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USSR MAIN ENGINES FOR HEAVY-LIFT LAUNCH VEHICLES: STATUS AND DIRECTION

by

B.I. Gubanov

Chief Designer of the Energia Heavy-Lift Vehicle,
Soviet Union

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Abstract

An overview of existing first and second stage launch vehicle propulsion in the USSR is presented. Specifications and design summaries are included for the RD-170 liquid oxygen/kerosene and the RD-0120 liquid oxygen/liquid hydrogen engines. New directions for engine development are described in the areas of reliability and safety improvements. Additionally, three-propellant rocket engine development activities are introduced.

Introduction

Development of main engines for medium-class and heavy-lift space transportation systems have been realized within programs working specific launch vehicles.

There are several well-known phases of main engine development in the Soviet Union (Fig. 1). A new phase was begun after first Zenit and Energia launch vehicle flights in 1985 and 1987. The Proton Launch vehicle program may be considered the first main engine development phase. In the Soviet space program the Proton has been used as one of the principal launch vehicles for more than 25 years since 1965.

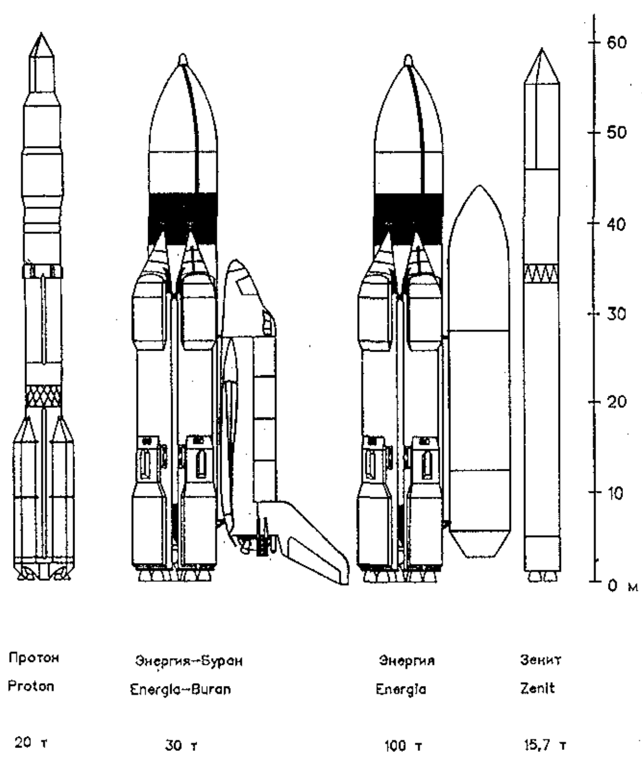


Fig 1

* Chief Designer of the Energia Heavy Lift Launch Vehicle, Soviet Union.

The two-stage Energia launch vehicle (launching mass of 2400 tons) delivers 100-tons payload to the reference orbit. Its first stage consists of four Zenit first stage modules (four strap-on boosters). The propellants are liquid oxygen and kerosene. The second stage is a single-piece (monolithic) module. The propellants are liquid oxygen and liquid hydrogen.

In order to understand the complex structure of these launch vehicles, a special explanation is required for development peculiarities. Note that different launch vehicles are related closely by a general idea. The common feature is the same main propulsion on the first stages of both vehicles.

Thus, since 1974 the Soviet Union has been developing two main engines: the oxygen-kerosene RD-170 engine and the oxygen-hydrogen RD-0120 engine. Both main engines are unique representatives of the current development phase.

The results of studies showed that liquid rocket engines used for the Proton launch vehicle and solid rocket boosters used for Space Shuttle were not suitable for advanced space transportation systems on account of their low effectiveness and ecological problems.

RD-170 Engine

The RD-170 is the most powerful liquid rocket engine in the world. It has four chambers and operates with postburning of oxidizer-rich generator gas. The engine is in successful service on first stages of the Energia and Zenit launchers.

The RD-170 engine is produced both as an expendable and reusable (up to 10 flights) unit. Its exploitation opens great possibilities for creation of permanently operating orbital stations and realization of manned flights to Mars. The RD-170 engine was developed by the Energomash Design Bureau during 1974 – 1990 under the leadership of V. P. Glushko and V. P. Radovsky.

