RUSSIAN SPACE PROGRAMS: ACHIEVEMENTS AND PROSPECTS OF AUTOMATIC CONTROL APPLICATIONS

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Abstract*:* Performance of manned and automated spacecraft and vehicles greatly relies on the control system operation quality. The account hereunder describes the history and evolution of control systems, principles of their development, and the latest concepts.

Keywords*:* attitude control system, automatic rendezvous and docking system, pulsed linearization, width-pulsed modulation, gyrodyne, gyroscope, strap-down inertial navigation system, Sun tracker, star tracker.

1. INTRODUCTION

As from the first most spacecraft and vehicles, the motion control system belonged to the very important onboard systems and its role was enhancing with the expansion of its functions. All technical characteristics of modern spacecraft and manned systems are linked in one way or another with the control system performance and rely mainly on its operation quality.

The successful expansion of technology (in the first instance, onboard computers) has given way for the wide modification of control systems and enhancement of their capabilities with the concurrent improvement of their performance.

Progress in improving control systems gave the ability to develop multi-purpose, multi-mode control systems with a long active life, systems allowing maximum autonomy and automated control processes.

The modern systems have the benefit to execute not only simple (program-time) motion control of spacecraft, but also the more complex control functions based on a positional method.

A 60-year experience gained from the development and operation of control systems enabled to define certain philosophy and trends in developing systems to handle both program and crew safety assurance.

2. AUTOMATIC CONTROL SYSTEMS

2.1 AUTOMATIC ATTITUDE CONTROL AND STABILIZATION SYSTEMS

2.1.1 Pulse-relay systems

The automatic Luna-3 spacecraft that has photographed the far side of the Moon in 1959 and spacecraft Vostok that has flown Yu. Gagarin in the first orbital flight are regarded as the first controllable spacecraft.

The attitude control systems employed by the first space objects were designed for the short-term

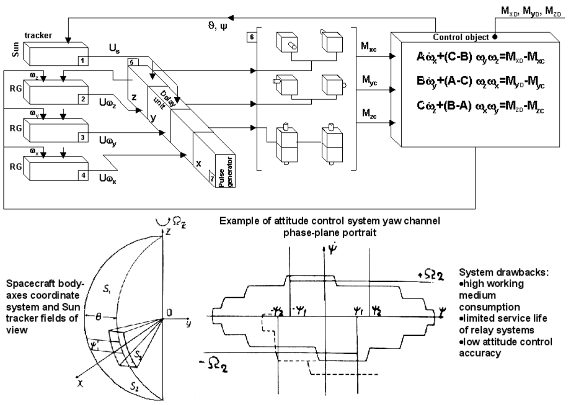


Fig. 1. Spacecraft Vostok pulse-relay system (Direct control – single-loop system)

operation and restricted functions. Presently, the first attitude control systems are better attractive from the historical viewpoint. However, the fact that those systems have not had prototypes neither in our state, nor abroad is worthy of attention. Their development, with the consideration of radically new specifics of activities in space environment, appeared to become a rather complex technological problem.

As an example, consider now the Vostok automatic attitude control system. This is a pulse-relay control system designed for the single-axis Sun-pointing before the braking engine ignition. The system simplified block-diagram is shown in Fig. 1.

The system is designed of a photo-electronic Sun tracker 1, three RG (rate gyro) 2,3,4 responding to the spacecraft angular rate projections ω, logic module 5, and actuators 6 – low-thrust attitude control engines using compressed nitrogen. The Sun tracker responds to the spacecraft rotations (relative to Sun-pointing) around axes 0Y and 0Z.

The simplest task is to control rotation around axis 0X. The control is confined to damping the spacecraft angular rate ω_x.

The stabilization along axes 0Y and 0Z is done by comparing signals from the Sun tracker and the respective angular rate sensors in logic module 5.

The experience gained from successful flights of Vostok and Voskhod spacecraft gave evidence to the possibility of achieving the required attitude control accuracy, provided the given system would be employed.

2.1.2 Direct control analog systems with pulsed linearization

A number of shortcomings were peculiar to the pulse-relay automatic systems. Among these are:

- relatively high working medium consumption;
- limited service life of relay elements;
- low accuracy of attitude due to hysteresis and large dead zones of sensors.

That is why the development of direct control analog systems with the pulse-width modulation became the next step in the evolution of control systems for automated and manned spacecraft.

The control systems of such spacecraft as Zenith, Mars, Venera, Progress, Soyuz, and Space Station Salyut fall into this type.

All the aforesaid systems employed attitude control jets in the capacity of actuators. The jets are fired by an analog control signal that is generated as a combination of signals from the angular displacement and angular rate sensors.

The law of pulsed linearization makes for the increase of pulse duration τ and duty factor τ/t, as the control signal enhances.

With the signal decrease to a certain threshold value, a minimum time signal is generated on the pulse converter output and the attitude control jet is capable to respond to this signal. The threshold values are selected, assuming that the fuel consumption is minimized.

The further development of the direct control with pulsed linearization made a basis for control systems of a large family of spacecraft over 20 years.

Evolution of such systems was completed during the development of Space Station Salyut integrated in the orbital complex Soyuz-Salyut-Progress.

