersion of unguided rockets exceeds dispersion of the usual artillery missiles by a few times. This is explained mainly by the fact that the powered flight section of an unguided rocket, i.e., that section, in which it gains speed, is not sufficiently stable while an artillery missile in the section of acceleration, i.e., in the gun barrel, has a strictly given direction of flight. This is confirmed also by the fact that unguided rockets, for which the period of burning of the solid fuel charge is finished still on the directrixes, have a dispersion of approximately the same order as artillery missiles.

Thus, if one were to discuss causes provoking dispersion of rockets, then one should distinguish causes effective in the active section and in conditions of free flight.

The basic causes provoking dispersion of rockets in the active section are eccentricity of application of the tractive and aerodynamic forces, and also external perturbing influences forcing the rocket to turn in flight about its transverse axis.

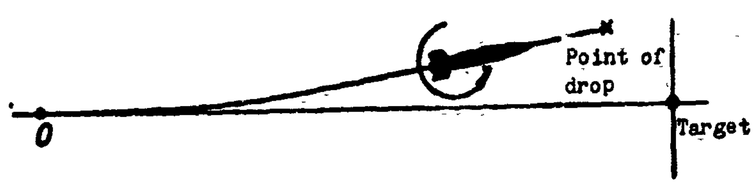


Fig. 9.63. Deflection of a rocket from direction to the target under action of an eccentrically applied thrust.

With an eccentrically applied tractive force a constantly acting moment appears, forcing the rocket to move continuously on a curvilinear trajectory and deflecting the rocket from the target (Fig. 9.63). In other words, for the active section the essential fact appears that the direction of the tractive force constantly follows after the direction of the turning axis of the rocket. If this force deflects the rocket axis from a given direction, then it turns after it itself and continues to further deflect the rocket from the direction to the target.

Aerodynamic eccentricity analogously affects the active section, inasmuch as it changes the direction of the rocket axis, and at the same time the direction of the tractive force.

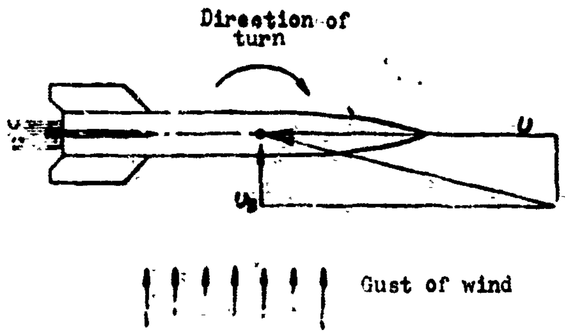


Fig. 9.64. Influence of wind on the flight of an unguided rocket.

The influence of external perturbing factors turns out to be very unique. For instance, a lateral gust of wind in the active section forces the rocket to deviate from the course not with the gust, as in the motion of an artillery missile, but against it. Fig. 9.64 shows a flying rocket which is influenced by a lateral gust of wind. The statically stable rocket will turn by resultant flow like a weather vane and further will continue flight under the action of the tractive force in a new direction. In this sense it is unprofitable to have a rocket with a large reserve of static stability. Such a rocket will sensitively react to accidental perturbations of a similar type, and dispersion will be large.

To lessen the action of eccentricity of thrust and aerodynamic forces the unguided rocket is spinned in flight, i.e., it is forced to comparatively slowly revolve about its longitudinal axis. This rotation should not be confused with that fast rotation which is induced in turbojet missiles for stabilization. Here rotation of the rocket is necessary only as a measure of averaging the moment of eccentrically applied forces.

Rotation is induced in the rocket either still on the guiding apparatus with the help of screw runners, or by means of installation of slanted stabilizers on the rocket. In certain cases turning is carried out by the reactive force of streams ensuing from little additional nozzles (Fig. 9.65).

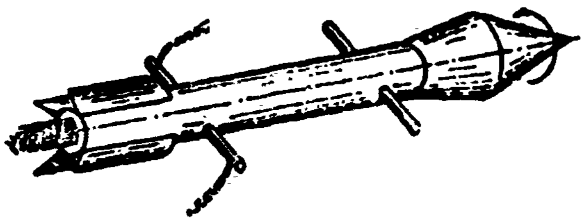


Fig. 9.65. Rotation of a rocket in the active section.

Besides the enumerated basic causes affecting dispersion of rockets, there are other causes which we partially mentioned earlier. This is technological variations in dimensions of rockets and in the size of charge, and also variations in the burning process itself, to maintain the stability of which from rocket to rocket is sufficiently difficult.

In the section of free flight dispersion is determined basically by variation of aerodynamic properties from rocket to rocket, especially eccentricity of aerodynamic forces. In the presence of aerodynamic eccentricity the rocket flies with a certain angle of incidence. Here a transverse force appears, deflecting the rocket aside.

From a mathematical point of view the phenomenon of dispersion can be considered as a result of changes induced by different causes in the parameters of the trajectory.

For instance, in a simple case for a rocket or artillery missile launched at an angle to the horizon, in the absence of resisting forces, the range is

$$L=\frac{v_0^2}{g}\sin 2\theta_0.$$

In this case dispersion by distance can be considered the result of deflections Δv_0 and $\Delta\theta_0$ in magnitudes v_0 and θ_0. Considering the change of magnitudes v_0 and θ_0 small, we get

$$\Delta L = \frac{\partial L}{\partial v_0}\Delta v_0 + \frac{\partial L}{\partial \theta_0}\Delta\theta_0$$

or

$$\Delta L = \frac{2v_0}{g}(\sin 2\theta_0 \Delta v_0 + v_0 \cos 2\theta_0 \Delta\theta_0).$$

From the last expression it is clear that deflection in magnitudes v_0 and θ_0 affect dispersion by distance in different ways, depending upon speed v_0 and angle of departure θ_0. In particular, if firing is conducted under conditions of maximum distance, i.e., at $\theta_0 = 45^\circ$, dispersion by distance is determined only by magnitude Δv_0.

Relative dispersion by distance for the considered example is

$$\frac{\Delta L}{L} = \frac{2}{v_0}(\Delta v_0 + v_0 \operatorname{ctg} 2\theta_0 \Delta\theta_0).$$

Magnitude $\Delta L/L$ has a minimum at $\theta_0 = 15^\circ$ and turns into infinity at $\theta_0 = 0$ and $\theta_0 = 90^\circ$, inasmuch as here distance L becomes zero.

A similar kind of analysis of dispersion by variation of parameters may be performed for rockets taking into account resisting forces of air and certain other factors. In this way we can clarify not only the qualitative, but in certain cases the quantitative side of the question and approximately estimate the magnitude of dispersion of the designed rockets.

Dispersion of Controlled Ballistic Rockets

As it was shown above, dispersion by distance of controlled ballistic long-range rockets is connected mainly with methodical errors in range control. One of the measures essentially lowering dispersion of ballistic rockets is improvement of range control methods. Essentially smaller influence on scattering by distance is rendered by scattering of impulse of after-effect at engine cut-off and other factors of dispersion in the section of free flight, as, for instance, variation of the atmosphere from standard, uncertainty of stabilization at entrance into the atmosphere, and so forth.

Lateral drift renders a basic influence on lateral dispersion. If a rocket has a certain disymmetry of streamlining, then during motion in dense layers of the atmosphere appearance of a lateral aerodynamic force is possible. Lateral force also appears if the line of force of thrust possesses an eccentricity and is not directed along the axis of the rocket.

Under action of lateral forces the rocket receives lateral displacement (Fig. 9.66). This displacement or, so to speak, lateral drift will occur without turning of the rocket axis (inasmuch as such turning is not allowed by a stabilization automaton).

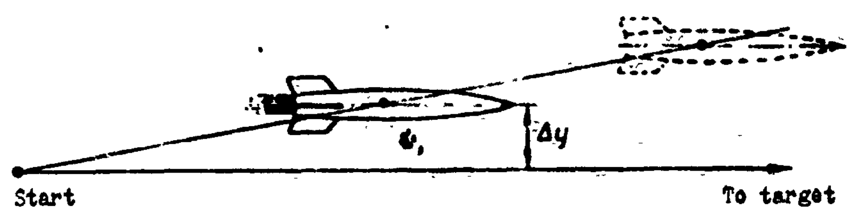


Fig. 9.66. Lateral drift of a rocket with a stabilization automaton.

Gyro instruments react only to angular displacements of the rocket, and lateral drift is not counteracted by them. As a result, a noticeable lateral dispersion appears.

To eliminate lateral drift is possible only with the aid of special devices.

One of the possible ways is creation of an autonomous integrator of lateral overloads. By means of double integration of lateral overloads it is possible to obtain the magnitude of lateral drift and to pass corresponding signals to rudders I and III.

Another method of elimination of lateral drift is application of a system of lateral radio-correction which was discussed above.

Ch. X. Ground equipment, rocket and rocket-engine tests.

L. LAUNCHING SYSTEMS.

The complex of ground equipment represents an independent and extremely important part of rocket engineering. The success of experimental launchings and of the practical use of rockets depends on the quality of the ground servicing, the education of the personnel, and the organizational set-up.

The ground complex includes all the equipment and organization associated with the launching of a rocket and with the subsequent observation of its flight and operation.

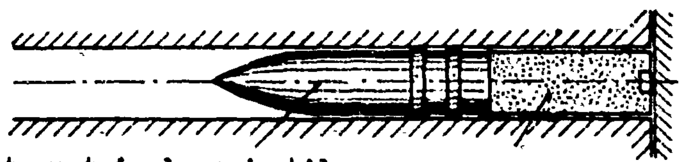
In practice, launching problems are solved in different ways according to the purpose and construction of the rocket. For large long-range antiaircraft and ballistic rockets, the organization of the launching represents a complicated problem which must be considered separately. For small powder fuel rockets, these problems are much simpler and are no more complicated than, for instance, the firing of artillery. Here questions of organization are of secondary importance, and the success or failure or launchings will be determined basically only by the technical side of the matter, i.e., by the correctness of the design adopted and by the degree of perfection of the actual construction of the rocket.

In this connection, the concept of <u>ground complex</u> as a complicated organizational-technical organism applies only to long-range remote-controlled antiaircraft and ballistic rockets. For powder rockets sent from the ground, aircraft, or ships, the term <u>launching system</u> is used. According to their principle of operation, launching systems are divided into active reaction systems, dynamic reaction systems, and rocket systems.

By an active reaction system, one should understand a gun used to shoot a rocket-sustained projectile (see Chapter II, Fig. 2.14). The projectile acquires its initial velocity from a charge of powder in the gun (Fig. 10.1a). Additional momentum is imparted to it through the burning of the rocket charge.

In the dynamic reaction system, the gases from the solid fuel blow out into a closed volume (Fig. 10.1, b). The projectile acquires its initial velocity both from the thrust of the engine and from the pressure acting on the maximum cross section of the rocket.

