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MODELING THE FUNDAMENTAL CHARACTERISTICS AND PROCESSES OF SPACECRAFT FUNCTIONING

V. I. Bazhenov, M. I. Osin and Yu. V. Zakharov

Translation of Modelirovaniye osnovnykh kharakteristik i protsessov funktsionirovaniya kosmicheskikh apparatov, Mashinostroyeniye Press, Moscow, 1985, 240 pages

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Reviewed by Doctor of Technical Sciences V. E. Ishevskiy

Bazhenov, V. I. et al.,
"Modeling the Fundamental Characteristics and Processes of the Spacecraft Functioning," Mashinostroenie Press, Moscow, 1985, 240 pages.

The book considers the fundamental aspects of modeling of the spacecraft characteristics by using the computer means. Particular attention is devoted to the design studies, the description of the physical appearance of the spacecraft and simulated modeling of the spacecraft systems. The fundamental questions of organizing the on-the-ground spacecraft testing and the methods of mathematical modeling are being presented. Also, brief information about the experimental test prototypes and some test flight results are being presented.

The book aims at design engineers, the spacecraft designers and test personnel.

"What has appeared to be unattainable for ages, what even yesterday appeared to be a daring dream, today becomes a reality and tomorrow - the reality will be realized. There are no barriers to human thought!"

S. P. Korolev

Preface /3

Modeling as a method of scientific study is being used extensively in engineering activities, embracing all stages of design and refinement of spacecraft design, for interplanetary flights, for functional work on the planets and in space, in the vicinity of the planets.

From the very start of the project, in the very beginning of estimates and evaluations, the approximate analytical and statistical relationships are being utilized, which are to model the relationships between the spacecraft parameters. All subsequent phases of design are supported by a variety of modeling rules, although the term itself is only seldom used today, by experts in gas dynamics, ballistics, control, durability and tests of all spacecraft systems.

For quite some time the concept of "modeling" was associated with the investigation of dynamic properties by using the analog-to-digital machines or by testing the prototypes in aerodynamic tunnels with subsequent flight experiments. The development of powerful computers made it possible to conduct high volume calculations of the parameters of motion and heat exchange in the presence of different environmental conditions, laying the foundation to a new trend in modeling the processes of spacecraft functioning.

The further expansion of modeling work is associated with the simulation studies of the spacecraft behavior while in use. And finally the new models have been developed recently which describe the functioning of the onboard systems in the spacecraft and the informational models which reflect the design process of the device as a whole.

With the appearance of computer graphics the new methods are being intensively developed for mathematical modeling of geometric configurations which include the structural synthesis, methods for the description of the spacecraft exterior surfaces, the programmed methods to depict and memorize the graphic design components, the schematic diagrams of propulsion units, and other components.

Today, it is possible to develop detailed dynamic models which will reflect the loads and interaction between components of the design and the geometric models which replace the drawings of components and assembly units.

In parallel with the mathematical methods of modeling, also the physical modeling is being extensively developed by using the experimental assemblies which are to simulate the environmental effects, to study the repsonses of onboard systems, of control systems, various mechanisms and design components as they are being exposed to the operations on the spacecraft (including the situations which were not planned).

The value of calculations and of such analytical and physical /4 methods of modeling is not so much to resolve the existing questions, as to enrich the data base, with the ability to consider some new problems. In such a way, these methods which were developed as the analytical instruments for already finished designs become the means for the synthesis of other design schematics and operational modes, the means for further research studies.

The modeling tasks are quite concrete, they are specific and are narrowed down to a specific type of airborne device and the specific nature of its functioning. However, there are some general fundamental principles and rules of modeling, of which the reader should be cognizant. This book describes the major trends in modeling during the design process and during the spacecraft operational refinement on the ground. The book includes some explanatory examples of the design problems, as well as experimental studies.

The material in this book utilizes many years of experimental work of the authors, who are familiar with design work and with testing on the ground of airborne devices of various types.

The authors express their gratitude to the doctors of technical sciences, V. E. Ishevskiy and Ts. V. Solov'ev, for their valuable suggestions and comments, made by them during the review of this manuscript.