In solving the problems of how to control the orbital complex, one of the most important tasks was to assure a high quality of attitude control and stabilization processes with the consideration of the structure flexibility throughout the mission, i.e. irrespective of the added-on complex elements, re-distribution of cargoes and propellant, etc.

The principle of width-pulse modulation that formed a basis of direct control systems was extended equally well to 2-circuit systems controlled from an adjustable model.

2.1.3 Analog and discrete control systems using simple strap-down inertial navigation systems (SINS)

The direct control method that laid a foundation for the whole series of control systems, as mentioned above, had a number of essential drawbacks. Among these are:

- sensitivity to noise, short-time ingress of "foreign" objects in the field of view of sensors, dynamic instability of sensors;
- high requirements for linearity characteristics of angular sensors;
- complexity at multiple jumps from one base to another because of a lack of "memory";
- impossibility to maintain permanent attitude on shadowed orbits (recessing stars, Sun).

The way out of this situation was to abandon the idea of direct control via actuators and introduce a gyroscopic system as a part of a control system to be adjusted through the exterior information sources and plot a specified reference coordinate system aboard a spacecraft.

An increased mass peculiar to complex gyroscopic systems forced to give up the use of a gimbal suspension and turn to design "loose" (no gimbal) strap-down inertial navigation systems.

For fair, such a decision was attended with a certain technical risk because of lacking any experience in operating the SINS systems. The decision was made under authority of the following considerations:

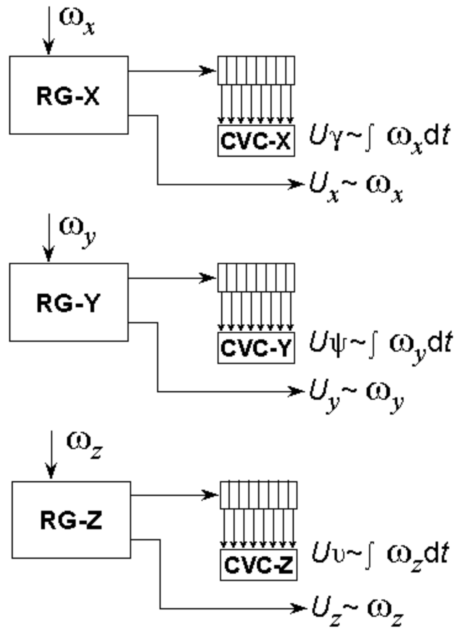
- A large duration of control processes acceptable for space objects brought down difficulties associated with the presence of high-frequency components in the base angular rate spectrum and gave a hope of achieving a high accuracy of the SINS systems under development.
- The chief advantage to the strap-down inertial navigation system, i.e. no gimbal suspension, was of paramount importance just for the space objects which control required execution of unlimited angular maneuvers (e.g. when changing the reference coordinate system, observation objects on the Earth or coelosphere, etc.).

The first simplified, non-adjustable version of the SINS system was employed on the Soyuz-M spacecraft.

The system is based on a gyro-electronic angular rate sensor unit that generates signals proportional to the spacecraft quasi coordinates, i.e. integrals from projections of its angular rate ω: $\int\omega_x dt$, $\int\omega_y dt$, $\int\omega_z dt$.

Owing to the control via quasi coordinates, the spacecraft programmed steering was performed only by its sequential spinning around structural axes.

The non-adjustable SINS system was integrated into the main control system during the Soyuz-Apollo program implementation. The similar SINS system was employed in one of two control systems of Soyuz T and Soyuz TM spacecraft. (Fig. 2)



RG – Rate Gyro
CVC – Code-To-Voltage Converter

Fig. 2. Soyuz-M strap-down inertial systems

Another modification to the control system containing the adjustable SINS was brought about as a control system Cascad installed on Space Stations Salyut.

The feature peculiar to the Cascad system was the abandoned pulse linearization principle in the control of actuators and its replacement with a discrete (threshold) control method that turned to be rather feasible owing exclusively to the SINS system.

The threshold control method accepted to control actuators made for a significant increase in the system efficiency and gave the ability to permanently maintain the station attitude over a long period. For the LVLH (local vertical/local horizon) frame this time made several months.

2.1.4 Digital two-circuit control systems based on adjustable SINS systems

The next fundamental (qualitative) step in evolution of control systems was the development of SINS systems based on an onboard computer.

This step had become possible owing to progress in the onboard computer technology.

The digital SINS systems and control systems derived from them offer a number of important features due to their versatility and wide functional capabilities of the computer system. Among these are:

1) possible and easy interfacing of systems with the exterior data sensors of different types, e.g. with sensors providing data on the object attitude in different frames;
2) possible simultaneous operation with a redundant number of exterior data sensors and inertial (gyro) sensors;
3) efficient use (through mathematical processing) of redundant information to improve accuracy and reliability of the systems;
4) simultaneous plotting and keeping in memory of several coordinate frames (frames referenced to the Earth, Sun, specified areas of coelosphere, other spacecraft, etc.);
5) in consequence with the aforesaid, execution of complex programs requiring a subsequent attitude control of a spacecraft in different coordinate systems.