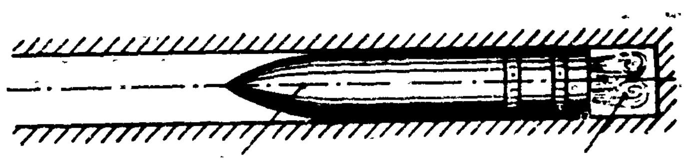
In the rocket syste, (Fig. 10.1c), the stream of gas flows out freely. The system does not receive any repulsive forces. The guide rails are acted on only by the forces of friction from the moving projectile and by the aerodynamic



Rocket-sustained projectile

Propel.ing charge

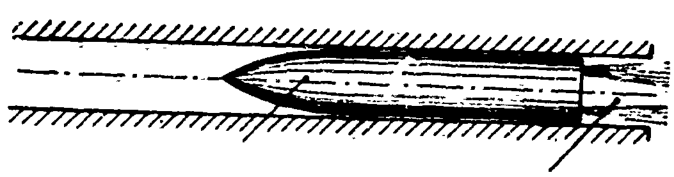
(a)



Rocket-propelled projectile

Exhaust gases

(b)



Rocket-propelled projectile

Exhaust gases

(c)

Fig. 10.1. The division of launching systems according to their principle of operation.

a- active reaction system b- dynamic reaction system c- rocket system.

forces of the jet stream. Systems of this type are the lightest and simplest, and at the present time they are used most extensively.

The launching systems can be for ground use, shipborne use, and airborne use.

Ground launching systems are divided into stationary types, mechanized types, connecting systems, armored systems, tank systems, railroad systems, portable manual systems, etc.

The most characteristic ground launching systems for military purposes are the mechanized rocket barrage weapons from the Great Patriotic War (Fig. 10.2). This launching system, which is mounted on a highly maneuverable truck, consists of a guide truss. The guides are fastened to a turning frame used to set the firing azimuth. A lifting mechanism is used to obtain the required angle of elevation of the guides. The launching system is equipped with a sighting device and electric motors.

Fig. 10.2. Launching systems mounted on trucks (parade on the Red Square).

The firing takes place from the cab of the truck. Inside the cab, in front of the gunner, there is a panel on which it is possible to establish any firing order: one shot at a time; two shots at a time; three shots at a time; or a volley of shots. The rocket charge is ignited by means of an electric fuse.

As guides, girders with groves extending most of their length (Fig. 10.3) are used. The rocket slides along them by means of special pins with heads. For some projectiles, the guides have a spiral shape, so as to impart to the rocket a rotational motion relative to its longitudinal axis.

Figs. 10.4 and 10.5 show rocket-launching systems made of tubular guides on an armored carrier and on a tank.

A mortar which is transported by a truck is shown in Fig. 10.6. The turbo-jet mine thrown by this mortar was shown in Fig. 2.2.

For aimed shooting with powder-fuel rockets, as in shooting at tanks, guides in the form of a long portable tube are used.

Fig. 10.3. A launching system with girder guides mounted on an armored carrier.

Fig. 10.4. A launching system with [illegible] guides mounted on an armored carrier.

Fig. 10.5. Launching system mounted on a tank.

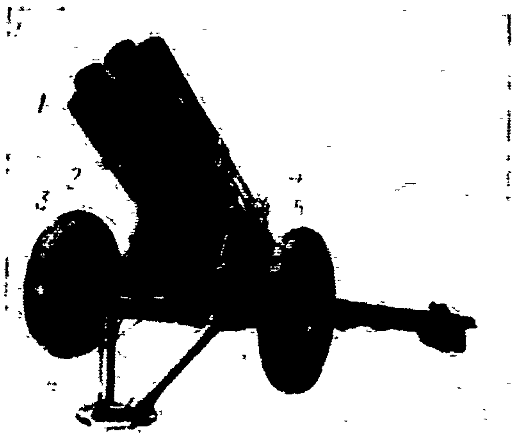


Fig. 10.6 German six-barrelled mortar.

1- tubular guides, 2- sector rack, 30 pinion, 4- sight, 5- handle for adjusting the elevation, 6- upper part of the carriage, 7- lower part of the carriage.

Fig. 10.7 illustrates shooting from an antitank rigle. The rocket is inserted in the tube from the back end. The projectile is designed so as to have the propulsion charge burn up completely while the rocket is still moving in the guides. The stream of gas emerges from the back end of the tube.

Fig. 10.7. Shooting from an antitank rocket rifle.

Ground systems for shooting unguided antiaircraft missiles differ sharply other ground systems in their speed of aiming. They have circular aiming and large angles of elevation. If relatively small projectiles are shot, then, as in shooting at ground targets, the launching systems are made of blocks of tubular guides or girder guides, and the rockets are shot in volleys. If large guided missiles are used, they are shot singly from installations similar to that in Fig. 10.8.

As in the case of a system for controlling antiaircraft gunfire, a rocket-launching system is equipped with many auxiliary devices for accurate aiming: range finders, tracking radar, computing devices for determining the necessary angles of lead, etc.

Shipborne launching systems are ordinarily intended to launch rockets from the deck of a ship for protection from air or sea attacks or to support landing operations. The distinguishing features of these launching systems are rapid aiming, which makes aimed shooting possible under conditions where the ship is rolling, and the special construction of the electric equipment and of the actual launching system in connection with the requirement that they be hermetically sealed. There exist systems with mechanized feeding of the projectiles.

Rocket launching systems on aircraft differ from ground systems and shipboard systems primarily in the absence of an aiming mechanism. The required shooting direction is given by turning the airplane itself.

The length of the guides of launching systems on aircraft is considerably smaller than that of ground systems.

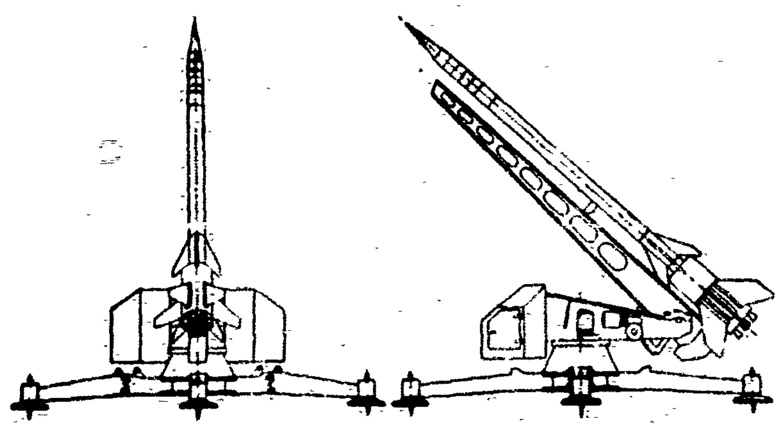


Fig. 10.8. Military installation for launching antiairc aft missiles.

Fig. 10.9 shows, at the moment of loading, two three-barrelled rocket-launching systems placed beneath the wings of an airplane.

In Fig. 10.10 two guides can be seen under the wing of an airplane, to each of which two rockets are suspended.

More recent structures are represented in figures 10.11 and 10.12. Here the powder-rocket car iers are moved out from the lower part of the fuse age of a jet airplane before shooting. The front wheel (see Fig. 10.12) does not form an obstacle in the shooting since it is drawn up into the fuselage during the shooting.

Large aerial rocket-propelled projectiles and large guided torpedoes are suspended to an airplane in the same way an ordinary bombs. However, there is one special feature connected with the protection of the airplane from the action of the blast of the torpedo's engine in starting. Either the torpedo is suspended at a ~~far~~ fairly large distance from the wings (Fig. 10.13), or it is suspended from a parallelogram-shaped bracket along ~~wit~~ which the torpedo can be moved further away from the fuselage before it is launched. It is also possible to start the engine of a torpedo some time after it has been dropped.

Fig. 10.14 shows the suspension of a guided aerial torpedo underneath the fuselage of an airplane.

Fig. 10.9. Airborne rocket-launching systems at the time of loading.

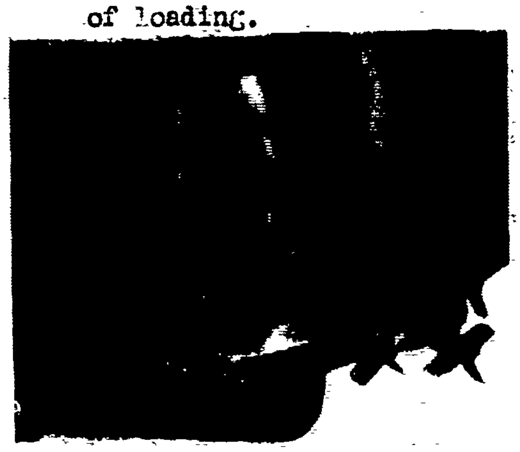


Fig. 10.10 Twin guides with rockets attached to them.

Fig. 10.11. Extensible carrier with aerial rockets.

Fig. 10.12. Extensible rocket carrier on an airplane.

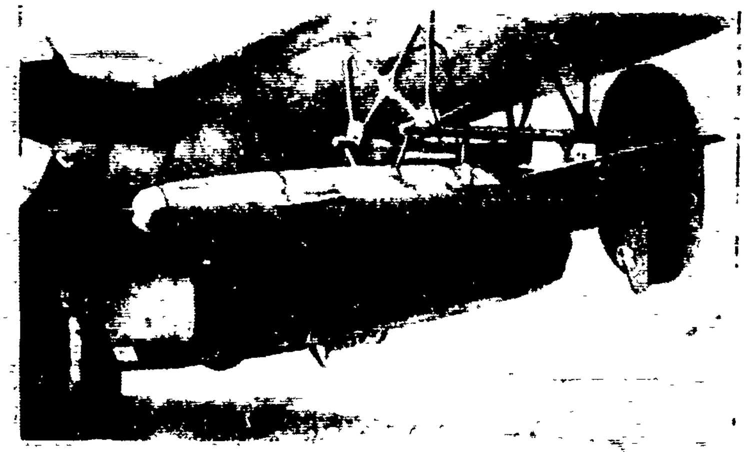


Fig. 10.13. The suspension of an aerial torpedo beneath the wing of an airplane.

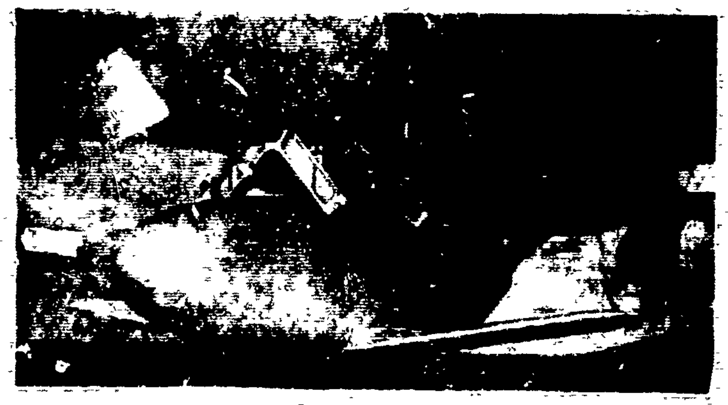


Fig. 10.14. The suspension of an aerial torpedo underneath the fuselage of an airplane.