The book may turn out to be useful for the design specialists, for the personnel who are involved in spacecraft ground tests, the individuals who are involved in developing the methods of mathematical test planning and development of experimental assemblies, and also to teachers, graduate students and students in the appropriate fields.

The authors will be grateful for all comments and responses of the readers in regard to this book which may be addressed to the Mashinostroenie Publishing House.

MODELING THE FUNDAMENTAL CHARACTERISTICS AND PROCESSES OF THE SPACECRAFT FUNCTIONING

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PART I /5*

METHODS OF MODELING IN DETERMINING THE SHAPE OF SPACECRAFT AND IN DESIGNING VARIOUS ASSEMBLIES

The calculations and analytical modeling is subjugated to the design purposes and is an analytical instrument, while sometimes it is being used for the synthesis of design decision-making. A number of models which describe the operational modes, streamlining and heat exchange are well known, and have been considered in numerous studies [8, 27, 33, 50]. Less known are the formalized models of spacecraft system interaction and the methods to model some uncertainty factors which are inherent in the initial stages of design.

In order to concentrate the reader's attention on the problems of design initiation, the first part of this book will consider some important models for the design decision-making, to synthesize the spacecraft structural composition which so far has only been poorly investigated. In parallel with such analysis of procedural decision-making, we shall describe the mathematical and data models, defining the configurations of the assemblies, subassemblies and specific design components for such projected spacecraft.

Some aspects of such approach have been reflected in the studies [45, 48], which have investigated in detail some specific features of the initial stages of spacecraft design, including such questions of analysis and synthesis of new design decisions which include the uncertainty factor from the very start.

CHAPTER 1.

MODEL OF A TYPICAL DESIGN PROJECT START-UP IN THE DESIGN BUREAU DEVELOPING THE SPACECRAFT

Before we proceed with the description of computational models which are used by designers, let us consider the activities of the designers themselves and let us try to consider some of the

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*Number in the margin indicate pagination in the foreign text.

procedural relationships involved in the design start-up.

The investigation of these procedures, with the study of data flow between different departments and design bureau experts, with the analysis of suggestions from specific participants of the collective project, may be assumed to be the basis of the technological process design and of the operational model which will describe the search and selection of the spacecraft external features.

Such a model includes the distribution of responsibilities between the participants and working groups, the list of stages, aimed at the design project refinement, from the first conceptual idea to the point when the design documentation has been developed, the description of the starting data and the finalized documentation, containing also the formal-mathematical presentation of the problem as to the general physical appearance of the spacecraft, the type and nature of the data which is to be handled as per request and during the feedback to all individuals who are to develop and make the design decisions.

Let us immediately state that a thorough model of such typical start-up design process cannot be developed at this time. The selection of the general features of various devices is too specific, the activity of spacecraft designers who utilize their past experience, supported by intuition, is too complex and too multifaceted. It takes a certain skill to select the unique design systems, taking into account the contradictory conclusions of experts in regard to different aspects of the project.

It is known that the proper selection of a complex spacecraft is impossible without errors and some non-optimal decisions, and the dielectics of the design search process consists in the refinement and corrections of the initial concept. How is one to include into the model of the design process a number of unavoidable errors? How is one to take into account the culminating moments of "clairvoyance" which cannot be explained on the basis of the gathered data? How is one to formulate the requirements imposed on the project, the target and the end-goal of the design if the hypothetically ideal design solution is subject to a strong and devastating criticism during the very first discussions by the experts, in regard to the durability, ballistics, gas dynamics, control, reliability and efficiency of the spacecraft.

One cannot always explain the fact that in designing the spacecraft and striving for the rational, or on the basis of some criteria, optimal shape, composition, structure and operational modes of motion, the designers are not always able to explain how they do the job. Precisely for that reason, sometimes the very problem of the design of an ideal hypothetical system of the "computer designed" spacecraft turns out in practice to be consideraby more complex than the design of the spacecraft itself.