Due regard should be given to the future development of systems including SINS. With the proper exterior information sources involved, such systems would permit to solve, besides attitude control, such tasks as in-orbit autonomous navigation, maneuver and orbit re-boost control, rendezvous control, and finally, de-orbiting and motion control in the Earth's atmosphere. In other words, the digital SINS system provided with rather high-capacity computing facilities would serve as a basis for an integrated system capable to solve all navigation and motion control tasks.

In parallel, up to 6 frames are plotted that enables the dynamic loop to operate relative to any of them.

Fig. 3 shows the Yamal-100 motion control system block-diagram.

The SINS system adjusted from the attitude control sensors forms the first (kinematic) loop, i.e. the object status model. The second loop (stabilization loop) controls the object attitude relative to this model. The width-pulsed modulation of commands

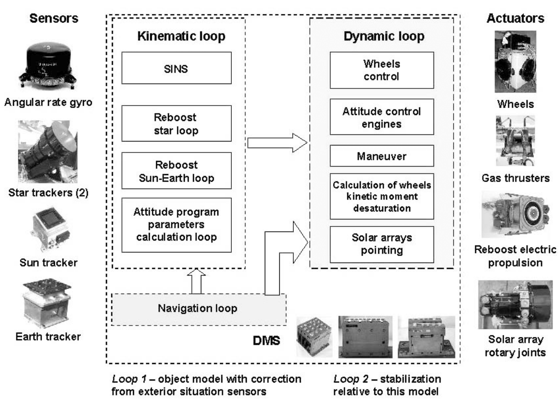


Fig. 3. SC Yamal-100 Guidance, Navigation, and Control block-diagram

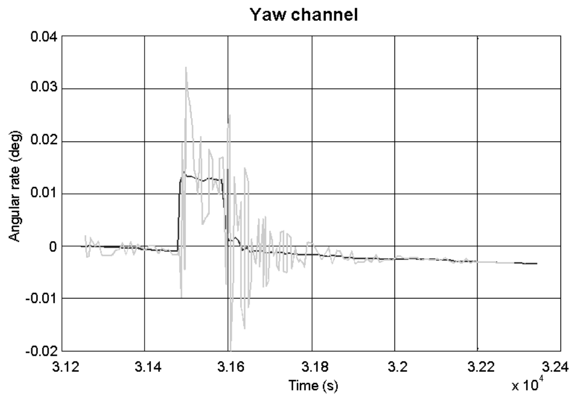


Fig. 4. Effect of spacecraft elastic oscillation angular rate

served out to the actuators is executed in the onboard digital computer.

The attitude control closed loop includes an elastic oscillation adaptive filter with an adjustable onboard model of the controlled object. The telemetry information processing results show (Fig. 4) that the angle velocity of other oscillations is greatly reduced owing to the filter, and thereby a high accuracy of the spacecraft attitude control is achieved. Similar filters are used on all current-technology spacecraft developed at RSC Energia.

2.2 *AUTOMATIC RENDEZVOUS AND DOCKING SYSTEMS*

2.2.1 Automatic rendezvous

The Soviet artificial Earth satellites Kosmos-186 and Kosmos-188 (automatic Soyuz spacecraft) were the first ever to perform the automatic docking in space on November 1, 1967.

In developing the rendezvous control system, the major challenge is to minimize the propellant consumption by all thrusters involved in this process.

From this point of view, the best rendezvous algorithms are complex, because for their execution it is necessary, in one way or another, to operate with three coordinate frames simultaneously.

As a matter of fact, the gravity acceleration components are defined in the LVLV coordinate system. The relative motion parameters of two objects can be measured from the chaser spacecraft only in the beam coordinate system which one axis is permanently pointing to a "target". And, finally, all actuators and the docking assembly itself are rigidly joined with the object structure, i.e. stay within the body-axis coordinate system.

Therefore, to optimize rendezvous algorithms, high-capacity computers were needed, and such computers were not available at the time of developing the first rendezvous control system. So, the rendezvous procedure that had been proposed and developed for the Soyuz spacecraft ensured the angular rate of centerline $\mathbf{\Omega}$ to be maintained within specified tolerances and a rendezvous range rate to be changed according to a specified law. Such a procedure was conventionally called as a "proportional rendezvous procedure". To perform this procedure on the chaser spacecraft, data on a range between the spacecraft ρ, rendezvous range rate $\dot{\rho}$, centerline angular rate, more precisely, its two components, ω_η and ω_ζ, were required, along with the angles of the spacecraft rotation relative to the centerline.

Having regard to the structural features of spacecraft, thrust limitations of the main and small engines, and a necessity to specify a relative position of spacecraft before docking, the whole process of automatic docking was split in two phases: far range rendezvous (23 km – 400 m) and approach (from 400 m to 0).

In the far range rendezvous phase the control system had to point a thrust force vector of the orbital maneuvering engine within a guidance plane, i.e. the plane passing across the centerline and relative velocity. (Fig. 5)

On the chaser spacecraft the guidance plane is constructed by the spacecraft spinning around axis 0_ξ unless component ω_η measured by the guidance equipment equals zero. At that moment, component ω_ζ will be equal to the complete vector of centerline angular rate and the engine thrust vector will appear in the guidance plane.

Next, it could be possible to align the engine thrust vector with the relative motion velocity vector (with known components of rendezvous range rate).