2. GROUND EQUIPMENT.

The Ground Equipment for Long-Range Ballistic Rockets.

The organization of the launching of long-range ballistic rockets represents an infinitely more complicated problem than the launching of tactical rockets.

The complex of personnel and equipment on the ground is used to carry out the following basic types of work in a definite order:

1. Loading a rocket or its structural components into railroad cars or airplanes.

2. Transporting the rocket and its components by rail or by air.

3. Unloading the rocket or its structural components onto conveying devices operating on the ground.

4. Transporting the rocket over roads.

5. Transporting servicing parts by various means of transportation.

6. Generating electric power and charging the batteries.

7. Production and drying of compressed air.

8. Conducting preliminary tests of the rocket in a horizontal position.

9. Unloading the rocket and its structural components onto adjusting devices, and assembly of the rocket.

10. Setting up the rocket in its firing position on the launching pad.

11. Conducting autonomous tests (tests of individual units) and [illegible]

complex tests of the automatic parts of the rocket.

12. Filling the rocket with the fuel components and with compressed air.

13. Providing means of extinguishing possible fires.

14. Heating (where necessary) individual components of the rocket.

15. Aiming the rocket.

16. Adjusting the remote-control devices.

17. Preparing the devices for controlling the flight and checking the trajectory.

18. Starting the engine.

19. Preparing the devices for determining the landing point of the rocket.

20. Emptying the fuel from the tanks of the rocket in case the launching does not take place.

21. Removing the rocket from the launching pad, dismantling it, and loading it on to transportation devices.

The equipment of the ground complex used to solve the above problems is usually grouped at the servicing location and at the starting location.

The set of equipment at the servicing location includes two power plants; one to supply the ground equipment, the other [illegible] apparatus [illegible].

Moreover, at the servicing location there are [illegible] with [illegible] devices, machines for the transportation of cables, and instruments for horizontal tests.

After the rocket has been checked and tested at the servicing location, it is moved to the starting location. ~~[illegible]~~ Here the nose-cone is fitted on the body of the rocket, which is in a horizontal position. For this, a crane or a special device is used.

After the nose-cone has been installed, the rocket is set up in a vertical position on the launching pad.

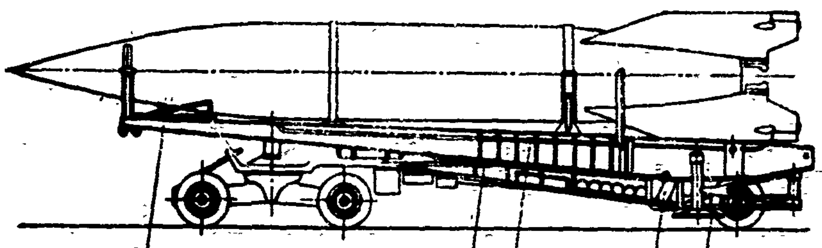


Fig. 10.15. Rocket on a carriage (long-range ballistic rocket, V-2). 1- folding ground jack, 2- hydraulic lifting jack, 3- upper folding platform, 4- carriage boom, 5- hand rails.

For driving up and setting up a V-2 rocket, a ground carriage is used (Fig. 10.15). Before ~~lifting~~ the rocket is lifted, the carriage is fixed relative to the immobile launching pad.

By means of hydraulic jacks, the boom of the carriage and the rocket can be lifted to a vertical position (Fig. 10.16). Thus, the rocket is set up on the feet of its four stabilizers ~~and~~ on the launching pad. The structure of the stabilizers and of the tail section of the rocket contains special reinforcements which take the weight of the upright rocket and transmit it to the basic lower transverse, which is joined to the tail section. To this same traverse, the thrust is imparted in the flight of the rocket.

To lift the carriage boom (4) (see Figs. 10.15 and 10.16), there are two hydraulic telescopic jacks. Since the center of ~~xx~~ gravity of the rocket is shifted backwards relative to the points of support (the wheels), the carriage contains two mechanical jacks. They can be rotated relative to the vertical axis, and thus the points of support can be moved back. The jacks in Fig. 10.16 are seen in the position ~~where~~ in which they are on the ground. In Figs. 10.17 and 10.18, the rocket is shown on the carriage before transportation and during installation on the launching pad.

To set the rocket up on the pad, different types of lifting devices can be used. Fig. 10.19 shows the installation of the American Corporal rocket on a launching pad, while Fig. 10.20 shows the installation of the Redstone rocket.

In some cases (not for military rockets), the rocket is delivered to the launching pad in separate sections, and it is assembled directly on the pad.

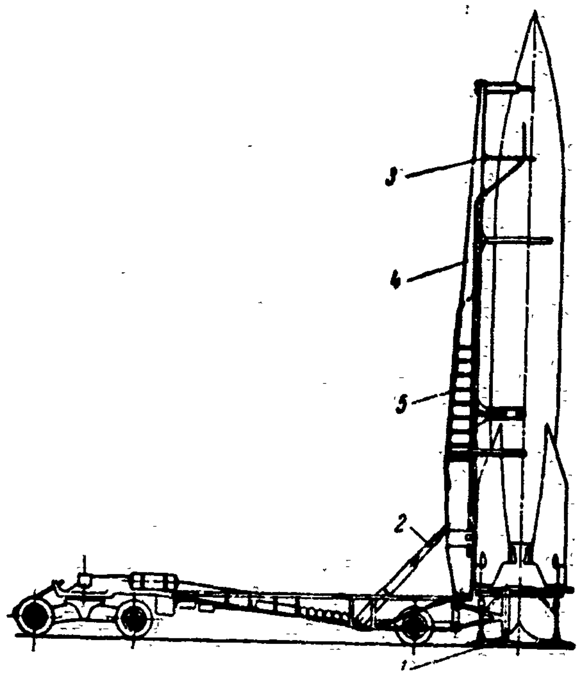


Fig. 10.16. A rocket raised to the vertical position. The location is the same as in Fig. 10.15.

Fig. 10.21 shows the transportation and Fig. 10.22 the installation of the central part of the Viking - an American meteorological rocket - on a launching pad. After the body of the rocket has been raised, the nose-cone is installed by means of a crame while a stabilizer is attached to the lower part.

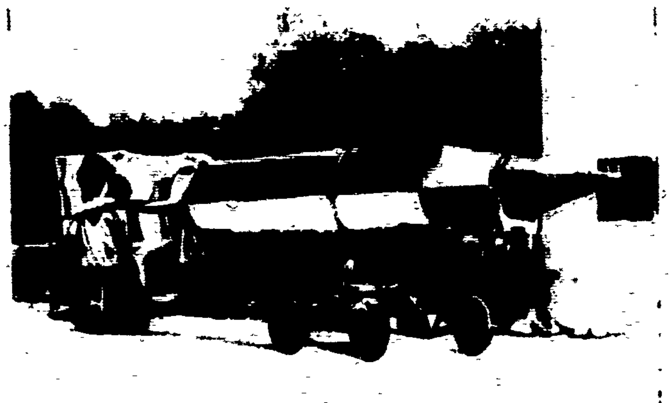


Fig. 10.17. Rocket ready for delivery to the launching pad.

Fig. 10.18. R ising the rocket together with the car iage boom.

After it has been installed on the pad, the V-2 rocket is freed from the bands tying it to the carriage boom (Fig. 10.23 shows the rocket just as these bands are being removed). Then the carriage is moved a little way from the pad (fig. 10.24), and during the subsequent preparations for launching it is used as a servicing truss, which is used by the servicing personnel to reach the instrument section and the warhead. To make the work easy, folding platforms (clearly seen in Fig. 10.24) are lowered. One of these platforms is located rear the doors providing access to the engine, while another platform is located ~~the~~ at the height of the instrument section.

To service the rocket during the prelaunching preparations, it is possible to use other equipment, such as that shwn in Figs. 10.25 and 10.26.

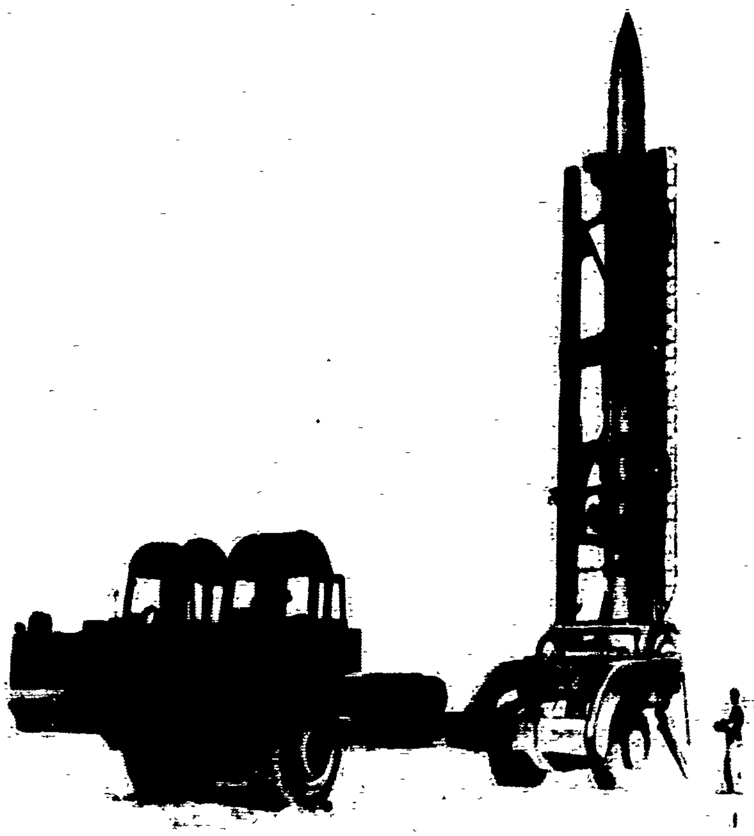


Fig. 10.19. The installation of the Corporal rocket on a launching pad.

Fig. 10.27 shows the V-2 rocket on the launching pad. To the pad a mast is attached to support the cables which connect the electric apparatus on board the rocket with the electric equipment on the ground during the prelaunching tests. After the launching, the contact-breaking plug is pulled out and the cables hang from the mast. At the same time, the actual mast is folded aside through approximately 30°.

Fig. 10.28 shows a diagram of the launching pad.

