Review Of Russia's Advanced Liquid -Propellant Rocket Engines

RD-170 FAMILY

The RD-170 liquid oxygen/kerosene rocket engine was designed in 1974-1986 at *NPO Energomash* by a team led by Chief Designer **V.P.Radovsky** for the side engine units (first stage) of the *URKTS Energiya-Buran* Heavy Multipurpose Space Rocket Transportation System. Until today, the RD-170 remains the world's most powerful and quite promising engine design.

RD-170 is a four-chamber, recoverable liquid-propellant rocket engine with the fuel turbopump-feed system. The design provides for high-pressure afterburning of the exhaust turbine gas in the main combustion chambers. The ignition of fuel components is ensured by self-igniting starting fuel.

RD-170

The engine's fuel system includes the Main Turbopump Unit (MTU) incorporating a single-stage active axial-flow turbine co-shafted with rotors of the oxidizer and the fuel pumps. The turbopump assembly also includes two screw boost pumps: the oxidizer pump driven by the gas turbine, and the fuel pump driven by the hydraulic turbine. The pneumatic control system includes: pressure helium bottles, solenoid-operated electro-pneumatic valves and system piping.

According to Mr. **Feliks Chelkis**, Chief and Lead Designer of the RD-170 rocket engine, there is hardly another national or Western liquid-propellant engine showing same performance in terms of full-thrust gaining smoothness, which makes the loads applied to the engine and the launch vehicle considerably lower.

Engine designers say development of a recoverable high-thrust power unit is a challenging task. To ensure controllable throttling to full thrust and cut-off smoothness, it is vital that consumption of propellants is adequately regulated by means of powerful electro-hydraulic devices.

The described engine went through full-scale development, bench trials and flight tests. Ground test-firing of the engine was conducted on a fully built-up unit.

As development works continued on the RD-170 engine, the designers encountered fire accidents in the engine's hot gas-flow duct resulting in that MTU assembly bearings were destroyed, in addition, there were high-frequency vibrations in the combustion chamber. The fire accidents were found to have been caused by the presence of foreign materials (like chips or fillings) in the gas ducts. Certain bottlenecks in the turbopump assembly operation had to be overcome, as well. Yet, the high-frequency vibration able to cause the injector assembly to destroy was something the designers had to address again and again till the engine was just about to be commissioned.

The combustion chamber development phase included as many as 300 bench firing trials of summarized duration of 23,000 s; a total of six different cooling system options were tested along with more than 20 injector design options for a variety of nozzle parameters. A replacement of the combustion chamber head does not require the engine to be totally disassembled. The complete RD-170 engine can be re-started at least 20 times before total engine overhaul is required.

So far, RD-170 has had a record of over 120,000 s in bench firings, showing high reliability. Program managers, however, seem to be not quite satisfied with the already attained reliability level, and have been planning development of a more reliable and more economical RD-170 design option for reduced fuel consumption.

The RD-170 engine is designed to ensure failure-free operation, i.e., provide normal functioning under any operating conditions without failures that might result in any disastrous consequences for the LV's or the power plant's relevant systems, like hydraulic drive piping destruction or seal failure.

The design of the RD-171 engine (a modification of RD-170 installed in *Zenit-2* LV's first stage) incorporates four pivoted combustion chambers, unlike the RD-170 version for the *URKTS* System, where two chambers are pivoted. Pivoting is ensured by auxiliary power units (APU) operating on hydrazine decomposition products. Each booster unit of the *URKTS* System's first stage includes one APU; similar APUs were installed on the *Buran* orbital spacecraft to provide for actuation of the aerodynamic control devices (like elevons, flaps, rudder) and landing gear. Table 1 below compares performance of RD-170 and the United States' F-1 liquid-propellant engine.

Starting from late 80s, agencies responsible for the national space program have been displaying interest in exporting RD-170 to gain much-desired convertible currency. After per-

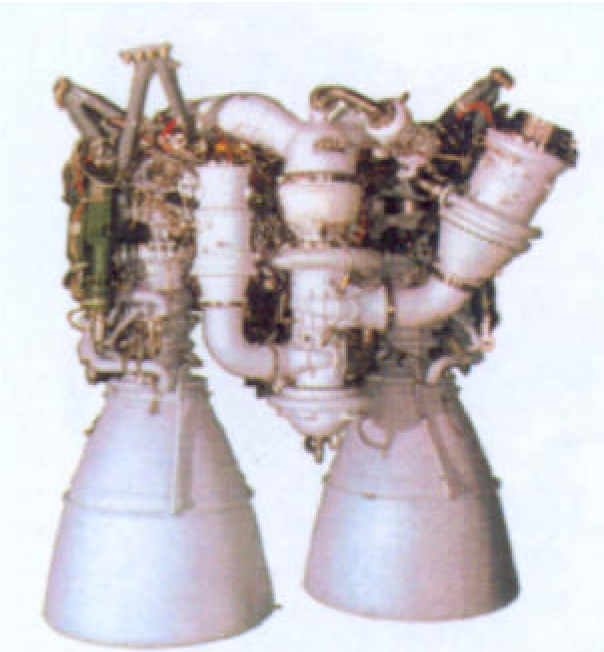
Table 1. *RD-170 vs. F-1: COMPARED PERFORMANCE*

Performance Characteristic	RD-170	F-1*
Engine configuration	Closed-loop	Open
Oxidiser	Liquid oxygen	Liquid oxygen
Fuel	Kerosene	Kerosene
Oxidiser/fuel ratio	2.6	2.2
Combustion chamber pressure, atm	250	80
Ground thrust, tf	740.2	690.1
Vacuum thrust, tf	806.7	792.7
Specific impulse on ground, s	308	265.4
Specific impulse in vacuum, s	337	304.8
Fuel consumption, t/s	2.4	2.6
Nozzle area ratio	33.5	16.1
Engine mass, t	5.2	7.9
Engine mass-to-thrust ratio, kg/tf	6.45	9.97
Height, m	4.0	5.6
Diameter, m	3.8	3.8
Turbopump assembly horsepower	220,000	60,000
Rated duration, s	140	150