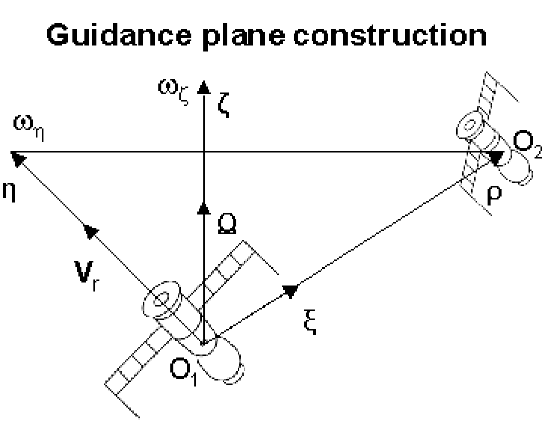


Fig. 5. Automatic rendezvous

Overall, it was a success to develop a system being moderate in requirements for accuracy of measuring relative motion parameters.

The first rendezvous control system has been successfully employed on Soyuz and Progress spacecraft over 20 years. It enabled to dock 65 times in space.

At the same time, the system suffered from the low efficiency (5-6 m/s per kilometer), a large number of propulsion unit firings in the rendezvous phase, difficult agreement between the far range guidance control laws and approach phase, and other shortcomings characteristic of this system. All this motivated further development efforts.

The development of onboard digital computers and high-sensitive angular rate sensors that made a basis for the strap-down inertial system served as a precondition for a new system development.

The new rendezvous system was first used on spacecraft Soyuz T.

The change from the direct control relying on measurement data to the control based on the spacecraft model corrected from the measurements and executed in the onboard digital computer was a radical departure from the previous system.

To make the navigation parameters more accurate, the onboard motion model, just as the whole SINS, is corrected in flight relying on data from inertial sensors and relative motion parameters measured by the radio system.

In designing, the most acceptable solution to the rendezvous control synthesis task was to split the control algorithm into a number of independent blocks-procedures capable to solve separate tasks. Among these, the trajectory guidance, navigation, and control tasks are considered to be the most important.

To make the chaser spacecraft state vector more accurate, the Lewinberger's dynamic filter was synthesized in the control system. The system ensures the filtration process stability, estimation algorithm protection against interference, and adequacy monitoring of received data.

The rendezvous process is managed in the following way. The value and direction of the required corrective pulse are obtained from the two-burn transfer formulas. The estimated rendezvous time τ is also calculated from the expected minimum propellant consumption in the given span of time.

The reboost solution is made relying on a certain logic.

As a whole, the propellant consumption for the autonomous rendezvous operation with Soyuz T is a factor of 2.5 below that of Progress spacecraft and a number of corrections is reduced by 3.5. A spread of the time moment of transfer to approach is about the same.

2.2.2 Automatic approach

As mentioned above, the approach phase starts from a range of 300–400 m and follows the proportional rendezvous process. The chaser spacecraft motion is ensured with a relatively low-thrust ("coordinate") engines and, actually, is translational that makes the docking assemblies to align.

When the system is operating at distances comparable to geometrical sizes of the spacecraft, it exhibits a number of features associated with an effect of parallaxes to measurements of relative motion parameters. In this situation, it should not be any longer considered that the measurement are made relative to the centerline.

It can be demonstrated that measurements of angles at small distances are entailed with both a change of a scale and additional errors from the passive spacecraft pitch, yaw, and mutual roll orientation.

When a range is measured in the radio system with an active response, essentially, a half-sum of ranges from the chaser spacecraft emitter to the passive spacecraft receiver and from the passive spacecraft emitter to the chaser spacecraft receiver is measured.

Similar effects cause additional errors in measurements of the centerline angular rate, as well.

All these errors lead to the undesirable feedback and interference in the control system channels.

The chaser spacecraft control at small distances becomes increasingly complicated when passive spacecraft are large orbital stations not capable of performing angular maneuvers to make the rendezvous process easier. In such cases, the chaser spacecraft, after it has approached the target spacecraft, shall fly it around.

To automatically fly around the passive spacecraft is far less easy comparing to the approach and docking process. Without dwelling on details of this task solution, it is as well to note that the automatic flyby (the first in the history of space technology) has been performed by Soyuz TM spacecraft. It flown around Space Station Mir.

3. SENSOR ATTITUDE CONTROL SYSTEM DEVELOPMENT

3.1 AUTOMATIC CONTROL SYSTEM OPTICAL SENSORS

The first optical sensors developed for spacecraft motion control systems in the USSR were static, i.e. without movable mechanical parts, of a null-indicator type, with an error making units of degrees (Kosmos, Voskhod spacecraft). To meet the challenge of spacecraft increased accuracy control, the Sun and planetary (infrared plotters of infrared vertical) trackers with a movable line of sight or mechanical scanning have been developed and successfully operated on dozens of spacecraft in our country. For example, both the star coordinates plotter installed on Mir and the astro-corrector on Space Shuttle are first pointed to a star domain containing the specified bright star by means of a 2DOF (degree of freedom) gimbal. Once the specified star is captured, on signals from the photo receivers, the gimbal motors point the instrument axis to the star and coordinates of stars are determined from the angular sensors located on the gimbal axes.