The upper part of the pad consists of two rings 1 and 2. The lower ring rests on four jacks. On top of the lower ring, there is a groove filled with ball bearings, on which the upper ring rests. Thus, both rings form a structure similar to a ball thrust bearing. The rocket is set up on the support caps, 11, of the upper ring. The upper ring and the rocket installed on it can easily be rotated through the required angle when the plane of the rocket I-III is brought into coincidence with the plane of firing. The ring is rotated by means of a ratchet and pawl 4, which is connected with a chain 10.

Beneath the launching pad, there is a jet reflector. In launching, the jet stream of gas from the engine hits the reflector and flows aside beneath the pad.

After the rocket has been set up, the jacks of the launching pad are used to adjust the axis of the rocket to a vertical position. This is checked in two planes by optical instruments. Then the upper part of the launching pad is rotated to bring the I-III plane of the rocket into coincidence with the firing plane.

In the vertical position, the rocket is first subjected to autonomous tests to check the pneumatic system and the guiding system. The operation of the starting switch

Fig. 10.20. Installation of the Redstone Rocket on a launching pad.

and of the circuit-breaking plug is verified. Then complex tests are conducted to imitate the operation of the automatic systems of the rocket at launching time and in flight.

To conduct the vertical tests, it is necessary to have a compression station which supplies compressed air to a power plant, to supply the ground equipment and the actual rocket, and finally a truck to transport cables, apparatus and space instruments and devices.

Fig. 10.21. Transportation of the body of the Viking rocket to the launching area.

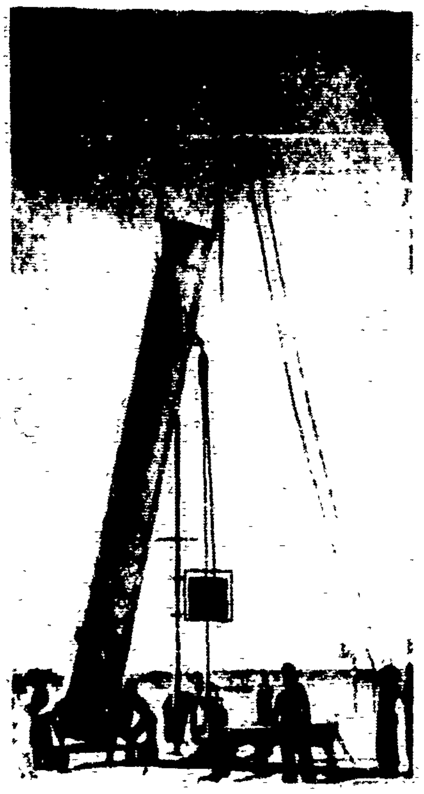


Fig. 10.22. Installation of the body of the Viking rocket (without the nose or the stabilizers) on the launching pad.

The spare instruments and devices also includes a set of spare apparatus. If any of the apparatus fails in the tests, it can be replaced. This truck also carries fuses and lead rudders in a special hermetic packing.

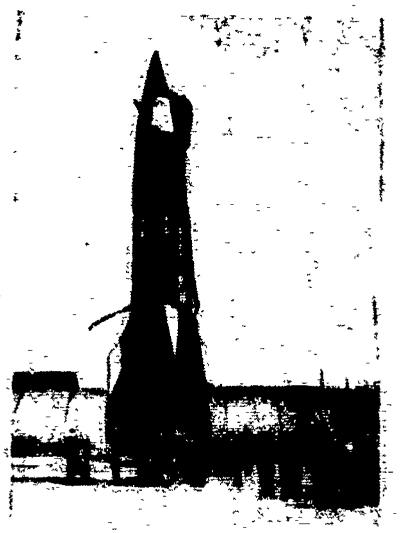


Fig. 10.23. Setting up the rocket on the launching pad and releasing the bands.

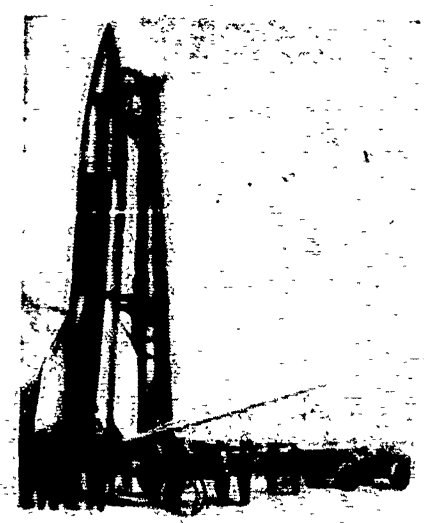


Fig. 5.24. The rocket set up on the launching pad.

After complex tests the rocket is filled with fuel. The fuel components in tank ars are fed to the rocket directly from the position of the tank cars on the railroad tracks. The filling equipment includes a tank with pumping stations, a heater for the hydrogen ~~prixpartoxide~~ peroxide, a machine for washing, a fire truck, a system for supplying oxygen, a mobile aerial battery, etc.

Fig. 10.29 shows a rocket being loaded with liquid oxygen. The oxygen is fed through an elastic tube which is covered with a layer of white frost, as can be seen from the photograph.

Oxygen which vaporized in the tank emerges though on extension of the drainage tube.

A general view of the starting area at the time of loading the rocket with fuel is shown in Fig. 10.30.

Immediately before the launching, all the auxilliary apparatus is removed from the launching area except for armored control machines. In the launching of a V-2, a machine of this type is at a distance of 150 m from the pad. From this machine, the command for launching is given.

The sequence of the work of the automatic equipment of the rocket in launching is described in Chapter IX.

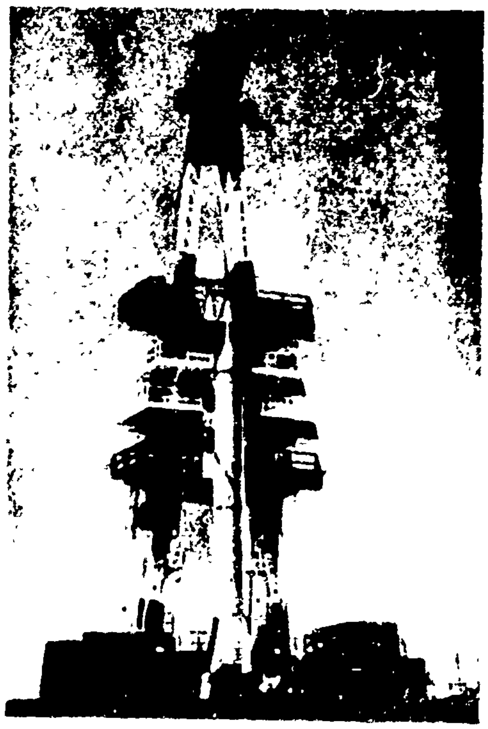


Fig. 10.25. Servicing of a [illegible]listic rocket at the starting area.

Launching a long-range rocket from the ground is not the only possible method. During the Second World War, there were also attempts to launch a rocket directly from a railroad flatcar.

Fig. 10.31 shows a railroad flatcar with a hoisting boom and a transportable launching pad. The pad is set up on the rails as can be seen in Fig. 10.32. A rocket ready for launching is shown in fig. 10.33.

Ground Equipment for Antiaircraft Guided Rockets

The requirements for ground equipment for antiaircraft guided rockets are determined by the tasks in antiaircraft defense which this type of weapon must fulfill.

The modern system of antiaircraft defense of large objects usually involves a circular belt of launching stations. The distance between stations ~~are~~ is selected so as to ~~have~~ form an unbroken defense line about the rocket, given the existing range of action of antiaircraft rockets.

The equip of launching stations must be adequate to prepare rockets for launching rapidly and to launch a large number of rockets at any time. From this it follows that it is necessary to keep a large number of loaded and completely prepared rockets at a station for a fairly long time.

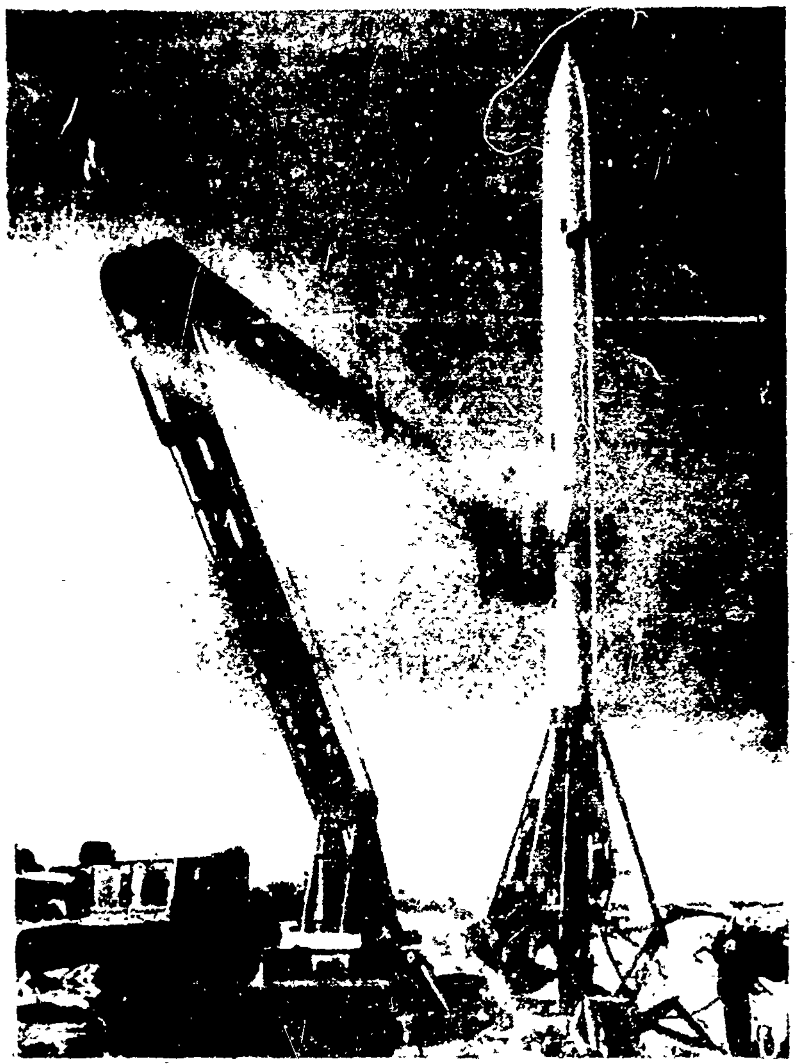


Fig. 10.26. Mechanism for servicing a ballistic rocket.

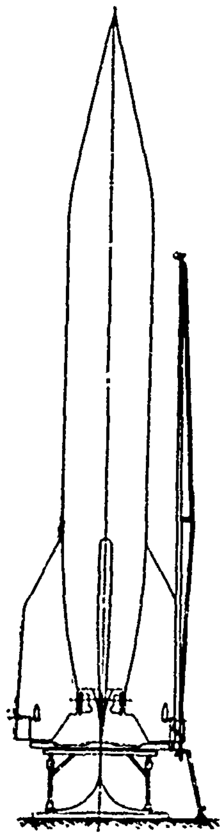


Fig. 10.27 The V-2 rocket on the launching pad.