**) F-1 engine was developed by Rocketdyne in 1959-1967 for Saturn-5's first stage.*

Table 2. ***RD-170M, RD-180, RD-191 ENGINES PERFORMANCE***

Performance Characteristic	RD-170M	RD-180	RD-191
Engine configuration	Closed-loop, w/afterburning		
Fuel components	Liquid oxygen/kerosene		
Ground thrust, tf	780.5	375.1	188
Vacuum thrust, tf	807.4	408.2	204.6
Combustion chambers	4	2	1
Height, m	4.0	4.0	3.9



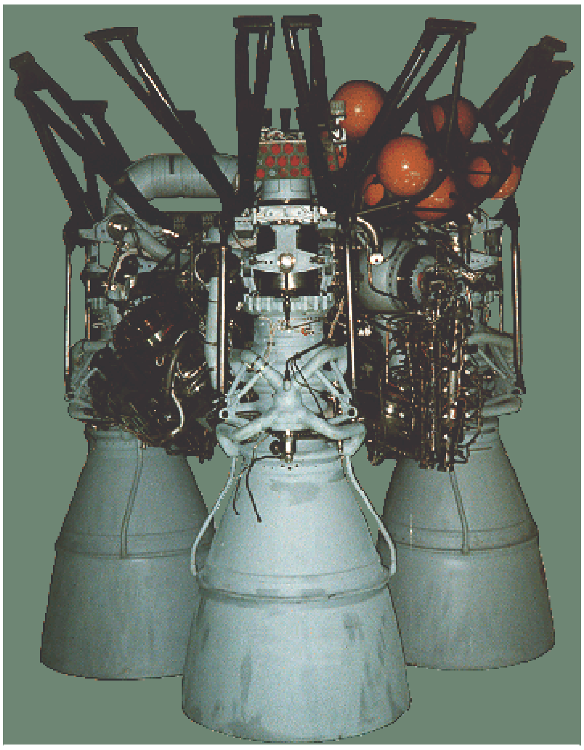
RD-180

mission had been granted to sell the engine abroad, representatives of *NPO Energomash* turned their eyes westwards immediately. In an interview Mr. F.Chelkis said the engine did not require additional modification to suit any specified rocket and could be offered for sale at any time out of the available stock. However, according to *NPO* leaders, RD-170 will not to be supplied for military applications.

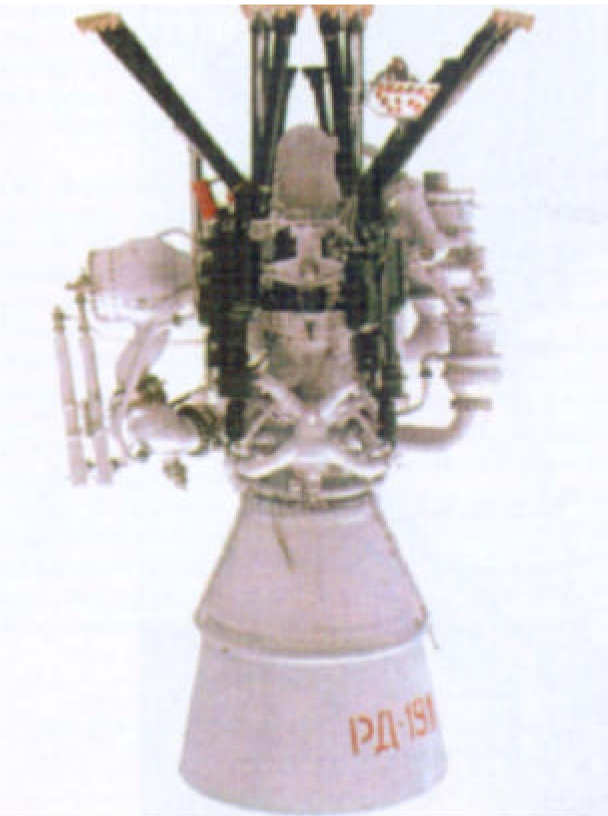
Top managers with *Energomash* did their best to make sure the issue of technology transfer was addressed adequately as RD-170 was offered for sale, and assumed that the design of the liquid propellant engine being sold could be protected as a commercial secret.

Mr. F.Chelkis said in end-1991 that the RD-170 engine augmented by *Energomash's* manufacturing and testing capability could "dramatically reduce the expenditure" by the US on their national space program. However, apart from the obstacles resulting from trading restrictions, any application of a liquid-propellant rocket engine of that power has been and remains limited by non-existence in the United States of a launch vehicle in which it could be incorporated.

Engineers with Pratt & Whitney are known to have been considering a two-stage, non-recoverable medium-class LV with RD-160 as the first stage engine. Provided the second stage of the described LV includes the lox/kerosene RD-120, the system could be used to inject max. 12.7 t payload into a low orbit, i.e., to show performance similar to *Zenit-2*. In the meantime, with the advanced



RD-171



RD-191

lox/hydrogen RL-200 liquid-propellant rocket engine by Pratt & Whitney as the second stage motor, the payload capacity could be raised to as much as 22.45 tons causing the LV to surpass the performance of *Proton*. Finally, with the second stage provided with two traditional lox/hydrogen RL-10A-4 engines (an advanced modification of Atlas-Centaur's second stage engine) the PL capacity could reach 9.75 tons for low orbit missions, placing the LV in between *Soyuz* and *Zenit*-2 LVs.