It was frequently attempted to formulate mathematically the design problems, but no practical and applicable results have been attained. This is explained by the abundance of uncertainties and the continuous improvements in the course of design, by the presence of contradictory criteria of different levels of importance, the quantitative and qualitative requirements in the presence of inaccuracies in the computed models and the not quite adequate real functioning processes, inherent in the spacecraft.

The operational model of the design process includes a sequence of typical activities which results in the development of first the intermediate, and later of the end design results.

Let us formulate what we mean by the "design result."

In many teaching manuals the various typical problems associated with the design parameters are presented in the following manner: in having a set of starting data, to find an unambiguous answer by employing concrete computational algorithms or graphic procedures which would elucidate the values of design parameters.

However, the problem of spacecraft design is considerably more complicated because of three reasons.

The first one is the requirement to accurately refine, during the research project stage, the range of the starting data so that /7 in selecting the general appearance, in the spacecraft design, various combinations of such starting data would be taken into account. Among these factors is the very important information for the spacecraft as to the environmental conditions on the planet for which the spacecraft is designed, the efficiency factors of some new heat-shielding and design materials, of the fuels and of the starting and end conditions of flight. The designer must aim at such levels of starting data which would guarantee that the design parameters of the spacecraft selected by him would ensure the attainment of targets for which he is responsible.

The next reason why the design study is more complicated, when compared to simple calculations of a graphic problem, is that the results of the search for the best design displays some ambiguities. There is a great abundance of various design diagrams which are quite characteristic for spacecraft, and which require a thorough search for the "best" combinations of the systems onboard, with the singling out (or at least of some elucidation) among several "competing" designs.

The third reason which makes the activities of the designer more complicated is that he is insufficiently well informed and sometimes has no clear concept as to the design limitations and fundamental criteria. During the search for the best design, the designer, while learning, must comprehend what is the "best" solution, he must be able to judge properly the "price" of conceding or reducing any of the applied criteria, while debating the spacecraft general features, as he considers one rational variant after another.

To summarize, in order to attain positive results in design, it is necessary to analyze and outline the initial data which defines the flight conditions and the "environment" in which the spacecraft is to function, to formulate the basic criteria, to impose specific requirements on the project with the appropriate constraints in terms of some specific parameters and flight modes, and also to search for the design variant with the best or limiting design parameters.

The questions of basic criteria modeling are described in detail in Chapter 3, which presents the recommendations as to the proper selection of the design decisions in the presence of uncertainty.

Let us consider for the time being a simplified model of the activities of the designers, which assumes that the detailed accounting for the uncertainty factors will be carried out at the final stage of the search which will involve the comparison of several rationally selected design variants.

The design goal (the attainment of results) defines the sequential activity - an operational model in the search for the best design.

Let us describe a conditional operational model on the assumption that the activities of the designers are coordinated by one designer who is in charge of the collective group of design engineers. The complex relationships between the project designer will be considered separately, as we describe the collective group interaction, those who are involved in the design decision-making.

In the conditional operational model we shall not adhere religiously to the Unified System Design terminology, since it formulates the extended stages of design (it disregards the procedures of the design concept development) tying-in each stage to the output product (the stages of preparing technical assumptions, the design sketch, the development of technical design projects and /8
of operational projects).

Let us consider the activities of designers at the starting point, as they begin to develop the general design concept.

In designing the spacecraft the starting phase is the description of the goal-oriented purpose and the problems which have to be tackled by the future spacecraft.

At the same time it is necessary to define the available resources and design constraints (lifting capacity of the carriers, size of the streamlined parts, type and dimensions of the useful loads, operational range, the starting and end conditions of the flight).

This initial phase is sometimes called the "external" design work. Its purpose is to set up intelligently the task at hand, which would fully exhaust the definition of the assignement, to analyze the goal-oriented target, to select the typical, standard and non-standard flight programs, to describe typical useful loads and to define the major parameters of the goal-oriented systems.