As distinguished from the take-off area of ballistic rockets, the characteristic feature of the take-off area of antiaircraft rockets is its stationary nature.

The takeoff area of anti-aircraft contains several launching devices and means of delivering rockets to them, as shown for instance in fig. 10.34. Besides arranging the launching devices in a radial position, it is possible to place them in a row.

Fig. 10.35 shows a battery of antiaircraft rockets arranged in a row. One of the means of delivering and suspending a missile to a launchigg device is shown in Fig. 10.36.

Upon arrival from the plant, the missiles are taken to the servicing area. Here they are unloaded, unpacked, assembled and then loaded onto transportation devices.

The rocket tests (single and complex) are conducted in a horizontal position on a special testing stand. If the rocket is in perfect order, either it is sent to storage, or it is filled with air, propellant and oxidizer, and equipped with a warhead. For filling it there are special stationary filling stations with washing and neutralizing facilities. In addition to the stationary filling stations, there are transportable duplicating filling devices and air compressors.

Once a rocket has been filled with oxidizer (acid), it can be stored only for a limited period of time.

At the talke-off site, there must be a warehouse of military parts, a power plant, a boiler heating installation, garages, maintenance shops, laboratories and personnel facilities.

Antiaircraft missiles can be launched from both vertical and inclined guides. The process of launching of an antiaircraft guided missile has no preliminary stage - it is said to be ~~of the~~ a canon-type launching. In some designs, the missile is fastened to the launching device with special bolts. These fix its position before take-off, but are broken at the time of take-off. The control of the rockets is centralized, being carried out from a special bunker.

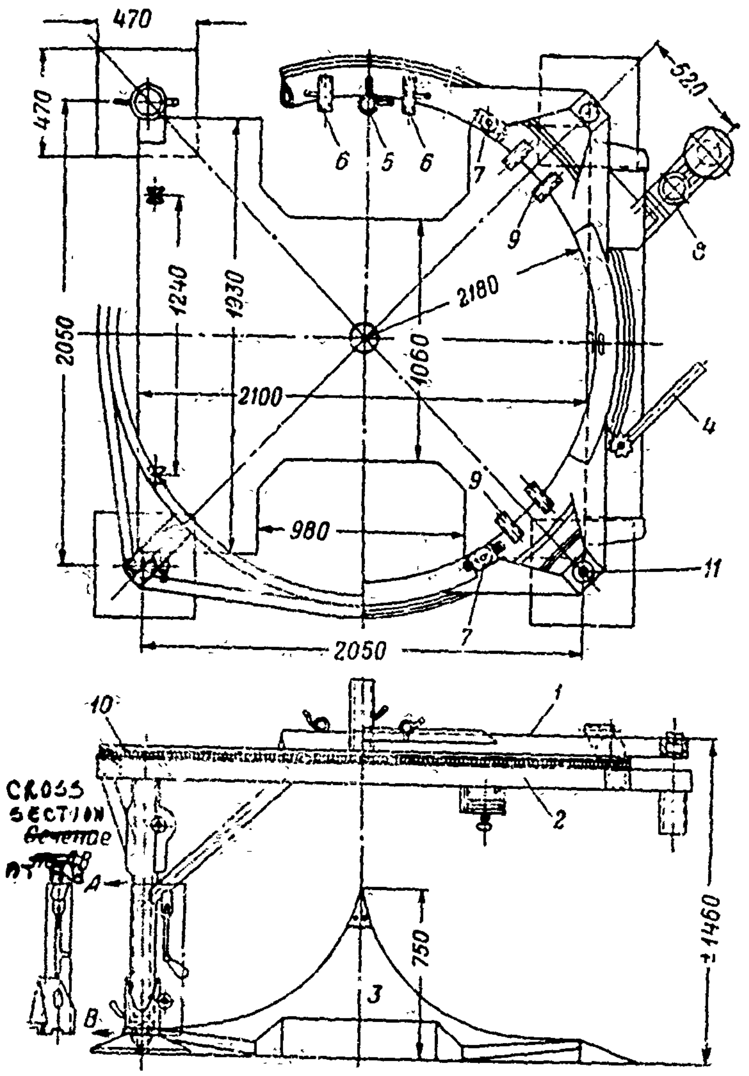


Fig. 10.28. Launching ~~and~~ pad.

1- upper ring, 2- lower ring with four jacks, 3- jet reflector, 4- ratchet gears for rotating the upper ring, 5- tube connecting with the pneumatic control board, 6- fastening devices for the block with five connecting pipes, 7- stopping clamp, 8- support for the mast, 9- fastening tubes for the fin supports, 10- chain, 11- support caps.

Fig. 10.29 Loading a rocket with liquid oxygen.

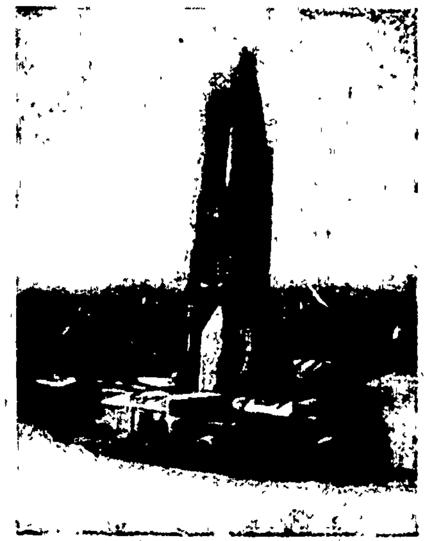


Fig. 10.30 General appearance of the take-off area when rocket is filled.

Fig. 10.31. Railroad flat car (carriage) with a hoisting jib and a take-off pad.

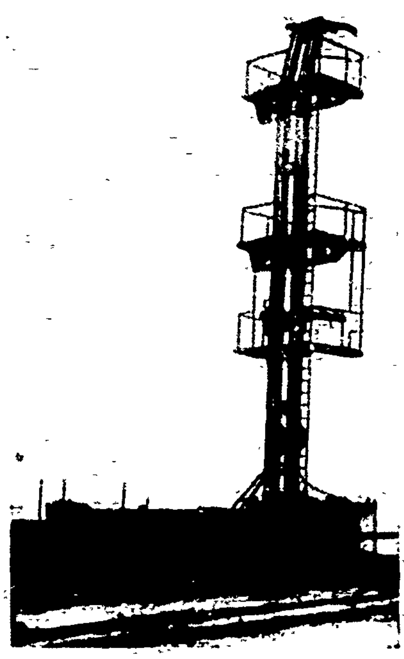


Fig. 10.32. Railroad carriage with erect jib.

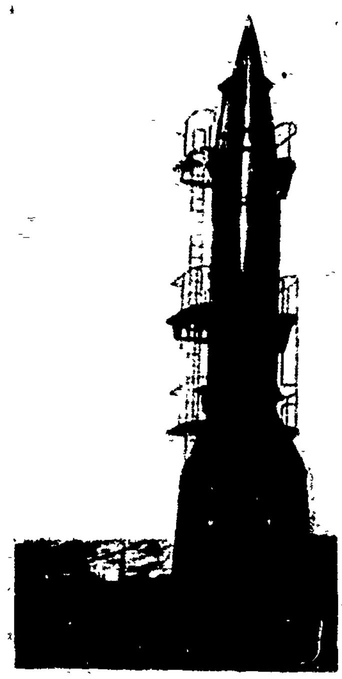


Fig. 10.33. The position of a rocket on a railroad bed.

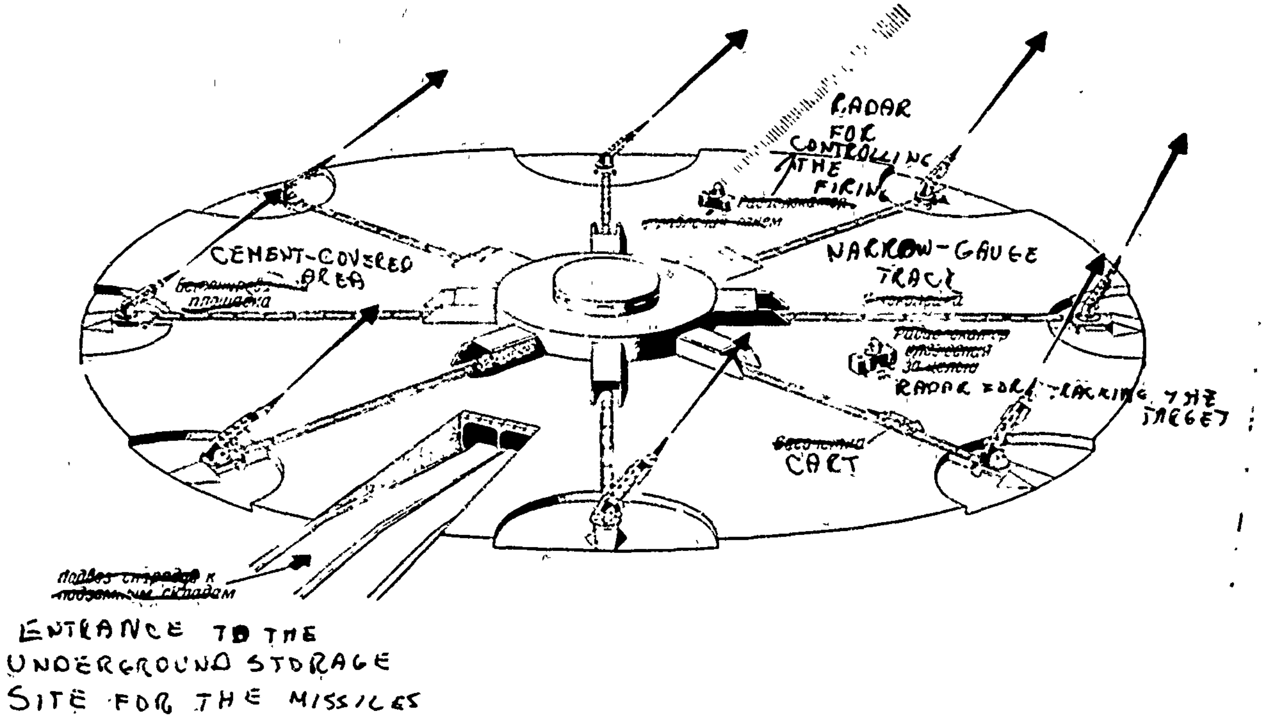


Fig. 10.34. Take-off area of antiaircraft guided missiles

Fig. 10.35. Battery of Nika anti-aircraft missiles.