Scanning with instruments to define the Earth center direction coordinates of a secant type (vertical to International Space Station, vertical of NEC Company) is done through mechanical swinging of a mirror on a ball-bearing or torsion gimbal.

The horizon coordinates are determined from a position of the mirror that deviates the line of sight at the moment when the Earth-space boundary is crossed in the IR band of spectrum.

The experience gained from operating such sensors show that a significant part of the sensor functional failures in durable operation owes to mechanical units (e.g. Sun trackers on Space Station Mir).

Once the mosaic type photo receivers (bars, matrices) with a "rigid" geometrical scheme based on microelectronics technology have become available, that opportunity was taken to develop high-accuracy, static-type instruments measuring coordinates of starts within the angular field of vision.

In the 80s of the past century, static, high-accuracy sensors have been developed to determine coordinates of stars in tens of degrees in the angular sight of vision. The instruments have been successfully tested on Space Station Mir in a period of 1989-2000. Photosensitive CCD (charge-coupled device) instruments with a single element size of 18x24 μm and the total number of elements in the matrix about 250000 were used in the capacity of optical emission receivers. The measurement accuracy of higher than 10 ang. sec. has been achieved in the simulated coordinate systems. Instruments of essentially lower mass and capable to identify any sectors of starry sky have been developed for spacecraft Yamal-100 and Yamal-200. Similar instruments are under development and modification in Germany (Optotronics) France (Sodern), Japan (NEC), and some American companies.

The Sun coordinates plotters and planetary trackers operating within the capture limits and based on bars of CCD devices are employed both in Russia and abroad. Burns and Roe (USA) and Sodern (France) have developed efficient sensors to plot the planet center coordinates in IR (14-16 μm) range.

The static sensors plotting coordinates of the Sun, starts, and planets will continue to replace instruments using mechanical scanning. Such sensors will be also used in the spacecraft rendezvous system for relative navigation. It is of particular significance to incorporate static instruments in control systems of spacecraft designed for a 15-year service life and beyond. Mass is sufficiently saved by replacing the optical-mechanical instruments with optical electronics, thus giving the ability to increase a spacecraft payload mass, improve its configuration, etc. The experience gained by RSC Energia from operating spacecraft Yamal-100 and International Space Station Service Module enunciates the advisability of such a change.

3.1.1 Angular rate vector gyroscopic meters

Float gyroscopes making a basis for the SINS systems of the highest accuracy had come into the most widespread use among the gyroscopic sensitive elements.

The development and modification of float gyroscopes had been under way for about 40 years and resulted in the theoretically feasible capabilities owing to a number of innovative technologies, such as: magnetic centering of a float gimbal, replacement of the major axis bearings with gas-dynamic bearings, application of special materials and technologies.

The history of using by RSC Energia of instruments for SINS based on float gyroscopes is shown in Table 2.

In addition, the gyroscope high accuracy is obtained at the expense of its active two-level thermal conditioning system with an error of up to 0.1°C throughout the ambient temperature variation range.

Table 1. Angular rate meters for SINS

Parameters \ Instruments	ARS-1	OPT	Omega	Rate gyro Yamal	Rate gyro for ERS
1. Range of measured angular rates, °/s	3	10	3	0,4	2,5
2. Number of measuring channels	3	3	3	4	4
3. Zero shift, °/h	2,16	1,08	0,03	0,001	0,001
4. Output information pulse factor, ang. sec.	6,3	6,3	1,8	0,04	0,04-0,01
5. Consumed power, W	50	40	150	75	80
6. Instrument mass, kg	23	13	32	12	12,5
7. Service life, h	1000	2000	20000	90000	90000

The gyroscopes using new physical concepts are under development over many years. With respect to the principle of operation and potential capabilities, laser gyroscopes and fiber-optics gyroscopes are just perfect for the SINS systems, specifically, in active phases (under conditions of large angular rates and accelerations). The aforesaid gyroscopes, as opposed to the electromechanical ones, are theoretically free of errors caused by acceleration and, besides, have a potentially higher scale parameter that is of particular importance for SINS operating in active phases.

Intensive work on the development of micromechanical gyroscopes had been under way over the past years in Russia and the West countries. Their development is based on the electronics industry technology. An inertial measuring unit being developed will be sized to a microcircuit.

4. DEVELOPMENT OF COMPUTERS FOR SPACECRAFT CONTROL SYSTEMS

For more than thirty years in Russia, as well as in other countries, computers have been used to develop the on-board control systems and ground test facilities for the unmanned and manned spacecraft.

The decision to use computers aboard manned space vehicles was made once and for all in 1970, when design efforts based on digital control systems became irreversible. A computer system Argon-16 was installed on the Soyuz-T space vehicle for the first time. It had modest processing capabilities that enabled to solve complex technical tasks: attitude control; orbit reboost; guidance and navigation; rendezvous and docking.

All the tasks were solved by means of the following processing resources:

- about 200 thousand operations per second processor productivity (0.05 MiPS);
- about 64 Kbytes memory size;

- the programs have been written in machine codes, later the autocode was developed by RSC Energia.

Digital computers permitted to change to the control systems based on the adjustable strap-down inertial navigation systems and dynamic filters. Owing to the vehicle motion simulation, the diagnostic and adaptation capabilities had been developed for the control, which in turn essentially improved spacecraft performance.