Another potentially feasible application for RD-170 is the engine's incorporation into liquid-propellant boosters designed to replace Space Shuttle System's solid-propellant booster units. According to engine designers, that would enhance the launch vehicle's flexibility and reliability owing to the added thrust control and liquid-propellant engine's in-flight cut-off capabilities not available to solid-propellant rocket engines. In addition, liquid-propellant boosters show greater environment-friendliness and provide for Space Shuttle's greater payload capacity in low orbit injec-

Table 3. ***RD-170 vs. RD-120K: COMPARED PERFORMANCE***

Performance Characteristic	RD-120	RD-120K
Engine configuration	Closed-loop, w/afterburning	
Oxidiser	Liquid oxygen	
Fuel	Kerosene	
Oxidiser/fuel ratio	2.6	2.6
Ground thrust, tf	-	80
Vacuum thrust, tf	85.0	86.72
Specific impulse on ground, s	-	304.4
Specific impulse in vacuum, s	350	330.5
Specific fuel consumption, kg/s	242	263
Combustion chamber pressure, atm	166	179.8
Nozzle area ratio	114.5	4.96
Engine mass, kg	1,285	1,080
Engine mass-to-thrust ratio, kg/tf	15.12	12.45
Height, m	3.87	2.435
Diameter, m	1.95	1.080
Turbopump assembly horsepower	17,500	20,600
Turbopump assembly r.p.m.	19,000	2,220
Guaranteed duration, s	2,220	2,220

tions.

The RD-170 liquid-propellant rocket engine combined with an external expendable fuel tank and three SSME lox/kerosene engines of the Space Shuttle System could provide basis for the development of a heavy launch vehicle designed to bring payloads into low orbits. A rocket of that configuration with two liquid-propellant boosters could put a payload of 70.76 tons into orbit, while six boosters would raise the payload capacity to as high as 176 tons.

Various RD-170 design options have been under development since late eighties for advanced national and foreign launch vehicles. Specifically, the thrust of the basic liquid-propellant rocket engine was supposed to be increased by raising the chamber pressure from 250 to 263.5 atm, which required insignificant modification of certain components so as to reduce the overall engine mass. The described design option was proposed for *Zenit-2* LV modification intended to carry both unmanned spacecraft, like the cargo *Progress* or modules of orbital stations, and launch manned transportation spacecraft like Soyuz.

It was at that time that designers with *NPO Energomash* proposed a new engine based on RD-170. The advanced **RD-180** could be used on both the new or advanced, national or foreign LVs and as a liquid-propellant booster for the Space Shuttle System. The design of the RD-180 engine includes two of the four combustion chambers available on the predecessor, and several standard or upgraded units and assemblies of the basic design engine, like the turbopump assembly or other fuel feed assemblies. The Russian-US *RD-AMROSS* Joint Venture made up by Russia's *NPO Energomash* and the United States' Pratt & Whitney of United Technologies is known to be manufacturing the engines for the American advanced launch vehicles of Lockheed Martin's Atlas-3A and Atlas-5 family (*see NFM Special Issue 8/99 for details*). Current stock of orders counts as many as 18 engines (*Editor's note: Lockheed Martin intends to buy from Energomash a total of 101 engines over the next 10 years*), and there is a possibility to initiate manufacturing of RD-180 in the United States for installation on launch vehicles designed to orbit governmental payloads.

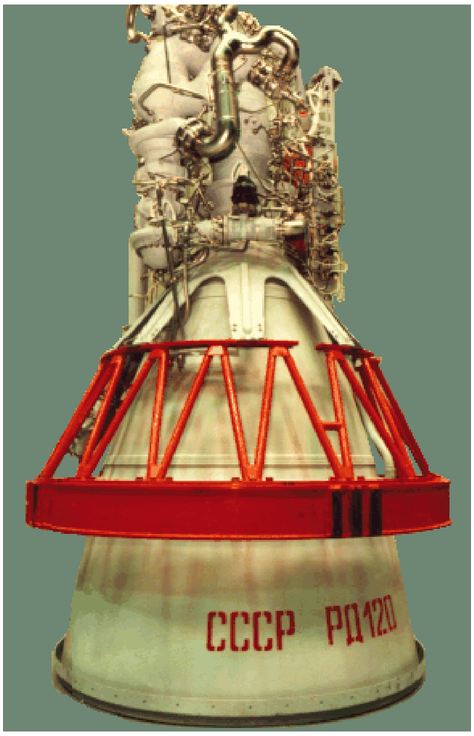
Furthermore, work is underway at *NPO Energomash* to develop the single-chambered RD-191 engine based on the RD-170 combustion chamber design, new automatics and new turbopump assembly. Brief performance characteristics of the modified RD-170 engine are presented in Table 2 along with RD-180's and RD-190 performances.

RD-120/RD-120K

The development of the RD-120 liquid oxygen/kerosene engine that was presented originally at the *Aviadvigatel-91* air engine show was led between 1976-1986 by Chief Designer *V.P.Radovsky* under the Statement Of Work issued by *KB Yuzhnoye* Design Bureau. The engine was intended for the second stage of the medium payload capacity *Zenit-2* launch vehicle earmarked as future replacement of the *Soyuz* family launch vehicles.

The RD-120 is a high-altitude engine with turbopump-assisted fuel feed system. The design of the engine provides for exhaust turbine gas afterburning. Fuel components are ignited by means of self-igniting starting fuel.

The engine includes the combustion chamber with dual-component (gas-fluid) injectors, the high-altitude nozzle, the turbopump assembly incorporating the oxidizer gas generator in which the entire liquid oxygen (i.e.,



RD-120

RD-120K

the oxidizer) is being converted and heated, and the automatic start-up and operation control systems. Thermal insulation of the combustion chamber's wall is ensured by the fuel-enriched wall boundary layer generated by the chamber injector head, and two film cooling belts (slots).

The engine went through full-scale development, bench trials and flight tests. Ground test-firing of the fully built-up engine with a nozzle was conducted on the exhaust diffuser test bench providing for simulated high-altitude operation of the engine.