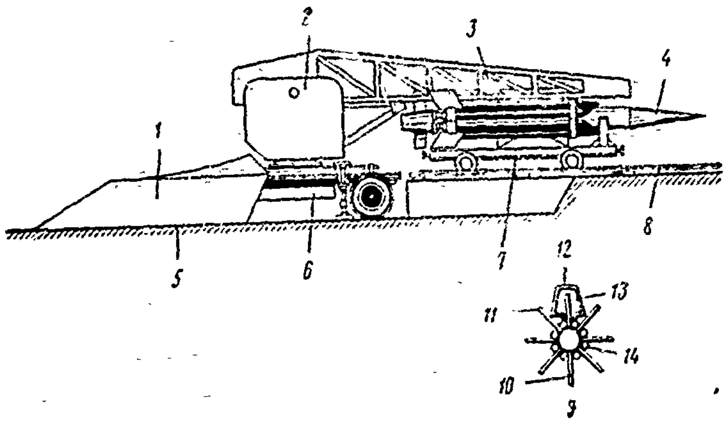


Fig. 10.36. Suspension of a missile to a launching device. 1- Flame reflector, 2- rotating mount, 3- boosters, 4- missile, 5- cemented area, 6- movable take-off platform, 7- wagon, 8- narrow-gauge tracks, 9- cross section of the missile and of the guides.

3. Testing of Rocket Engines and Rockets

Testing of Engines

Every rocket, especially long-range rockets, during adjustment passes through an extensive program of tests both of separate elements, and of the whole. A ballistic long-range rocket is a too complicated and responsible apparatus to build it immediately without thorough experimental tests of units and assemblies.

The single most responsible task is testing and completion of the engine.

Depending upon problems, set before the test it is possible to note the following types of tests of rocket engines, disposing them in order of increasing complexity.

1) technological and control-delivery tests of serial engines;

2) completed tests of modernized serial engines or new engines;

3) structural and problematic research tests, necessary for creation of new and better rocket engines.

Technological and control-delivery tests of serial rocket engines and their elements are very extensive. These tests are necessary for control of dimensions of prepared elements, quality of their materials (especially after welding and heat treatment), and also durability and airtightness of elements and joints.

Such types of tests of rocket engines are very important, since there are a large number of welded, pressed, threading joints and joints, made by expansion and rolling, which should be not only durable, but also tight, i.e., not to allow significant pressures of component of leakage through them from the main lines of the engine or mixing in unassumed places inside the engine.

In all cases tests are conducted first on durability, and then on airtightness.

Durability of joints and details is checked by muffling by special plugs all holes connecting the internal cavities with the atmosphere and creation in these cavities of a pressure exceeding the operating pressure by definite norms of durability magnitudes. The necessary pressure is created by some liquid passed into the cavities of the elements.

After checking durability, tests of elements and joints for airtightness are conducted in this same way. In this case the necessary pressure in the cavities of elements is created by a liquid or air. Defects are revealed by the speed of pressure drop in the tested cavities.

To such tests are subjected many units of the engine and the supply system: tanks, pipelines, valves, the engine chamber itself and its elements — head, cooling jacket and so forth.

Tests of this type are not specific; they differ from analogous tests of units of other machines only in the form of the tested details, which requires special, sometimes very complicated attachments.

The gathered units of the engine, being the completed assemblies — chamber, pumps, valves, reductors etc. — are subjected (still before assembly into a single engine) to functional tests determining correctness of action, reliability of work, and also the conformity of their data to calculate characteristics.

Here the engine chamber is tested on a special so-called fire stand. On this stand in the engine chamber fuel is burned in conditions coinciding with conditions of work in a liquid-fuel rocket engine. Such a test is called burning out of the chamber, it reveals all earlier unrevealed defects of manufacture of the individual units.

A defective chamber, depending upon the seriousness of the revealed defects is either totally rejected or is sent for correction. After correction the chamber is burned out again.

Testing of pmps and turbines of a liquid-fuel rocket engine in principle differs little from tests of analogous machines in other branches of machine building. Practically important peculiarities of these tests are large flow rates of liquid through pumps, large numbers of turns and correspondingly high power of pumps and turbines at short duration of work, and consequently also at a shorter duration of tests.

Suitable units, passing the functional tests, are used in rocket engine assembly. Moreover, in the process of assembly a number of tests are also conducted, the goal of which is to control airtightness.

The assembled engine is subjected to a number of additional tests. In the first place it passes a functional cold (without burning in the chamber) test, in the process of which the regularity of interaction of individual assemblies of the engine is checked, and it is established how reliably its fittings works, whether they move the liquid component through inlet and cooling channels of the engine, and also through the head of the chamber sufficiently well. These tests can be conducted in specific conditions characteristic for conditions of use in the given rocket: at heightened or lowered temperatures, at lowered atmospheric pressure, at increased humidity, in conditions of great dustiness of air, etc.

Furthermore, during engine assembly tests are conducted for checking and adjustment of hydraulic resistance of the main channels and guarantee the necessary flow rates of fuel and oxidizer. Such tests of the engines are called cold runs.

Cold tests and cold runs of the engine are carried out usually not with the fuel components themselves, but by means of pumping of water through engine, to which are added substances not allowing corrosion of engine elements. The results of the tests are used for determination by means of simple calculations of the flow rates and pressures, which will take place during pumping of real components of fuel through the engines.

After the cold run the engine is subjected to fire tests on a stand. Here from the working engine the most important characteristics are revealed, which allow us to determine how much the given engine corresponds to technical conditions. Simultaneously the degree of reliability of the engine is checked. By the results of these tests a technical log book of the given engine is made.

At small-lot production of liquid-fuel rocket engines, which was characteristic for the initial stage of development of rocket technology every unit and every assembled engine was subjected to similar control-delivery tests. Due to an increase of the quantity of monotypic motors of a given series and due to improvement of technology of their manufacture, it is possible to subject not every unit and engine, but only their individual samples to such thorough control-delivery tests and especially tests on fire stands. Such tests are called <u>control-selective.</u> The quantity of tested engines or units is in definite proportion to the number of released articles.

<u>Finishing tests of the assemblies,</u> units and the whole engine complete the structural development of a new or modernized engine. Besides numerous cold and fire tests, in the process of which non-completion and defects of newly designed engines are eliminated they include tests on working of the resource. These tests allow us to determine the possible duration of work of the [illegible] possible

number of launchings at which its reliable work is guaranteed.

An important element of any tests is obtaining reliable data on the magnitude of measured engine parameters.

The number of parameters measured depends on the type of test. In control-delivery and finishing tests a minimum quantity of the most important parameters is measured: engine thrust, flow rate of fuel components, pressure in combustion chamber, pressure of fuel components in different points of fuel and oxidizer channels, certain parameters of the turbopump unit or assemblies of the displacive supply system.

However, in the finishing tests there frequently appears the need to measure many additional parameters for clarification of the cause of any malfunctions revealed during the tests.

In scientific research tests the number of parameters measured increases very significantly. Here it becomes necessary to measure the temperature, determine the chemical composition of combustion products, measure the speeds of gases, liquids, measure the heat flow, measure voltage and deformations in engine details, etc. One should note here that measurement of such magnitudes as, for instance, speed and temperature of combustion products in different points of the chamber, is a very difficult problem, growing sometimes into an independent scientific problem. In many cases direct measurement of certain magnitudes turns out to be generally impossible and is replaced by indirect determination. Measurements during fire tests are hampered by the fact that all of them have to be conducted remotely and, furthermore, many components of basic and auxiliary fuels can be harmful or low boiling point liquids.

The most complicated and specific tests are fire tests of the chamber and rocket engine as a whole.

For the special fire stands are created, which are complex structures, while stands, intended for tests on big liquid-fuel rocket engines have, furthermore, very impressive dimensions.

For firing tests of liquid-fuel rocket engines, at the stand there should be set up large storehouses for storing the fuel components; there should also be flow tanks from which the components are fed to the test engine or chamber.

During erection of the fire stand it is necessary to consider regular discharge from the stand and engine of the stream of fuel combustion products. The stream of gases has a high temperature and high speed and has strong erosional influence on the ground, various buildings and structural materials. Therefore, combustion products are discharged in such a way so that the stream meets the ground at significant distance from the nozzle end.

For this reason small chambers and engines are tested in a horizontal or slightly slanted position (Fig. 10.37); in this case

Fig. 10.37. Stand tests of liquid-fuel rocket engines in a small fire box.

the stream either in general does not touch the ground, or meets it very weakened. It is expedient to test large engines in a vertical position or, in any case, with location of their axis with a greater angle to the horizon. In such case fire stands are conveniently built on the edge of natural or artificial mountain precipices or

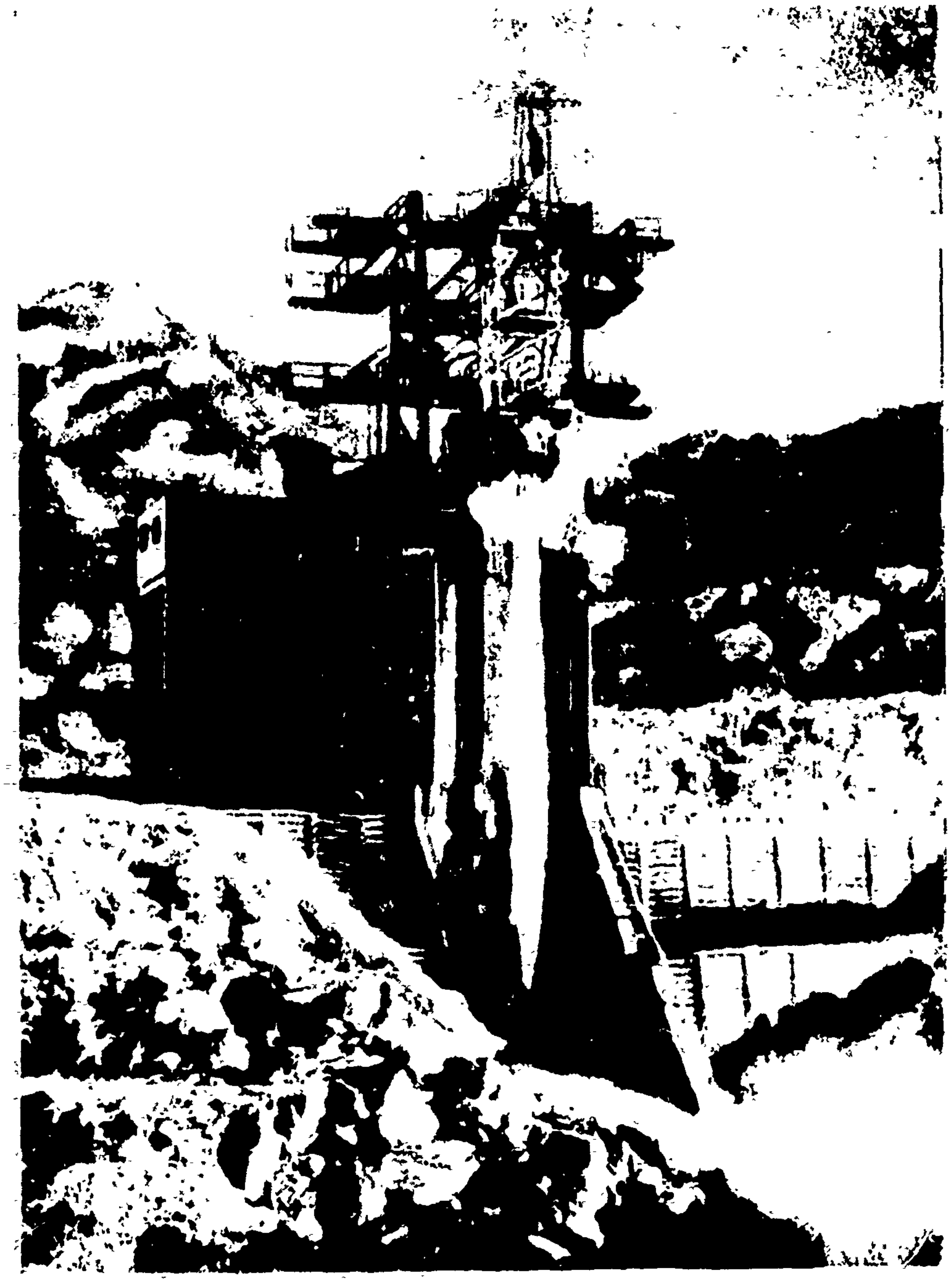


Fig. 10.38. Stand tests of a large rocket engine.