The orbital station Mir started to operate in February 1986 after the launch of its core module. The core module had an onboard computer system based on digital computers Argon-16Б and Salyut-5Б.

Initially, the system design assumed its evolution as a network structure distributed among several modules and built-up as the station configuration was incremented.

By the moment of Mir de-orbiting, it had the following configuration:

- the network combining 5 onboard computers via an interface;
- total memory size of more than 1 Mbytes;
- Salyut-5B processing productivity was approximately 490 thousand operations per sec;
- about 20 portable Laptops to control scientific experiments and process scientific data.

Work on the orbital vehicle Buran was held simultaneously with the orbital station Mir development. The control system prime designer managed to construct the most powerful for the moment onboard computer system Biser-4 capable to execute 1 million operations per second, thus giving the ability to solve the orbiter control tasks with the memory size beyond 2 Mbytes.

The onboard software architecture included four levels of control. These are:

- spacecraft mission program control level;
- spacecraft flight phase control level (reference flight operations and control procedures);
- control of functionally interfacing systems (integrated control algorithms);
- control of dedicated systems and equipment (individual directive logic).

The software developed for the Buran vehicle was remarkable in the following features:

- adaptation and teaching;
- deep structuring of decision-making algorithms for onboard systems control provided the maximum number of algorithms to implement for solving each task;
- each higher level solved its own target task taking into account the possibility of control process options at the lower hierarchical level;
- the hardware health status monitoring and diagnostics were structurally brought in an independent functional system providing the algorithms with not only current data, but also recommendations on what control option is better to select.

The Yamal-100 and Yamal-200 computer systems are designed of a Host Computer and a number of matching units. The Yamal spacecraft computer system architecture follows a bus-module principle and includes:

- three Host Computers based on microprocessor M80C186EB;
- three matching units УС-11 on Yamal-100 and УС-14 – on Yamal-200 to control onboard systems;
- three matching units УС-12 on Yamal-1 – and УС-17 on Yamal-200 to acquire and process telemetry parameters.

The Host Computer speed is 3 million operations per second (that corresponds to ≈0.5 MiPS).

The RAM/ROM capacity – 256/768/256 Kbytes.
The number of control commands and inputs – ≈500.
The number of processed input parameters – about 1200.

All users of the onboard cable network are connected to the multiple exchange channel universal interface made in accordance with MIL STD 1558B.

Currently, the Onboard Computer System which characteristics are presented in Table 2 is being developed.

Table 2. Comparative characteristics of domestic onboard digital computers

Name	Capacity, MIPS	PAM capacity, Kb	ROM capacity, Kb	Mass, kg
Argon-16, Soyuz, Progress	0,05	2	64	80
Salyut-5Б, Mir station	0,1	16	128	72
Yamal-100 and Yamal-200 onboard digital computer	0,5	256	768	36
Victoria	6	2048	2048	30

5. ACTUATORS

The first spacecraft attitude was controlled by jet engines using compressed gas. These engines providing a thrust of up to 1 kg were used on the Vostok, Voskhod, and automated spacecraft, such as Mars and Venera. As moments of inertia were increasing, one-component and two-component liquid propellant rocket engines generating a thrust from one to 10 kg were introduced. Spacecraft Soyuz was the first to use such engines.

The electric propulsion (Fig. 6) using xenon occupies a rather important place in the development of rocket engines.

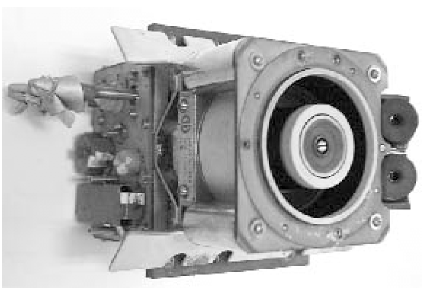
Work on the electric propulsion was initiated in the USSR late in 50s and the first electric propulsion engine had been proposed as far back as in early 30s.

In the second half of the passed century the theoretical basis for the electric propulsion operation has been worked out in our country and electric propulsion engines of several types have been developed.

Currently, the started elaboration of electric propulsion is rather sufficient to enable the solution of different tasks and, first of all, to develop electric propulsion units for communications satellites of a long in-orbit life.

The electric propulsion is widely used by RSC Energia for the platform Victoria that has made a basis for the development of communications spacecraft Yamal being currently operated.

The electric propulsion offers a high advantage and in a number of cases is the only one solution for spacecraft designed for the durable in-orbit



- Specific impulse, m/s 14100
- Nominal thrust, N 0,038
- Consumed power, W 680
- Mass, kg 3,2
- Designer SDO Fakel

Fig. 6. Electric propulsion

functionality, transportation of heavy payloads from the low Earth to high orbits, as well as missions to other planets, e.g. Mars. This advantage of electric propulsion gained over other rocket engines is achieved owing to a high-speed efflux that is an order of magnitude and more exceeding this value, as compared to other types of space rocket engines. However and to large regret, the electric propulsion can operate only in vacuum.

As the spacecraft size increases, so too do working medium G consumption needed to maintain the spacecraft attitude.