Similar to what happened as RD-170 was developed, the designers of RD-120 encountered fire accidents in the engine's hot gas-flow passage, failure of the turbopump assembly bearings and high-frequency vibrations in the combustion chamber. Some experts would explain the accidents by the less than adequate design of the combustion chamber injector.

The RD-120K engine, a modification of RD-120 distinguished for a shorter nozzle, differs from the RD- 120 design in modified layout of the engine's top. Indeed, the RD-120 version installed in the central opening of the *Zenit-2* second stage's toroidal tank looks somehow compressed on its sides and extended vertically. Unlike that, the RD-120K mounted on LV's bottom face is shorter and differs in the layout of the top components, including the combustion chamber, turbopump assembly, gas generator, piping and automatics.

The RD-120K developed as part of home initiative by *NPO Energomash* is designed for use in the first stages of future light- and medium-class launch vehicles. To support the proposed applications, the engine is provided with both the incorporated thrust control capability (3.5 percent throttling up or 8.5 percent throttling down) and the mixture ratio variation capability (± 10 percent). The engine can be fixed or hinge-mounted, as required to ensure thrust-vector control capability within ± 6 degrees.

The described engine is proposed for installation on lower stages of the advanced *Soyuz-2* (*Russ* Project) and *Yedinstvo* (Unity) launch vehicles developed, accordingly, by the *Progress* Design Bureau and the *Makeyev*. A comparison of RD-120 and RD-120K performance is provided in Table 3.

RD-161/RD-161P

The experimental RD-161 and RD-161PV engines, which are among the latest developments by *NPO Energomash*, were originally presented at the *K Zvezdam '93* (To the Stars-93) show.

The liquid oxygen/kerosene RD-161 engine offered by *Energomash* is a home initiative development designed for liquid-propellant engine applications in upper stages of launch vehicles and/or orbital transfer vehicles (*Block DM*), including both the existing and future vehicles. The design of RD-161P in which the propellants are high-concentration (93-97 percent) hydrogen peroxide and kerosene is based on an earlier prototype initially designed for the orbital maneuver system of the *MAKS* Project's spacecraft.

Both engines are single-chambered units with turbopump fuel-feed system and exhaust turbine gas afterburning. The ignition of the fuel components in the RD-161 engine is ensured by plasma generated as high-voltage, high-frequency current is passed through gaseous oxygen (a development by *RKK Energia* initially implemented in *Buran's* steering microengines).

The engines include practically identical combustion chambers with dual-component (gas/fluid) injectors, high-altitude nozzles and turbopump assemblies. However, there is difference in the gas generation flow duct of the described liquid-propellant engines.

The gas generator duct of the RD-161 engine is a dual-component, single-injector unit (with rotary-type fuel injection and peripheral oxidizer atomization). The system is designed to operate under rich oxidizer conditions as it generates hot gas at approx. 550°C essentially

Table 4. ***RD-161 vs. RD-161P: COMPARED PERFORMANCE***

Performance Characteristic	RD-161*	RD-161P
Engine configuration		
Oxidiser	Liquid oxygen	Hydrogen peroxide
Fuel	Kerosene	Kerosene
Oxidiser/fuel ratio	2.6	5.9
Vacuum thrust, tf	2.03/2.0	1.5/2.5**
Specific impulse in vacuum, s	365/360	319
Specific fuel consumption, kg/s	5.5.6	7.84
Combustion chamber pressure, atm	120	120
Nozzle area ratio	19.25/18.75	16.28
Thrust duration, s	900	900
Total firings per mission	Max. 17	50+
Hinge angle, degrees	± 6	-
Engine mass, kg	141/119	105
Engine mass-to-thrust ratio, kg/tf	69.46/59.5	51.22
Height, m	2.205/1.700	1.450
Diameter, m	1.020/0.780	0.540
Turbopump assembly horsepower	444	500
Turbopump assembly r.p.m.	80,000	-

**) numerator/denominator values show performance of engine with/without nozzle extension;*

***) numerator/denominator values show single/dual-component mode performance*

composed of oxygen and a certain percentage of water steam and carbon dioxide gas.

The gas generator duct of the RD-161P engine is a single-component catalytic system: hydrogen peroxide passed through the catalyst pack is decomposed to produce hot steam-and-gas at approx. 850°C, containing basically steam of water and oxygen.

Upon working on the turbopump assembly blades, the steam-and-gas is fed to the combustion chamber for afterburning by means of the fuel. Owing to the single-component design of the gas generator, the fuel supply and engine start-up system are substantially simplified. There is a feature worth being mentioned specifically if RD-161P is to be described: in case no fuel is fed to the combustion chamber, the engine will operate in the so-called single-component mode and develop remarkably high thrust. The greatest difficulty in the development of the fuel supply system was encountered as the materials for the gas generator's catalyst package were selected. Table 4 presents compared performance of the RD-161 and RD-161P engines.

NPO Energomash have been out of small-size engine development business for almost forty years. Yet, designing a small liquid-propellant rocket engine with high specific impulse fuelled by environment-friendly fuel components is really on the agenda.

In reviewing engines operating on the hydrogen peroxide/kerosene pair of fuel components, one cannot but mention the family of Britain's Gamma liquid-propellant rocket engines developed in the early and mid-sixties and installed on the Black Knight high-altitude rocket and the Black Arrow launch vehicle. Perhaps, no one else but the British can boast expertise in designing, development and production of peroxide engines.

The scientific and production potential available to the *Energomash* Association allows new technology to be mastered within the shortest possible time if the required liquid-propellant rocket engine is to be created. Let us not forget the available expertise with experimental engines operating on hydrogen peroxide/kerosene and hydrogen peroxide/pentaborane fuel: those were developed and tested by *Energomash* in the mid-sixties as part of the Soviet Lunar program.

The RD-161 includes an original carbon plastic nozzle extension, some 500 mm in length. The radiation-cooled extension provides for the engine's Isp to be increased approximately 5-fold.