Specifications

Thrust at sea level, tons	740
Thrust in vacuum, tons	806
Specific impulse at sea level, s	309
Specific impulse in vacuum, s	337
Combustion chamber pressure, kg/sq cm	250
Dry mass, kg	9755
Propellants:	
oxidizer	liquid oxygen
fuel	kerosene
Mixture ratio (O/F)	2.6:1
Admittable mixture ratio deviation from nominal value, %	±7
Burning time in flight, s	140-150
Thrust throttling range, %	50-100
Nozzle area ratio	36.87:1
Engine dimensions, mm:	
length	4015
diameter at the nozzle exit plane (in transport conditions)	3656
Angle of engine chambers swinging in two planes for thrust vector control, degrees	±8

Engine Structure

The RD-170 engine consists of:

four chambers:

a high-pressure turbopump (HPTP);

a low-pressure fuel turbopump (LPFTP)

a low-pressure oxidizer turbopump (LPOTP);

two preburners;

a control block for automatic devices;

a set of pressure tanks;

a hydraulic system for automatic device actuators;

a hydraulic system for control actuators;

six actuators for automatic devices;

eight control actuators;

a preburner fuel flow-rate regulator;

two oxidizer throttles;

a fuel throttle;

oxidizer and fuel starting and isolating valves;

four ampules with starting fuel;

a small starting tank;

a thrust structure;

a heat shield base;

emergency protection system sensors;

two heat exchangers for helium (oxidizer tank pressurant gas) heating

The engine uses a staged combustion power cycle, i.e., working gas is routed to the chambers after the turbine.

The chamber is an inseparable unit and manufactured with use of brazing and welding processes. It consists of a main injector, a combustion chamber and a nozzle.

The nozzle and the combustion chamber are cooled with full-flow kerosene entering the chamber main injector.

HPTP uses a single shaft and consists of an axial single-stage reaction turbine, a two-stage screw-centrifugal fuel pump and a screw-centrifugal oxidizer pump.

LPFTP consists of a high pressure screw pump and a single-stage hydraulic turbine driven with kerosene supplied after the main fuel pump.

LPOTP consists of a high pressure screw pump and a two-stage turbine driven with generator gas supplied after the HPTP turbine.

The preburner produces the oxidizer-rich generator gas used for the HPTP turbine drive. It is a brazed-welded unit and consists of an injector and a housing that are connected by a separable flange.

Pneumatic valves are actuated by high-pressure helium through solenoid valves.

Checking-technological firing tests are performed with each flight engine unit to confirm its flying life. Heavy guarantee lifetime firing test program is applied to one engine of the twelve-engine batch with following disassembly, inspection and fault detection.

Launch vehicle main engines start is accomplished through a preliminary stage with controlled thrust variation vs. time. Before shutdown the engines are switched to the tailoff thrust regime that is 50% of the nominal thrust level.

Prior to January 1, 1991 there were 804 engine firing tests with overall fire duration of 93300 s and 22 engines were flight tested during the Zenit and Energia flights.

In accordance with plans, ground lifetime firing tests of the RD-170 engine modification will be completed in 1991 and demonstrate possibility of engine multiple (up to 10) flight use.

RD-120 Engine

RD-120 is a single-chamber liquid rocket engine with a turbopump feeding system providing postburning of oxidizer-rich generator gas. The engine is intended for the Zenit launcher second stage. Propellant ignition is executed by means of the starting fuel.

A combination of experience, tradition and advanced design and technology decisions has ensured successful creation of a highly reliable engine. Full scale development and a flight test program was successfully completed. RD-120 engine modifications can be used on first stages of launchers.

Specifications

Thrust in vacuum, tons	85
Specific impulse in vacuum, s	350
Propellants:	
oxidizer	liquid oxygen
fuel	kerosene
Combustion chamber pressure, kg/sq cm	166
Dry mass, kg	1125
Engine dimensions, mm:	
length	3870
diameter	1950

RD-0120 Engine

Development of the RD-0120 oxygen-hydrogen engine proved to be a very complicated scientific and technological problem for Soviet science and industry. Engine specifications and design features (i.e., thrust, pressure levels, temperature ranges, density of main structure elements, multiple starts, labor-consuming character and technology level) exceed all previous Soviet liquid rocket engines of the same class.

The RD-0120 engine was developed with use of the experience accumulated in the Soviet Union during development of oxygen-hydrogen engines with thrust levels of 7.5 and 40 tons. In addition, results of design and scientific research carried out by the Chemical Automatics Design Bureau (RD-57 engine) and the Energomash Design Bureau (RD-135 engine) were used.

More than 100 enterprises and organizations were enlisted for investigation of numerous technical and scientific problems and engine development.

The RD-0120 (11D122) engine was developed by the Chemical Automatics Design Bureau in 1976-1990 under the leadership of chief designer A. D. Konopatov.

The engine is used on the Energia launcher second stage (central block) for delivery to an Earth orbit of the Buran reusable orbiter and other heavy spacecrafts.