The "internal" design is accomplished sequentially-parallel with respect to the "external," continuing until the general project concept is finished, with the preliminary refinement of the project. These two phases are connected by the feedback, with periodic data exchange between the individuals who formulate the problem and make the decisions. The purpose of the "internal" design is to define the general features of the spacecraft, to approve and properly document its major parameters. Conditionally, "internal" design of spacecraft of new types consists of three cycles: a broad search for optimal and pre-optimal variants, the selection of rational variants and the well-substantiated development of the project.

The responsibilities of designers at the "external" and "internal" stages of design differ and the composition of the selected designers - the participants within these cycles, is also different.

In spite of the variety of names, in regards to the specific services, the organizational forms of the design process and different titles and responsibilities of the individuals who are participating in the design within the organizations which have different structure, the essence of interaction between the designers is the same: this is a group of a collective of specialists of different profiles, formally united (or informally) by the common idea of the project start-up.

More often than not, at the starting stage the search for the best design is accomplished by a group of designers, computing experts and design engineers and also by the specialists from various specific subject matter services, which are temporarily attached to the group for decision-making in regard to the specific subject matters. The functioning of such a group, for the project start-up, comes about by taking into account the existing administrative structure of the organization. As a rule, this structure is of matrix type, and the responsibility for the obtained results in the design bureau are subdivided, on one hand between the senior project leaders responsible for different subject matters (vertical leadership) and on the other hand - between the leading individuals of working groups on specific subject matter who are involved in studies of strength, aerodynamics, dynamics, control, propulsion units, the onboard systems of various purposes (vertical coordination).

There are also some auxiliary services which help in design and which are involved in the development of documentation, data exchange, the computational and graphic calculations, planning and coordination, fulfilling all technological requirements. /9

The selection of the start-up project group is also based on the matrix principle. In charge of it is an expert who is also the representative of the design subsection. As a rule this is a group leader (leader No. 1) who is responsible for the exectution of the component diagram and the component composition for the whole design. Just like the chief or leading designer for the specific project, he is in charge of the project start-up. Sometimes he is assisted by several individuals who calculate the total mass and work on the arrangement of components. As a rule, the individuals who are in charge of the component arrangement are the people who are in close contact with him, preferably from the same design subsection, who are responsible for accomplishing the flight program (the target-oriented efficiency), determining the composition of spacecraft components and assuring the proper tie-in in the spacecraft systems on-board.

Each specific professional subunit ordinarily appoints one responsible representative who participates in the work of the group which initiates the project. He is in charge of the computational and experimental results, he evaluates the aerodynamic parameters, the data of gaseous flows and heat exchange. The determination of loads and strength, of the selected propulsion design schematics, of the correlated propulsion and power units, as well as the radio systems, the calculation of operational modes of motion and the questions of control analysis - all these are the responsibility of this representative. As a rule the interaction between the individuals, who are involved in putting all parts together with the responsible representatives of the specific subject matter subunits, takes place at the meetings when the applied calculations of the results in regard to the specific project matters are being discussed. Sometimes (very seldom) the information exchange takes place by using in the project subunit the applied programs, which were develped by the specific subject matter subunits. More often than not, the design subunit has its own program department which develops the specific combined characteristics for the size and mass. Such program complex is the basis of the Automated Design Project subsystem, which is responsible for selection of the general features of the spacecraft, its simulated operation and the target-oriented effectiveness. The users of this program complex are within the same design department and conduct the calculations as an assistance to the individual who is in charge of the arrangement of spacecraft components.

The developed, automated subsystem which is to search for the optimum spacecraft features becomes an effective instrument which provides a reliable and rapid review of each proposed concept.

Therefore, the man who is in charge of preparing and handling the project task, by using a computer, becomes the closest assistant to the man who is in charge of putting together all components of the spacecraft and may be assumed to be the leader No. 2, who is in the group of men who are in charge of the project start-up.

In the practice of design which has been developed throughout, ordinarily the designers who are handling the spacecraft components are not well versed in the computer operations, they have no experience in programming and are only slightly familiar with the theory of optimization. This is precisely the reason why the intermediates between the group leaders, who are in charge of the project start-up, and the men involved in the automated design systems, we have a group of people who are developing the computerized search for the general features which the spacecraft must display.