The RD-161 engine should be described as an advanced unit: indeed, even though the more energy-capacious *Synthine* synthetic hydrocarbon fuel is not being used, the engine shows the highest specific impulse among all available Lox/kerosene engines. Experts believe turning to *Synthine* fuel is possible in the future: despite certain potential problems with cooling, the new fuel could add at least 10 units to the engine's specific impulse parameter.

In the proposed configuration, the RD-161P engine is quite small-sized, essentially because the nozzle is relatively short. Despite the fuel-saving scheme including generator gas afterburning, the 161P version provides comparatively low specific impulse (in fact, it is even lesser than that of similar thrust level engines by *KBKhM* fuelled by the nitrogen tetroxide/asymmetric dimethyl hydrazine pair of fuel components). Representatives of *Energomash* claim their RD-161P peroxide-fuelled liquid-propellant rocket engine is intended to prove feasibility of a relatively simple small-size recoverable motor to be fuelled by non-toxic, environment-friendly propellants, that could provide for essentially unlimited duration of space operation. No task to attain any ultimate specific impulse was set, for as long as the value could be increased by 10-15 units in the future, if required.

NK-33/NK-43

Many Russia's and international companies have been displaying major interest lately in the NK-33 and NK-43 rocket engines by *Dvigately NK* of Samara. The engines were presented to public for the first time at *Aviadvigatel-90* show.

The two products are Russia's first ever recoverable liquid-propellant rocket engines originally developed more than 25 years ago. Since then, the engines have passed full-scale

Table 5. *NK-33 vs. NK-43: COMPARED PERFORMANCE*

Performance Characteristic	NK-33	NK-43
Engine configuration	Close-loop. w/afterburning	
Oxidiser	Liquid oxygen	
Fuel	Kerosene	
Oxidiser/fuel mixture ratio	2.8	
Ground thrust, tf	154	-
Vacuum thrust, tf	171	179
Specific impulse on ground, s	298	-
Specific impulse in vacuum, s	331	346
Specific fuel consumption, kg/s	517.3	517.3
Combustion chamber pressure, atm	147	147
Nozzle exit section pressure, atm	0.55	0.13
Engine mass, kg	1,340	1,400
Engine mass-to-thrust ratio, kg/tf	8.1	7.8
Turbopump assembly horsepower	46,000	46,000
Turbopump assembly r.p.m.	20,000	20,000
Thrust duration, s	600	600

NK-33 mock-up

bench trials to confirm the initially incorporated performance. Yet, neither was ever installed on any craft, even despite batch production was in place.

Prototype NK-33 lox/kerosene engine known to broad public as NK-15 was developed between 1962-1970 at the *Kuznetsov* Design Bureau in Kuibyshev for the first stage of the super-heavy Lunar N-1 launch vehicle.

N-1's second stage included NK-15V engines - the high-altitude modification of the first stage liquid-propellant rocket motor, differing from the original engine mainly by greater nozzle area expansion ratio.

Flight tests of N-1 LV showed the first stage engines required additional development effort, let alone inefficiency of the adopted quality control practice applied to the liquid-propellant rocket engines being supplied. Those factors stimulated *Kuznetsov* Bureau already in mid-1970 to initiate development of basically new recoverable liquid-propellant engines for longer service life.

The NK-33 engine was developed between 1970-1974 based on a novel work statement accounting for the initial flight test results of N-1 rocket. The idea was to dramatically improve reliability, ensure failure-free performance and safety of the engine without changing the layout, though somewhat boosting the thrust and raising the specific impulse. Interdepartmental tests of the first stage engines (the NK-33) and the second stage engines (NK-43) were finally completed in September 1972.

Engine recovery capability allowed each commercial engine to be proof-tested first and retested later as part of the overall rocket system.

Similar to its prototype, the NK-33 engine is a single-chambered unit with a turbopump fuel feed system designed for environment-friendly non-self-inflammable fuel (the fuel is kerosene, the oxidizer is liquid oxygen). The unit is a close-loop engine with the exhaust turbine gas afterburned at high pressure in the main combustion chamber. The propellant for the turbopump assembly's turbine is the burning products of the main propellants burned in oxidizer-rich environment. Practically the entire oxidizer is gasified in the gas generator with just a little fuel added.

The NK-33 engine differs from the prototype NK-15 unit by a simplified hydro-pneumatic scheme, more advanced automatics and improved turbopump and combustion chamber assemblies. For instance, the numbers of igniter automatic components incorporated in the design dropped from twelve to only as few as seven. Plug-in connections and interchangeability of parts add to the engine's maintainability.

Certain NK-33 production units delivered the thrust of almost 205-207 tf on test bench as the gas generator burning was intensified (which manifested itself in temperature rise); that meant those engines were to be classified as units of a different thrust standard. The thrust control range (varied between 50 and 105 percent) for NK-33 was dictated essentially by the available service life of the engine. With the required service life reduced only insignificantly, the thrust control range could be increased to as high as 135 percent.

Despite the design of NK-33 includes a starting pyroturbine, that engine has lesser weight compared to *Energomash's* RD-253 installed in the first stage of the *Proton* launch vehicle. Indeed, NK-33 engine may well be described as one of the world's best in terms of mass-to-thrust ratio.

In its turn, the NK-43 engine is the world's most powerful high-altitude lox/kerosene rocket engine. The engine's top (combustion chamber, turbopump assembly, automatics and the initial section of the nozzle) is similar to that of the NK-33. However, the nozzle area expansion ratio is much greater. Engine reliability was verified by increasing total firings to as many as ten.

According to some foreign experts, the remarkable performance of the engines manufactured in Samara allows one to assume large-scale co-operation of foreign partners with the Russian manufacturer is quite possible, starting, for instance, from joint Russian-US design efforts for new liquid-propellant rocket engines to direct installation of engines by N.D.Kuznetsov on the existing and future foreign rockets.