ravines (Figs. 10.38 and 10.39). In case of absence of suitable slopes and ravines it is necessary to erect a high and sufficiently durable (usually concrete) tower which is the base of the stand (Fig. 10.40).

On the stand itself the engine is secured on a console site having a sufficiently great overhang, in order to reliably protect the wall of the stand and its foundation from the influence of the passing stream.

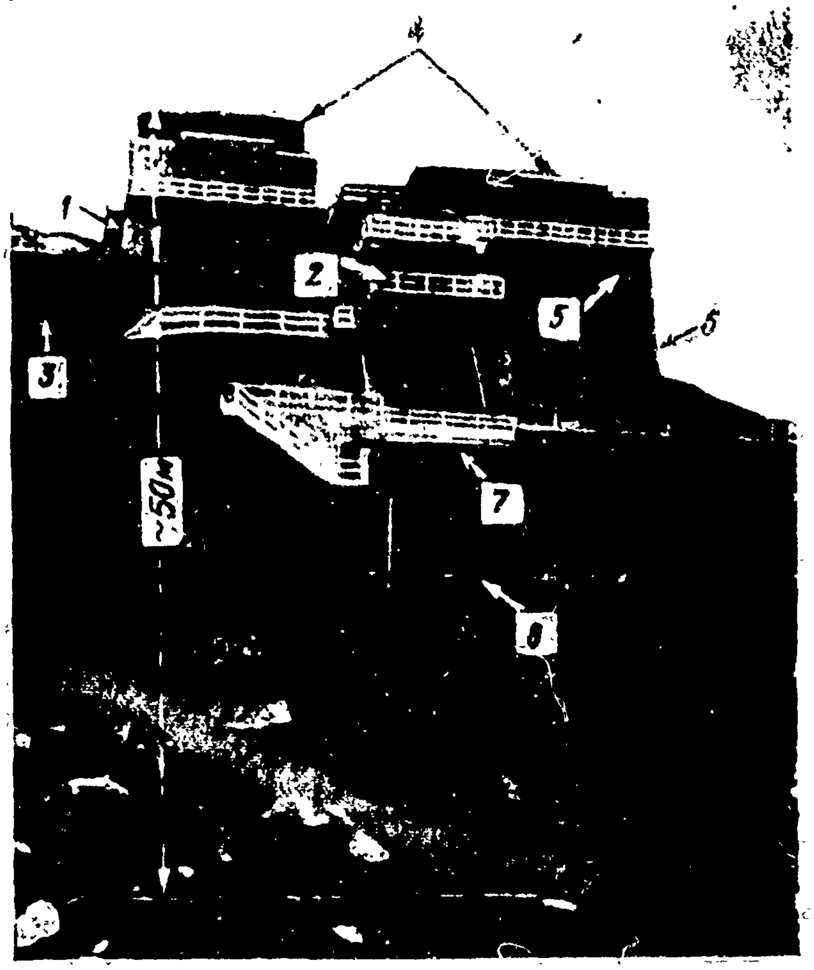


Fig. 10.39. Stand for testing large liquid-fuel rocket engines. 1) pump section, 2) frame for liquid-fuel rocket engine, 3) observation window, 4) fuel reservoirs of great capacity, 5) pump section, 6) fuel pipeline, 7) draw bridge, 8) stream reflector.

The chamber or engine is secured on the stand with the help of special machine allowing measurement of the tractive force. Here thrust is detected by a dynamometer. Figure 10.41 gives a diagram of a hydraulic dynamometer [illegible] a liquid-fuel rocket engine chamber. The tractive force creates pressure in a liquid filling the internal [illegible]. The magnitude of this pressure is [illegible] the tractive force; it is measured by a man[illegible] serve for determination of thrust. Before [illegible] dynamometer [illegible]

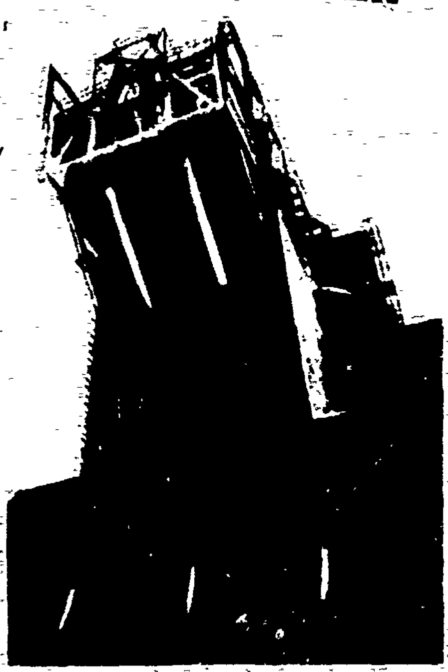


Fig. 10.40. Appearance of a stand for testing large rocket engines.

calibrated by applying loads with an exactly known weight. Error in measurement of thrust under a good system of measurements should not exceed one percent.

The fuel flow rate should be measured with the same accuracy, which is a sufficiently difficult problem. For the most part, especially on big stands, for measurement of per-second flow rate the method of volume measurement of the expended component is applied. In this case the speed of decrease in liquid level in a calibrated (carefully measured) tank is used. There exists other special methods of measurement of flow rate.

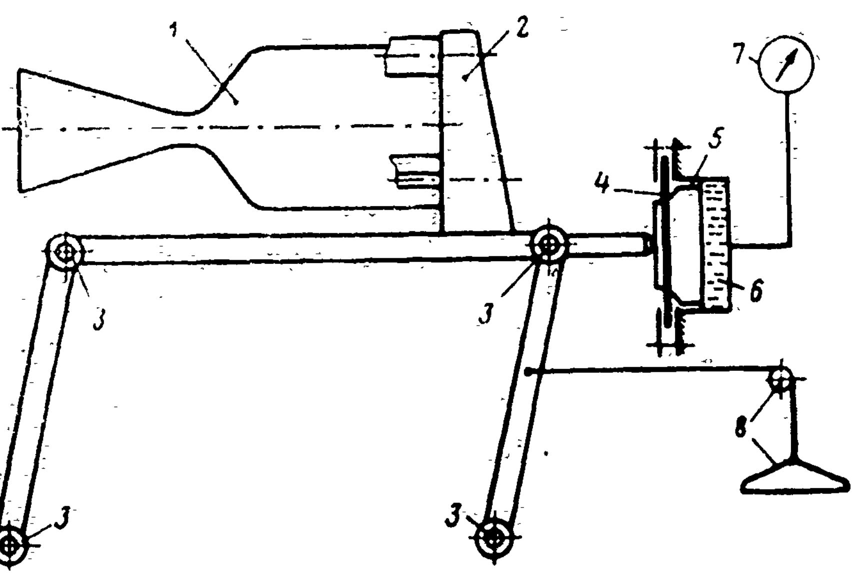


Fig. 10.41. Diagram of installation of a liquid-fuel rocket engine chamber during fire tests with measurement of tractive force. 1) liquid-fuel rocket engine chamber, 2) place of bracing of chamber to machine, 3) hinges of machine, 4) membrane of hydraulic dynamometer, 5) piston, compressing liquid, 6) liquid filling hydraulic dynamometer, 7) manometer, 8) calibration attachment.

Besides systems of measurement of thrust and component flow rates there are instruments for determination of many physical quantities on the fire stands.

In simple cases pressure is measured by common manometers. However, inasmuch as measurements are produced remotely, it is more conveninet to use low-inertia electrical methods with application of capacity or induction transducers — thin diaphragms bending under action of pressure and thereby changing capacitance or induced resistance in the circuit.

Temperatures less than 1000°C are measured usually by thermocouples or resistance thermometers.

Measurement of other physical quantities requires special methods, an account of which goes beyond the limits of this book.

Indication of instruments measuring some parameters by electrical methods are recorded usually by oscillographs. Indications of other instruments through definite intervals of time (5-10 sec.) are either recorded by special observers, or fixed by means of photographing instrument scales on a panel.

Control of engine or chamber function during fire tests is practically reduced to control of ignition and the fuel feed system.

For control of engines and recording of indications of the instruments a special well protected location on the stand is used, from which it is possible to watch the working engine. In this location there are control panels and measuring panels, where numerous measuring, recording, and control instruments are placed, by which it is possible to judge the work of the most important units of the stand and engine. On the control panel there are switches with which starting or turning off of the engine is conducted.

In the best stands the assumed program of tests is maintained automatically. For that any time mechanism (for instance, in the simplest case — a turning drum) consecutively activates the mechanisms of the stand and its automatic equipment pretuned to definite conditions. In this case to carry out tests it is necessary to pass only one signal to the operator.

In less perfect stands the necessary operating conditions of the engine are established after starting of the engine by indication of the instruments. In this case for realization of the testing program several men should be employed.

Fire stands with all their devices are placed, as a rule, in regions remote from populated places. Although in recent times rocket engines (if one were to talk of serial samples) have become very reliable, engine explosions nevertheless can take place. It is natural that the danger of explosions increases in the case of completion steps and especially in scientific research tests. Here the most frequent explosions accompany tests connected with the search for new fuels.

In conclusion, certain fuel components, as it is known, are poisonous; therefore, their vapors should not reach populated places.

For guarantee of safety and preventing of harmful action of components on people working during tests, a whole complex of measures is conducted. Fire boxes in which engines are situated are either totally closed, or open only from those sides where there are no people. The area for control and measurements, located near the box, where obligatory there are a certain number of people during the test, are built very durably. During work with harmful components, during servicing, pumping or joining of pipelines maintenance personnel should use special clothes. In certain cases it is necessary to use gasmasks or special protective masks.