Starting from certain values of spacecraft masses, the use of jet engines in the attitude maintenance mode is becoming impossible.

With inertial actuators requiring the active mass consumption only for desaturation modes, the G value is reduced by the order of 1 or 2. And, when the inertial actuators are used in combination with the gravitational or electromagnetic desaturation that is possible with the onboard digital computer, a practically non-propulsion attitude control system could be obtained.

The use of electric motors-wheels for attitude control of space stations like Salyut and Mir is limited by their power consumption due to a large amplitude of rotor kinetic energy fluctuations. This limitation is of paramount importance and cannot be essentially reduced by means of engineering tools.

The power consumption problem could be made much less critical, if gyroscopic momentum stabilizers were used instead of electric motors-wheels.

In selecting a momentum gyro stabilizer, a criterion $æ = H_0 / \Sigma_i \Gamma_i$ was used, where H_0 – radius of sphere inscribed in area S of gyro system H kinetic moment vector variation, Γ_i – modules of intrinsic kinetic moments $\mathbf{\Gamma}_i$ of all its rotors.

With reference to Space Station Mir, four potential types of momentum gyro stabilizers have been studied (see Fig. 7).

By evidence of the obtained results, it was decided to select the momentum gyro stabilizer of type 2 that had never been employed for the purpose of spacecraft attitude control. The momentum gyrostabilizers of this type were called “gyrodynes”.

The choice was based on the following considerations. As regards the æ characteristic, the gyrodynes rank only 1,27 below the 3DOF gyroscopes. At the same time, their weight

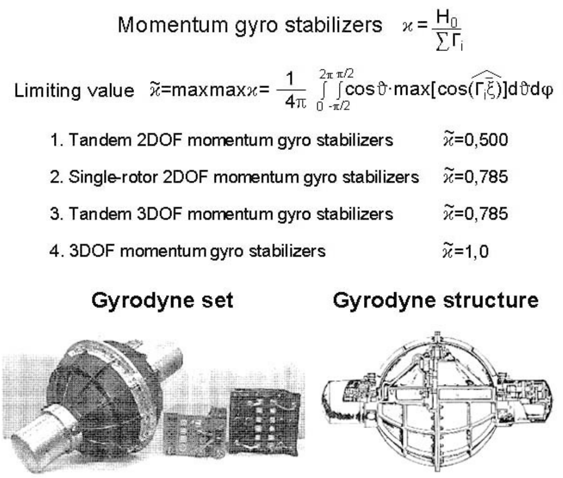


Fig. 7. Selection criterion

advantage, as compared to the 3DOF gyroscopes, is shown in the design simplicity – and thus their reliability. A critical importance of the latter is recognized for the man-tended, long-life orbital stations.

To obtain the highest possible value of æ approximating $\tilde{æ} = 0.785$ was the requirement that set the course for designing the gyrodyne system.

In the version providing the maximum value of æ, the axes of precession η_i of all gyrodynes shall make angles between each other to be different from 0 and $\pi/2$. With $n = 6$, the best result could be achieved when axes η_i are located along directions of normals to six unparalleled faces of a dodecahedron that makes possible to obtain æ = 0.714. It should be noted for comparison, that typical æ values of the traditional momentum gyro stabilization systems are not exceeding 0.033.

Since no sufficiently similar analogs to gyrodynes and gyrodyne systems exist both in the ground or space technology, a number of engineering problems have to be solved for their development. It is as well to note the first of those:

- The development of a brushless (magnetic) feed rotor gimbal that makes a service life of gyromotors practically unlimited.
- The development of reliable conductors on gimbal axes of gyro units having the unlimited range of rotation.

The problems have been successfully solved at the VNIIEM under the direction of academician N.N. Sheremetyevsky.

Among the fundamental problems, the central place is allotted to the control law for the strongly redundant gyroscopic system.

The major component of control moment **M** of the gyrodyne system is expressed as: $\mathbf{M} = \mathbf{Iu}$, where $\mathbf{u} = \boldsymbol{\varepsilon}$ (ε_1, ε_2, …, ε_n) – vector of control inputs and **I** (ε) – Jyakobi matrix.

Owing to the system redundancy, there is an infinite number of controls $\mathbf{u} = \dot{\boldsymbol{\varepsilon}}$ to obtain the needed value $\mathbf{M}^*$ of moment **M**.

When vector **H** falls into singular point $\mathbf{H}^*$ and $\boldsymbol{\varepsilon} = \boldsymbol{\varepsilon}^*$, the gyrodyne system will cease to be capable of generating a control input in one of three mutually perpendicular directions, and consequently, loose its functionality.

The singular points $\mathbf{H}^*$ are grouped into singular surfaces h^*.

The gyrodynes shall be controlled so that, with any required program **H**(t) for changing vector **H**, the crossover of all these surfaces is easily achieved. That is, angles ε_i are controlled in accordance with the law ensuring the bypass of singular hypersurfaces in space of vector $\boldsymbol{\varepsilon}$ (ε_1, ε_2, …, ε_n), which are similar to singular surfaces in 3D space **H**.

In parallel, it is necessary to support the gyroscopic system control torque to approach the maximum possible value with the current **H** value. Both tasks, if not in contradiction with each other, had been solved by introducing an additional component in the control law.