As an example, by replacing two booster engines currently installed on Atlas LV with two NK-33s, payloads injected to transfer orbits for eventual geostationary orbiting could be increased from 3,630 kg to 4,173 kg.

By agreement with *Dvigately NK*, Aerojet General doing marketing research for the engines in Western markets performed five bench trials of NK-33 at their test facility in Sacramento.

So far, *Dvigately NK* have been known to have orders in stock for delivery of NK-33 and NK-43 engines for future K-1 launch vehicles by the United States' Kistler Aerospace, as well as for *Yamal* LV (*see NFM 12/98 for details*) by *Energia* and the Air Launch developed by the corporation of the same name (*see NFM 15/2000*).

NK-31/NK-39

The attractiveness of the NK-31 and NK-39 engines by *Dvigately NK* is explained by essentially the same reasons as is the case with NK-33 and NK-43.

Development of the prototype NK-31 engine, known to public as NK-9 and designed originally for the first stage of the GR-1 global rocket by Korolyov OKB-1, began in 1959. The GR-1's second stage included a high-altitude version of the same liquid-propellant engine designated as NK-9V. That latter one was in fact the immediate predecessor of NK-39 and NK-31.

After development works on GR-1 were discontinued, the NK-9V version with ex-

Table 6. ***NK-39 vs. NK-31: COMPARED PERFORMANCE***

Performance Characteristic	NK-39	NK-31
Engine configuration	Close-loop. w/afterburning	
Oxidizer	Liquid oxygen	
Fuel	Kerosene	
Oxidiser/fuel ratio	2.6	2.6
Engine mounting	Fixed	Hinged
Vacuum thrust, tf	40.8	41.0
Specific impulse in vacuum, s	352	353
Specific fuel consumption, kg/s	116.1	116.1
Combustion chamber pressure, atm	98	98
Engine mass, kg	700	722
Engine mass-to-thrust ratio, kg/tf	17.2	17.6
Nozzle exit section diameter, m	1.3	1.4
Rotation speed, r.p.m.	32,000	32,000
Thrust duration, s	1,200	1,200

tended service life was used in the upper (the 3rd and the 4th) stages of the super heavy Lunar N-1 carrier rocket. In order to suit for the new launch vehicle, the engine power was degraded through a reduction of the turbopump operation intensity.

The NK-39 engine for N-1's third stage, and the NK-31 for the same rocket's fourth stage were developed between 1970-1974 based on a work statement issued anew to allow for the initial flight test results of N-1 rocket and improve reliability, ensure failure-free performance and safety of the engine without changing the layout. Interdepartmental tests of the engines were completed by 1972.

Engine recovery capability allows each commercial unit to be proof-tested first and re-tested later as part of the overall rocket system.

The NK-31 engine is a single-chambered unit with a turbopump fuel feed system designed for environment-friendly non-self-inflammable fuel (the fuel is kerosene, the oxidizer is liquid oxygen). The unit is a close-loop engine with the exhaust turbine gas afterburned at high pressure in the main combustion chamber. The propellant for the turbopump assembly's turbine is the burning products of the principal fuel components burned with high excess of oxidizer.

The NK-31 and NK-39 engines differ from the prototype NK-9V unit by a simplified hydro-pneumatic scheme, more advanced automatics and improved turbopump and combustion chamber assemblies. Plug-in connections and interchangeability of parts add to the engine's maintainability.

Pitifully, similar to NK-33 and NK-43, neither the prototype engine, nor NK-39 or NK-31 have never been flown realistically in any LV, even despite numerous proposals were made both before and after the described liquid-propellant engines were used in the N-1 Project. Certain interest in the engines has been recently shown by national and foreign companies (including a business from France planning installation on the VEHRA demo aircraft).

KVD-1

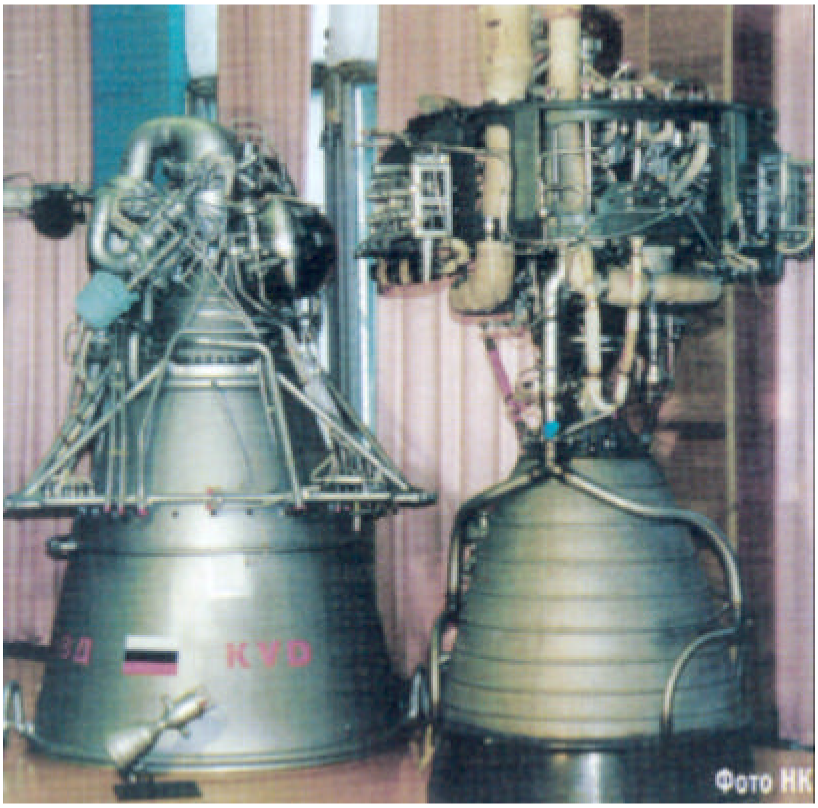
So far, the one and only oxygen/hydrogen liquid-propellant rocket engine in Russia known to have passed through full-scale ground testing routine has been the KVD-1 engine. Even despite more than 35 years have