Specifications

Thrust in vacuum, tons	200
Specific impulse in vacuum, s	455
Combustion chamber pressure, kg/sq cm	223
Dry mass, kg	3450
Propellants:	
oxidizer	liquid oxygen
fuel	liquid hydrogen
Mixture ratio (O/F)	6:1
Admittable mixture ratio deviation, %	±10
Burning time in flight, s	500
Thrust throttling range, %	45-100
Nozzle area ratio	85.7:1
Engine dimensions, mm:	
length	4550
nozzle exit diameter	2420
Angle of engine swinging in two planes, deg	±11

The engine provides helium heating for oxidizer tank pressurization and gaseous hydrogen supply for fuel tank pressurization and onboard power units drive.

Engine Structure

The RD-0120 engine consists of:

a chamber;

a high-pressure turbopump (HPTP);

a low-pressure fuel turbopump (LPFTP);

a low-pressure oxidizer turbopump (LPOTP);

a preburner;

pneumatic valves for engine start and shut-off;

an electrically driven regulator for thrust level control;

an electrically driven throttle for oxidizer flow control;

combustion chamber and preburner igniters;

a purging system;

emergency protection system sensors.

The engine uses a staged combustion power cycle with generator gas postburning after the turbine. The chamber is a brazed – welded unit and consists of a main injector, a combustion chamber and a nozzle. The combustion chamber and the nozzle are cooled with a certain portion of hydrogen supplied after the fuel pump.

HPTP uses a single shaft and consists of a two-stage axial turbine, a three-stage centrifugal fuel pump and two oxidizer pumps.

LPFTP consists of a two-stage turbine driven with gaseous hydrogen supplied from the combustion chamber coolant channel.

LPOTP consists of a two-stage axial pump and two hydraulic turbines providing separate drive for each pump stage and using liquid oxygen as working fluid supplied after the pump preburner stage.

The preburner provides hot fuel-rich generator gas for the HPTP turbines drive. It is a brazed-welded unit and consists of a housing, a fuel manifold and an injector.

Pneumatic valves are actuated by high-pressure helium through solenoid valves.

Checking-technological firing tests are performed with each flight engine unit before delivery. Heavy guarantee lifetime firing test program is applied to one engine of the five-engine batch with following disassembly, inspection and fault detection.

Four central block main engines start on the launch pad before the strap-on boosters main propulsion start. For the first 30 seconds of flight the RD-0120 engines operate at the maximum thrust level, then the engines are throttled down. After booster separation they operate at maximum thrust level again until the launch vehicle achieves suborbital velocity.

Prior to January 1, 1991 there were about 800 engine firing tests with overall fire duration of 165000 seconds. Two main engine sets were flight tested during the Energia flights on May 15, 1987 and November 15, 1988.

Directions in Development of Main Engines

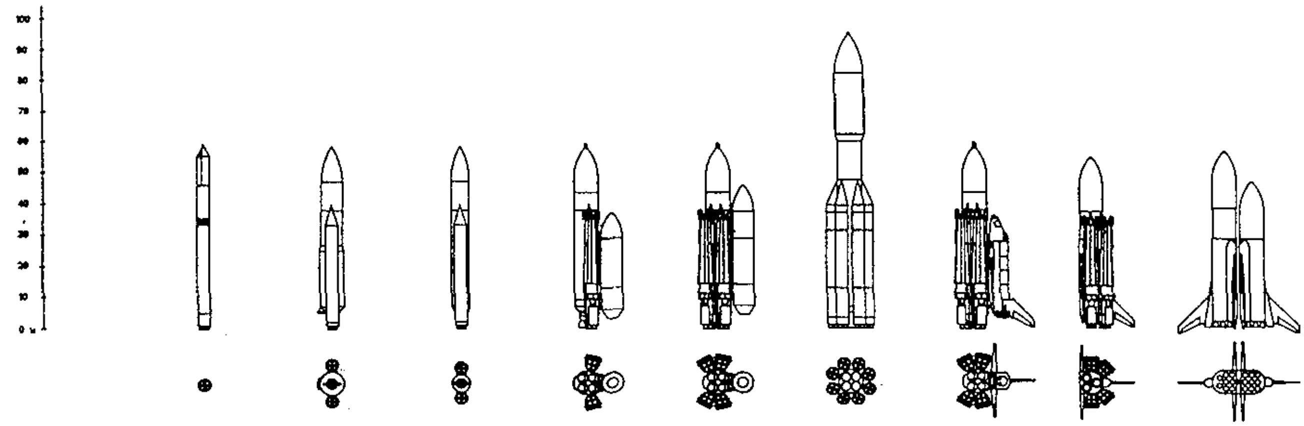
Rocket and space technology designers have to solve the problem of optimum direction forecasting and choices for further technology evolution. Achievements in the field of electronics, computerization, structural materials, and exact mechanics and tendency towards increase of payload traffic and expansion of space exploration make development of new concepts for future launch vehicles urgent.