The possibility to express such a function is illustrated by the following example. Assume that:

$$\mu(\varepsilon) = \det \mathbf{II}^{\tau} = \sum_{l \le i \le j \le k \le n} \Delta^2_{i,j,k} ,$$

where $\Delta_{i,j,k}$ – minors of matrix $\mathbf{I}(\boldsymbol{\varepsilon})$. The equalities $\varepsilon = \varepsilon^*$ corresponding to the inequality rank **I**<3 are existing only when $\mu = 0$. On the other hand, minors $\Delta_{i,j,k}$ are equal in module to volumes of parallelepipeds drawn, as on ribs, on triples of vectors $\mathbf{m}_i$ of control torques of certain gyrodynes. Practically, it turns to be possible to use more simple functions $\mu(\boldsymbol{\varepsilon})$.

Let us draw a vector of control torques **u**.

The task solution is expressed as:

$$\mathbf{u} = \mathbf{u}_1 + K\mathbf{u}_2,$$

where $\mathbf{u}_1 = -\mathbf{I}^{\tau}\mathbf{D}^{-1}\mathbf{M}^*, \mathbf{D} = \mathbf{II}^{\tau}$,
and $\mathbf{u}_2 = \left(\mathbf{E}_n - \mathbf{I}^{\tau}\mathbf{D}^{-1}\mathbf{I}\right)\mathbf{grad}\mu$.

In this case, the component u_1 makes for generation of the required control torque $\mathbf{M} = \mathbf{M}^*$ and component u_2 represents the desired law of the gyroscopic system adjustment (adaptation).

The described system has permitted the 15-year operation of Space Station Mir and its prototypes will make a basis for automatic spacecraft.

6. SUMMARY

Evolution of spacecraft control systems relies on the experience gained from using and advancing new ideas and challenges. At the same time, each step was remarkable in great steps forward to "technical revolution", when old principles were rejected or severely revised and the system based on another concept was becoming more advanced.

In the period after launching the first satellite, at least, three generations of control systems have been changed.

For example, the direct control relay and pulse-relay control systems developed for spacecraft Luna-3, Vostok and Voskhod, and unmanned objects Zenith have been already allotted a secondary place. A leading role was taken by the analog direct control systems using pulsed linearization (Soyuz spacecraft, Space Station Salyut, L-1, L-3).

The prototypes to new, no-gimbal inertial systems appeared on spacecraft Soyuz M, Progress, and the latest modifications of Space Station Salyut. On spacecraft Soyuz T and Soyuz TM those systems had already become the primary systems and on Space Station Mir – the only systems used.

A search of physical principles acceptable for designing primary elements of systems and ensuring their required properties, accuracy, and reliability is driven by a variety of tasks to be solved by motion control systems.

In the course of these developmental efforts, the whole spectrum of electromagnetic radiation has been considered. The Earth's ionosphere, its magnetic and gravitation fields, aerodynamic and reactive forces, and gyroscopic effect have been given proper attention.

Over the past years, more than 70 optical and optoelectronic instruments have been developed and commissioned by the related organizations according to the RSC Energia specifications. Over 20 gyroscopic instruments have been developed and passed flight tests on the RSC Energia objects to meet an increasing need to improve accuracy, extend service life, and enlarge the measured angular rate spectrum.

The system efficiency, i.e. the average power and working medium consumption, is one of the major criteria governing the system performance.

Certainly, the development of efficient systems is not consistent with a necessity of fast maneuvers and durable attitudes. The first systems developed for spacecraft Vostok and Voskhod have been rather inefficient. Their usage determined the short-term active life of the spacecraft and compelled to solve only priority tasks.

By introducing pulsed linearization in the control systems of spacecraft Soyuz and Space Station Salyut, the propellant consumption had been greatly reduced, however it appeared to be insufficient for the durable attitude control. The next system (Cascad) employed on Salyut, with all other things equal, was of higher efficiency. However, the increased requirements for the stabilization accuracy negated this advantage.

Table 3. Operation Accuracy and Service Life Requirements to Orbital Stations

Years	Accuracy	Service life (hours)
1960	1-2°	100
1970	10′	500
1980	3′	$5 \cdot 10^3$
1990	6″	$2 \cdot 10^4$
2000	0,1″	$2 \cdot 10^5$

With the further enhancement of requirements for the stabilization accuracy of space stations considered, all that permits to feel certain about a crisis of control systems using jet engines in the capacity of actuators.

It has come evident that further development must be devoted to non-propulsion systems, and in the first instance, to the momentum gyro stabilization in combination with the gravitational or magnetic desaturation.

The gyro control system developed for Space Station Mir had become the first domestic system of this type.

The ad hoc investigations carried out in anticipation of further advances showed the non-propulsion systems to be efficient on orbital stations outperforming Space Station Mir several times in mass.

With further enlargement of space stations, it will become impossible to use such systems as well, due to dimensions, mass, and mainly, power consumption of gyrodynes. In this case, the passive (gravitational) stabilization combined with the station oscillation damping via the comparatively low-power gyrodynes would most probably be the most efficient. One of advantages to this concept is a permanent, and consequently, smooth control in the neighborhood of zero, that is of a particular significance for large flexible structures.

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