Table 7. ***KVD-1 vs. RL-10A-3: COMPARED PERFORMANCE***

Performance Characteristic	KVD-1	RL-10A3-3*
Engine configuration	Close-loop	Close-loop
Turbopump assembly gas generation method	Dual-component gas generator	Hydrogen conversion in the cooling jacket
Control elements drive	Electropneumatic	Electromechanical
Ignition	Pyrotechnical	Electric spark
Combustion chamber pressure, atm	57	28.3
Gas generator pressure, atm	82.3	-
Cruising chamber thrust, kgf	7,100**	6,830
Specific impulse, s	462	444
Fuel consumption, kg/s	15,37	15.39
Components ratio	6.0	5.0
Dry engine mass, kg	282	132
Height, m	2.14	1.78
Diameter. m	1.58	1.0
Turbine rotor speed, r.p.m.	42,000	30,240

**) developed in 1958-68 by Pratt & Whitney for upper stages of Atlas-Centaur and Saturn-1*

***) the thrust of each of the two steering chambers: 200 kgf*



KVD-1 (left) and its precursor 11D56

gone since the time its development was first initiated, the engine has never been tested in flight. In the meantime, that is precisely the one Indian Space Research Organization selected for the cryogenic upper stage of their GSLV rocket.

KVD-1's prototype known as *11D56* was developed between 1965-1972 by *Khimmash* Design Bureau under a work statement issued by S.P.Korolyov's OKB-1 Bureau. It was initially designed for the fourth stage (the boost/deboost system) of a future version of heavy Lunar N-1 launch vehicle. Bench trials of the engine commenced in 1966.

KVD-1 engine is a single-chambered unit with a turbopump system designed to feed propellants; unlike RL-10A-3 - a similar US liquid-propellant engine of the same thrust standard - KVD-1's design includes afterburning: a feature characteristic of any powerful Russian liquid-propellant rocket engine design. Yet, while a gas generator producing oxidizing turbine gas (the so-called sour gas, i.e., containing excess oxygen) is used to drive turbopump assemblies of national non-hydrogen close-loop engines, the case is different with KVD-1's gas generator, where the propellant is burned with excess fuel to produce reducing (the so-called sweet) gas.

See Table 7 for compared performance of KVD-1 and RL-10A-3 engines.

KVD-1 includes a single-shaft turbopump assembly. To make the oxidizer and the fuel pumps comply with their cavitation characteristics at equal speed of the turbopump shaft, booster pumps are provided in the design. With built-in vane boost pumps incorporated in the propellant components feeder lines, the described liquid-propellant engine can operate with outside, external drive boost oxidizer and fuel pumps delivering the required propellant components pressure to ensure cavitation-free operation of the main turbopump assembly pumps.

Ignition of the propellant components during engine starting is attained by initially igniting the propellants in the gas generator (by means of pyrotechnical devices); thereafter, the gas flowing off the gas generator is fed to the turbine and later - to the combustion chamber for afterburning, where lacking oxidizer is added.

Despite no specific purpose of the KVD-1 engine was disclosed within the framework of current Russian space program, an apparent assumption is that the unit can be used in cryogenic upper stages designed to put payloads into high-altitude elliptical, geostationary orbits or escape trajectories.

The first ever cryogenic upper stage with KVD-1 will be 12KRB - the third stage of India's GSLV designed and built by Russian specialists of the *KB Salyut* Bureau and *Khrunichev*. The one to follow will be the KVRB alias the fourth stage of the future *Proton-M* LV which is to replace today's *Proton* with its lox/kerosene *DM* block. Utilization of the oxygen/hydrogen propellant instead of the oxygen/kerosene is anticipated to add some 30-55 percent to payload capacity of those LVs as satellites are put into geostationary orbits.

In addition, the oxygen/hydrogen upper stage can make a good substitute for the *DM* block of today's *Zenit-3* LV (at Sea Launch) or a future modification of same, thus allowing the LV's capabilities to be dramatically enhanced during high-altitude elliptical and/or geosynchronous satellite launches.

S5.92

The prototype S5.92 engine was developed as part of the Soviet N-1-L3 Project and was to be installed on the "*Zh*" block or the power unit designed for departure of the manned Lunar spacecraft integral to the N-1 - L-3 system from the Lunar orbit back to Earth. KRD-1 engines were also developed in parallel with the described engine to provide lift-off capability from the Lunar surface and return mission to Earth for the return rockets of the automatic space stations Luna-16, -20, -24; in addition, development works continued on the KTDU-417 engines designed to provide mission trajectory correction and Moon landing of those stations. In fact, the mentioned engines had much in common, even despite their propellants differed.

The development of today's S5.92 began in 1978. The engine is designed for the multifunctional rocket unit integratable into a variety of spacecraft, including *Fobos-1*, *-2* and *Mars-96*. The power plant of which S5.92 is a part was to boost up an automatic interplanetary station after separation from the final stage of the LV to as fast as the escape velocity and to ensure numerous flight trajectory corrections, deboosting, Martian orbit entry and orbit maneuvering.

Though the missions of the mentioned stations failed, nothing could be said to reprimand the power plant, and a proposal followed to use the S5.92 engines in both the existing and the future power plants of spacecraft and launch vehicles.