For washing off components spilled from the assemblies and pipelines of the stand it is necessary to have large quantities of water or other liquids, dissolving and neutralizing the dangerous component. Furthermore, for removal of components from protective clothes or from the skin it is necessary to have a sufficient quantity of washing points. All closed locations must have good ventilation.

Carrying out of fire tests requires precise order on the stand, strict observance of fixed rules of fuel component storage and their treatment, and also other measures undertaken for safety engineering and fire-fighting technology.

Stand Tests of Rockets

During creation of new rockets not only the engine installation is worked experimentally, but also the structure of the rocket as a whole. At different stages of designing and production of experimental rockets numerous tests are conducted on individual units and elements of their structure.

The most responsible elements of the structure's power circuit are prepared in a larger number of samples than is necessary for launching, and part of them is tested for durability. As a result of these tests the breaking load is experimentally determined.

As in aircraft building, creation of new rockets is accompanied by tests of rocket models in wind tunnels.

The control system is checked in detail. On special stands are tested steering machines, gyro-instruments, instruments of range control and other elements of the control system. Work of instruments in conditions of high-frequency vibrations, appearing in the rocket body during flight is specially controlled. Th[illegible] vibrati[illegible] simulated usually by special vibration tables.

After final assembly the rocket cannot be immediately given flying tests.

A specific peculiarity of adjustment of guided missiles and ballistic rockets is the absence of men on board. In this sense the problem of testing pilotless aircraft presents significantly greater difficulties than, for instance, adjustment of an aircraft which has a pilot on board. At simple unforeseen malfunctions the pilot usually has the possibility to correct the position on his own initiative, and also to observe the character of manifestation of those faults. After landing (if this is possible), the defects are eliminated, and the same aircraft can again be subjected to tests.

With pilotless flying objects the matter is significantly more complicated. For one rocket only one launching can be made. At the end of the test the ballistic rocket perishes independently of whether, the on-board devices and equipment worked correctly or not. In case of accident it is very difficult for the designers, and occasionally it is impossible to establish its cause, inasmuch as all judgements on the possible fault are in total dependence on transmitting equipment, of which the work of only some of the most important units of the rocket may be illuminated. Number of lauchings of ballistic and zenith guided rockets is very limited by their great value. Therefore, a statistical approach to results of tests is difficult.

A large quantity of linked devices and instruments on board the rocket, generates a so-called problem of reliability. In spite of thorough check and adjustment of every instrument, its every unit, contact, electron tube, and so forth, there always exists some minute probability of failure of a given element assumed to be perfect. This probability would have been possible not to be considered; however, on board the rocket there are a thousand similar units and elements, and failure of at least one of them can lead to failure of the system as a whole. Thus, the probability of failure of rocket automats presents perceptible danger. Namely therefore, during use of pilotless aircraft a complicated and prolonged system of checks is established.

For new unworked structures, tests are in great measure of a research character, and flying tests of rockets precede their stand testing.

Fire stand tests of rockets should be considered as a complex check of simultaneous work of all units with a working engine, i.e., in conditions approaching natural.

During fire test of the rocket on the stand the process of starting of the engine in a complex with a built-in supply system is investigated. Along with adjustment of starting, adjustment of the engine to a given thrust is made.

During stand tests work of the stabilization automat is checked. Vibrations of the body in places of installation of instruments are eliminated.

Along with adjustment of the rocket structure itself, during stand tests adjustment of the ground complex is made, including servicing, installation, pre-launching tests and so forth.

A stand rocket cannot be the exact copy of a rocket intended for flying tests, and in its structure more or less essential changes are introduced. The supply system is subjected to most changes.

During flight, component pressure on entrance into the feeding pump forms, as we know, from the pressure or pressure feed and inertial pressure of the liquid column. For a rocket on the stand, the pressure of the liquid column will be essentially smaller, inasmuch as inertial support is absent. Furthermore, if the pressure feed system anticipates use of impact pressure, as in a V-2 rocket, then this component is also absent on the stand. In virtue of the shown circumstances, for preservation of conditions of supply it is necessary to increase the pressure of pressure feed on the stand (pressure above the free surface of the component) with such thought so as to reproduce the law of change of inlet pressure in the turbopump unit during flight. In certain cases this leads to essential difficulties. The upper part of the carrier tank, in primarily its upper surface, is figured for a relatively low pressure of pressure feed, which has place in real conditions of flight. On the stand, inasmuch as this pressure sharply increases, it is necessary to consider st uctural changes of the tank in connection with the necessity to ensure its durability.

A stand rocket has essential changes in the cable network. These changes are caused by the programming of work of the automat in stand conditions. Inasmuch as the circuit of pressure feed is modified and control of its elements is produced from land, it is necessary to anticipate beforehand the laying of corresponding circuits on board. Furthermore, gyro-instruments on board the rocket are disconnected, and at activation of the stabilization automat during tests electric signals are sent, simulating flight perturbations. Here angles of displacement of the gas current rudder are measured remotely in conditions of a working engine. Temperatures are measured near the steering machines and in general, near units located in direct proximity to the engine.

The engine on the stand will be turned off either by a command from the instrument of range control (at corresponding variation of parameters of its adjustment), or by a command at emergency shutdown from the autonomous on-board devices, or, finally, by a command from the control panel of the stand.

The structure of the sample stand rocket as compared to the flying one also has changes connected with the necessity of fastening it to the stand.

The rocket testing stand differs from the engine testing stand in the means bracing, transportation and installation of rocket. A system of measuring adjustment and weighing of the rocket should be forseen on the stand. In everything else — in guarantee of the means of measurement, control, safety engineering and so forth — the rocket stand does not differ essentially from the engine stand.

In adjustment of new samples of rockets, stand tests give the designers very much valuable information. Visual observation of the work of the engine, inspection of the rocket after testing, analysis of results of treating numerous measurements during the stand tests along with the possibility of repeated startings in modified conditions allow introduction of needed changes in structure and reaching the desirable result at leisure.

Flying Tests of Rockets

Flying tests of distant ballistic rockets, as in general all big rockets, is a responsible and complicated task.

Flying tests have the goal to check the efficiency of construction in working conditions and to obtain additional definit characteristics necessary for final completeion of the machine.

During flying tests work of the propulsion system, supply system and control system is checked in real conditions; the degree of conformity of observed ballistic characteristics to the calculated is determined; the dynamics of free flight of the head part, the process of its stabilization or stabilization of the rocket itself if head part does not separate, are investigated.

Flying tests are the most responsible for checking of work of the control system, inasmuch as all preceding ground tests for the stabilization and automatic system of radio control do not give full confidence in sufficiently accurate imitation by these tests of flying conditions. Therefore, during the first launching of rockets special attention is allotted to control of work of the stabilization automat. For that it is necessary to measure real perturbations, which rockets experience, displacement angles of rudders in process of stabilization, and to analyze the possibility of appearance of self-imposed conditions. Simultaneously, during flying tests transmission (time of fulfillment) of basic commands of on-board automation is controlled.

Before flying tests, rockets are designed on the basis of calculating suppositional data on conditions of work of the structure in flight. During the first flying tests these data are definitized. Special value has determination of temperatures in different points of the rocket body and loads, especially for the head part during approach to the target. On the basis of obtained data the individual units are overhauled and in a number of cases are constructively changed. If the real working conditions turned out to be less dangerous than supposed, there is the possibility to lighten the structure. If in the process of flying tests unaccounted for peculiarities were revealed, negatively affecting the work of the structure, it is necessary to strengthen the corresponding joints of the rocket.

For rockets intended for first experimental starting the payload is usually replaced by an additional complement of measuring and transmitting equipment of telemetric control and ballast loads. In the body of the rocket special transmitting antennas are established.

On board the experimental rocket a large quantity of transducers is established for carrying out the whole complex of measurements by a program of tests. In connection with this, the experimental rocket as compared to a rocket of finally completed structure has a modified cable network.

Telemetric Control and Kino-Theodolitic Exposure

The basic means of measurement of controlled parameters in flight is the so-called system of telemetric control (STC). This system suppositionally consists of an on-board complex of equipment, including a system of transducers and a radio transmitting device, and a ground receiving-recording complex.

On board the rocket during experimental launchings different transducers are established. Mechanical transducers are connected with potentiometric devices, issuing corresponding voltages depending upon the magnitude of controlled parameters. Furthermore, on board the rocket there are also electrical transducers. Such are, for instance, temperature transducers, the action of which is based on change of ohmic resistance with change of temperature, transducers of deformations (stresses) working on the same principle, and so forth.

Electrical signals, proceeding from transducers are transmitted to earth through a radio channel.

Inasmuch as the number of measured on-board parameters significantly exceeds the number of radio channels of the transmitting device, in the system of telecontrol temporary separation of signals may be foreseen. On board the rocket in this case a program device is established connecting different transducers on a definite program for a short interval of time. At the ground station the dependence of change of parameters in time is recorded on tape no longer in the form of a continuous curve, but in the form of a curve marked by points. With temporary separation, the capacity of the telesystem, i.e., its ability to transmit a large quantity of information with a limited number of radio channels, sharply increases.

Results of recordings obtained with STC are thoroughly worked over. The obtained results are compared with the expected, and also with those observed during preceding launchings or stand tests. In case of deflections from the norm, during preparation before subsequent launchings into the corresponding units are introduced the necessary changes. If launching occurs with malfunction emergency, recording of STC serves as the basic material by which through means of analysis the cause of failure is uncovered.

The system of telemetric control is one of the most important areas of contemporary rocket technology. Adjustment of large ballistic and zenith rockets without STC is absolutely impossible, and flying tests without telecontrol lose all meaning.

During flying tests rocket flight is controlled also by kinotheodolitic exposure (KTE). In the zone of launching at a significant distance from each other several measuring points are established (not less than three), from which with kino-theodolites the rockets is photographed on film, the axis of the kino-theodolite (Fig. 10.42) is directed constantly towards the rocket, and every photographed frame has marks of angles by azimuth and altitude.

Fig. 10.42. Kino-theodolitic apparatus.

Photographing of rocket flight from different points is synchronized by time. Thus, on the basis of treatment of photographed frames at known angles and marks of time there is the possibility to completely establish the flight trajectory on the observed section.

The system of kino-theodolite exposure has special value for determination of the point of drop of the ballistic rocket. For this purpose points of observation are established in the region of drop. If the time of arrival and sector of observation in which rocket should appear are known, then during good visibility the rocket can be "captured" by several kino-theodolities, and then the concluding section of the trajectory and point of drop can be established.

Materials of kino-theodolitic surveys, supplementing data of telecontrol, are very important for adjustment of the rocket structure. In the absence of direct visibility, observation of the rocket on the trajectory is conducted with the help of radar.

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