In the USSR and the USA two trends have come out in development of space transportation systems (Fig. 2). The first one relates mainly to ballistic expendable systems descended from military warhead delivery means. Another trend relates to aircraft and winged-type reusable transportation systems with horizontal or vertical takeoff of the launch vehicle and horizontal landing of its reusable element. Both trends are equally valid according to our idea. But an economic effectiveness criterion is the same for all systems – the specific cost of payload delivery to the near-earth orbit, i.e., expenses per kilogram of payload inserted into orbit.

While the Earth gravity law exists and burned propellants are the source of power for movement, flights to the Moon, Mars and other planets will be performed with the use of expendable ballistic-type transportation systems. Also, while economic laws exist, aspiration for payload delivery to near space with the use of reusable transportation systems remains and promotes associated system development. Desire to develop such a space vehicle that lands on the Earth surface, and in several hours is ready for the next flight to space, is strong in launch vehicle designers.

Reliability and Safety Improvement

The program of propulsion systems for expendable rocket-space transport vehicle design has been, in general, specified. The program can be divided into two stages. The first stage includes permanent development of structural elements, propulsion system reliability improvement on the basis of statistical data acquisition, as well as improvement of mass characteristics. The second stage includes the change of strategic characteristics related to the propulsion system thrust augmentation (up to 230 t for RD-0120, up to 800 t for RD-170) through the change of the construction and configuration. These measures allow the use of all possible advantages of the structure and to realize the expendable launch vehicle program.



Launching mass,t	450	1060	1080	1600	2390	4800	2380	2180	2500
Payload Mass,t Low circular orbit (H=200 km)	15,7	30	35	65	100	200	30	30-50	50-70
geostationary orbit	2,8	5,4	6,3	11,7	18	36	5,4	5,4-9	9-12,6
First Stage main engine			RD-170 oxygen-kerosene engine with thrust of 806t						Dual-fuel Engine
engine quantity	1	2	2	2	4	8	4	4	
Second Stage main engine			RD-0120 oxygen-hydrogen engine with thrust of 200t						
engine quantity	1	1	1	4	4	4	4	3	

Fig 2

An emergency protection system holds a special position. The emergency protection system is a total combination of measures and control aids ensuring the propulsion system survivability and preventing dangerous development of propulsion system failures. Engine survivability is defined as unsusceptability to defects and failures of separate elements and units or an engine failure without dangerous consequences for adjacent systems of the rocket.

The results of statistical analysis have shown that failures are divided into three groups by the time of failure development from the moment when one can determine a failure by performance parameters of the propulsion system elements to the moment of the engine break-up or other consequences making the propulsion system inoperative.

Among the first group are failures for which the time of failure development, determined by performance parameters, is less than that required for carrying on protective operations. The minimum time for carrying a protective operations was estimated as a sum of the time interval for the equipment action (0.02 s) and the time for closing of valves (0.02 s). Among these failures are failures of the liquid engine turbopump for the most part. They account for 30% of all failures. Algorithms based on the analysis of performance processes are unsuitable to control such for the engine emergency switching-off, as the failure is detected in the last phase of its development. Besides, an explosion or fast combustion of structural elements occurs. The typical time of failure development on inflaming is about 0.01 – 0.001 s, i.e., even less than the time of closing of pyrotechnic valves. Therefore, to prevent dangerous consequences of such failures, it proved to be necessary to develop algorithms based on the use of information on the structural element state. The aim of creation of such algorithms is carrying-out of early diagnosis, i.e., failure detection at an early stage when the failure development process did not change into the final phase followed by an explosion or structural element combustion. For emergency protection in cases of the first group failure devices are created which use information of vibration acceleration sensors located on the flange connecting the oxidizer pump with the turbine.

The second group unites failures which development time, determined by performance parameters of the propulsion system elements, takes between 0.04 – 0.05 s. Among this group are principally failures resulting in a break-down of the oxidizer pump of the engine, e.g. gas inclusions at the pump entry, etc. Failures of the second group can be localized by algorithms using information on performance parameters (pressure, temperature, etc.). In this case, however, minimum time of the apparatus action and closing of valves is required.

The third group includes failures with typical times of several tenths of a second or more. These failures are successfully localized by the use of algorithms based on the information on the performance parameters. Among typical failures of this group is non-tightness of tanks with fuel components and gases, that can result in a loss of working medium components and a total failure.