The single-chambered S5.92 engine is an open layout unit with the propellants fed by a turpopump. The turbine is operated by the main propellants, the exhaust is effected through fixed steering nozzles. A feature of the engine is the unusually hinged (rather than gimballed) mount of the chamber to ensure its plane-parallel motion inside the power plant. The approach provides for the engine thrust vector to be displaced relative to the center of masses, which in the described power plant is found very close to the engine head (under the situation, gimballing would not provide the arm required to generate the moment of thrust). S5.92 engine is operational in two modes: the high-thrust mode and the low-thrust mode. High-thrust mode is applied in major speed alteration maneuvering, while low-thrust mode is

Table 8. ***S5.92 ENGINE PERFORMANCE***

Performance Characteristic	High-Thrust	Low-Thrust
Engine configuration	Open	
Oxidizer	Nitrogen tetroxide	
Fuel	Asymmetric dimethyl hydrazine	
Oxidizer/fuel ratio	1.95-2.05	2.0-2.1
Cruise chamber thrust, kgf	2,000	1,400
Exhaust nozzle thrust, kgf	40	19
Specific impulse, s	327	316
Fuel consumption, kg/s	6.12	4.43
Combustion chamber pressure, atm	98	68.5
Gas generator pressure, atm	118	61
Turbopump assembly r.p.m.	58,000	43,000
Total thrust duration, s	2,000	
Overall dimensions, m	0.677 x 0.838 x 1.028	
Dry engine mass, kg	37.5	

applied in maneuvers that require precision speed pulse variation.

Table 8 presents S5.92 performance characteristics

Among the features of the described S5.92 engine (that are similar to actually any liquid-propellant rocket engine by *Khimmash*) are the extremely small dimensions, low specific mass with quite high thrust and specific impulse values. By the said parameters, the engine outruns the similar class RD-161M by *Energomash*, even despite the more perfect close-loop configuration with afterburning is used in the latter. Portability of liquid-propellant engines by *Khimmash* is basically achieved owing to optimized combination of the turbopump assembly parameters, the combustion chamber pressure and the nozzle area expansion ratio. In addition, an open configuration allows a liquid-propellant engine wherein the vertical dimensions would not exceed the dimensions of the combustion chamber.

Most similar Western analogies to S5.92 are the liquid-propellant engines developed in the late eighties for the US Air Force ASPS upper stage. Those are: XLR-132 by Rocketdyne and Transtar-1 by Aerojet Tech Systems, both operating on the AT - Aerozin-50 propellant (a mixture of 50 percent hydrazine and 50 percent asymmetric hydrazine). However, neither of the two engines ever went beyond the test bench prototype.

The S5.92 engine has been installed on

RD-0110

Model of RD-0124M

the *Fregat* multipurpose rocket unit by *Lavochkin* (*see NFM 4/2000 for details*) - the one designed for use in the existing (*Proton*, *Soyuz*) and future (*Russ*) LVs as the top stage for payload injections into a variety of orbits, including solar-synchronous, high-altitude elliptical (geo-transfer), high-altitude circular (geostationary) orbits and/or mission trajectories to the Sun, the Moon, planets, comets or asteroids. The *Fregat* upper stage can perform as a cruise power plant for spacecraft, orbital transfer vehicles, orbital or orbital-landing modules.

RD-0124

The second half of the eighties was marked at the *KBKhA* Design Bureau for Chemical Automation of Voronezh by commenced development of the advanced version of a Lox/kerosene engine to replace the standard RD-0110 mounted in the third stage of the *Soyuz* family LVs. The work went on within the frame of the national space program and transformed in the early nineties in the *Russ* (*Soyuz-2*) Project.

The basic version of the RD-0124 engine designed to replace RD-0110 is a four-chamber unit. The pivot chambers are used to provide

Table 9. *RD-0110 vs. RD-0124: COMPARED PERFORMANCE*		
Performance Characteristic	**RD-0110**	**RD-0124**
Engine configuration	Open	Close-loop
Propellant components	Liquid oxygen/kerosene	
Vacuum thrust, tf	30.0	30.0
Specific impulse in vacuum, s	`320	353
Total combustion chamber	4	4

RD-0124

launch vehicle control at the third stage leg of the flight. Staying within the same dimensions as they switched over to close-loop configuration with exhaust turbine gas afterburning in the main chambers and increased chamber pressure and nozzle area expansion ratios, engine designers from Voronezh succeeded in raising the specific impulse by as many as 33 units, which is equivalent to adding 950 kg to *Soyuz'* payload capacity.

The engine has been through a comprehensive cycle of test trials, and preparatory work is going on now to launch batch production at the Voronezh Mechanical Plant.

An attractive option (for a foreign customer, in particular) is the single-chamber version of the engine designated RD-0124M complete with assemblies (turbopump, gas generator, piping, automatics) borrowed from the four-chamber version. The chamber is completely new, while the nozzle borrowed from a submarine-launched ICBM can act as either a high-altitude or a ground unit.

Some words also must be said about the most popular problem of using hydrogen for rocket fuel. Hydrogen is called the rocket fuel of the future. In this field the vast experience has been accumulated by the *KBKhA* design Bureau. The Russian **RD-0120** Engine is not at all inferior to the American SSME, which powers the Shuttles.

The engine is designed with the staged combustion cycle (the turbine gas is supplied to the main injector). RD-0120 is an advanced liquid rocket engine which combines high performance, long service life, and low operation cost.

The engine allows reusability applications and considerable thrust throttling. The engine has a system of diagnostics and safety system.

The engine operates with the ecologically tidy propellants - liquid oxygen and liquid hydrogen.

The engine provides helium heating for the oxidizer tank pressurization, produces gaseous hydrogen for the fuel tank pressurization, and on-board power supply systems.

During the phase of development the engine was fire-tested 800 times with total testing duration of 170000 sec. Eight engines were tested on the *Energia* launcher.

RD-0120 Performance features:

Vacuum thrust	200 tf
Vacuum specific impulse	455 kgf- s/kg
Propellants	LOX/LH_2
Combustion Chamber pressure	228 kgf/cm^2
Operating duration	500 c
Total engine weight	3450 kg
Overall dimensions	
Length	4550 mm
max. nozzle exit diameter	2420 mm

Rocket Engine Manufacturers In Russia