Available means of control, diagnosis and emergency protection do not allow prevention of all types of emergency situations (particularly the ones of the first group) occurring during the liquid rocket engine operation. Specifically, the turbine and gas passage wear during launching remains uncovered, as well as cracks in the turbopump rotor appearing during long-term operation, which are the main sources of emergency situations.

Studies are under way on the use of the induction vortex flow method of controlling cracks that appear in the turbopump rotor blades of the running liquid rocket engine, as well as the method of controlling acoustical signals of the running engine to prevent the abovementioned emergency situations.

Such investigations are of particular value for the service life increase and reusability of the propulsion systems.

The possibility has appeared to control ignition in oxidizer passages by electrostatic methods through application of ignition indicators. The electric potential difference between two probes inserted into the flow and isolated from the body is being measured. At the very early stage of ignition the signal grows sharply.

A system of control and protection diagnostics is being developed on the basis of measuring the frequency spectrum density with the engine running. This system gives warning information on the imbalance of engine dynamics.

Investigations have shown that the way of widening the coverage of possible emergency situations by the emergency protection and diagnostics system is the most important and efficient one. The coverage coefficient values of 0.9 and higher predetermine the confidence when developing multi-engine rocket-space transportation systems.

For advanced reusable systems used in the first and second single-module stages a three-propellant engine takes on special significance.

Three-Propellant Rocket Engine

The engine operates in two modes: the mode of maximum thrust of 200 t in vacuum with the use of three propellant components — oxygen, hydrogen and hydrocarbon — that conforms to the rocket operation in the mode of the first stage; and the mode of maximum efficiency with the partial thrust up to 40% of the maximum value with the use of two propellant components — oxygen, hydrogen — when the second stage is running.

The thrust chamber has three-component mixing elements and one common combustion chamber in both operational modes, provides the maximum specific impulse and most favorable operating conditions for main structural elements with regard to possible thermal- and gasdynamic losses.

In this case, theoretical specific impulse values are achieved that exceed the analogous specific impulse values with separate combustion of component pairs.

To provide the maximum efficiency of such thrust chamber at ground level and further at a high altitude, it is fulfilled with a movable nozzle with "screening" cooldown.

The specific impulse amounts to 416 s in the first operating mode and to 462 s in the second operating mode.

The cooling is performed with minimum hydrogen consumption (up to 5%) in the first operating mode.

The experience shows that the hydrogen-based reducing gas leads to hydrogen "embrittlement" with crack appearance in the most stressed structural elements. In this connection the oxidizing gas has been accepted as a working medium of the turbopump.

The turbopumps are separate for each propellant component.

The oxidizing circuit with afterburning is envisaged at a suitable temperature level of the turbine working medium, that makes it possible to provide the highest pressure within the combustion chamber (up to 350 kg/sq cm) in the first operating mode.

All liquid oxygen and a portion of hydrocarbon fuel required for production of a high-temperature oxidizing gas getting to the turbopump turbine drive are fed into gas generators. The remaining portion of the fuel and all liquid hydrogen are fed directly into the combustion chambers. In the second mode the hydrocarbon fuel is used only for gasification of oxygen in the gas generators.

The component supply system includes three booster and turbopump units for each component and two single-zone gas generators.

Structurally gas generators slightly differ from each other, that results from the necessity to supply a portion of the generator gas to the hydrocarbon fuel turbine drive. The booster pumps are of a screw conveyor type.

A hypergolic ignition system is used in the gas generators and thrust chambers, the starting fuel being in ampoules.

The pneumatic system provides control of the engine automatics units. The pneumatic system includes bottles with gaseous helium.

The engine assembly includes heat exchangers for heating helium and hydrogen used in the tank pressurization system.

Engine starting is performed in the low thrust conditions.

When transferring to the second mode, kerosene is cut off, and oxygen supply into the thrust chamber is reduced, accordingly. The pressure within the thrust chamber is set in the region of 140 kg/sq cm.

The reality of three-component engine development is based on the use of:

1. The experience of creating the RD-170 superpowerful reusable engine from which the following is adopted: the basic circuit of the gas generator chamber and its mixing elements, structural materials and coatings providing the ignition protection, supply units, engine starting principles, methods of diagnostics, control and protection.

2. A powerful stand and production base.

Principal works in terms of providing a high technological level of development are carried out within research and development of RD-170 propulsion unit at present.

Only studies of the processes of combustion and three-component mixing processes are specific for a three-component engine.

Such is the state of development of powerful launch vehicle propulsion systems in the USSR as presented by a development engineer of space transportation systems.