The department group for automated design, the leaders of which provide for the project start-up, are responsible not only for the quality of the design variants. Their functions within the group which "start-up" the project are similar to the functions of the auxiliary services within the general structure of the design bureau, in other words they are responsible for the provision of information (by using a unified data base of the project work), for the development and visual presentation by using the instrumental graphics, of the mathematical model for the configuration in question, they are also responsible for coordination and cohesive operational tie-in, which is necessary to generate the design calculations and to develop the appropriate documentation. /10

The cooperation between the design leader and the person who is in charge of the optimal dimensional parameters creates somewhat fundamentally different relationships within the project development, making it possible to consider in great detail the very first and initial concept and already on that basis, to come back and consider individually each new project concept.

There was a time when the projects of relatively simple airborne devices were designed by a few people only, but each of them exerted a creative influence on the whole project by penetrating in depth into the essence of the applied problems with which the project leader was involved.

The gradual increase in the number of designers of narrow specialization made the design control more complicated even disregarding the fact that the airborne devices became more complicated. The methods of automated design (if they are skillfully applied) make it possible, by retaining the close work of the experts involved, to centralize and ensure the whole project tie-in, in spite of the large number of designers who participate in preparing the data and who are involved in conducting the calculations and designing some applied programs. Such retreat to the centralized leadership project design, with the appropriate design decision-making, is accomplished nowadays on a new base, with the aim at methods of collective interaction.

The model of the design process which takes into account the collective interaction differs from the simplified schematic diagram of the project operational tie-in.

The major features which are introduced into the model of the design process by the factors of collective interaction are as follows:

1. The contradictions between experts of various project subject matters resuls in the need for a compromise as to the provisions of necessary strength, minimum mass, optimum carrying and braking parameters and the systems of flight power requirements and control.

2. It becomes possible to carry on with different tasks in parallel, with the total satisfaction of the requirements of different project design aspects, at separate project stages, independent of each other.

3. The groups of experts in different design subject matters are able to study in detail some problematic questions, becoming involved in the development of important design components, utilizing the thorough computational models or experimental assemblies.

4. The design of separate assemblies and units is based on the design developments which require the intuition, imagination, an approach which is not standard and not formalized, without any /11 ready-made recipes and computational models. In some organizations, the practice of competing designs embraces not only the stage of the project initiation, but also the periods of design sketching and operational drafting.

5. The need for a thorough documentation is satisfied, so as to effect information exchange between designers and consequently, it becomes possible to describe the results of any stage of the design research.

6. In the course of group discussions, it becomes possible to criticize the unreasonable design decisions, developing simultaneously the new, unusual ideas and concepts.

However, the separate decision-making of the finalized design solutions is preserved regardless how well the automated systems of group design are developed. The amount of information and objectivity in considering the existing opinions and other factors lower the technical risk, but they do not remove the responsibility of the leader of such collective interaction, they improve the requirements on management for the results of various designs and exchange of opinions in regard to different subject matters, inherent in the design project.

The inept organization of the data exchange, obtained in some of the departments, the distorted interpretation of calculated results by the experts in the adjoining operational fields, the errors while processing the computerized data, are twice as dangerous since they facilitate unreasonable decision making (or even erroneous decisions) of the finalized project solutions, introducing disharmony into the cooperation of various experts who obtain the same information generated by them, via the feedback, in an altered form, resulting in uncertainties in their own efforts and lowering the integrity of the design work leaders.

A group of project designers which functions predominantly at the beginning phase of the "internal" design work, as has been mentioned earlier, includes the leading project designer (leader No. 1) with several assistants who are to correlate the gathered data, describing the exterior, composition and internal design of the spacecraft, the leading designer who is involved in computations (leader No. 2) with the assistants for the automated optimized design search, a leader for the goal-oriented targets with the assistants who are to tie-in the flight programs, the appropriate functions and composition of the service and operational systems, the members of the groups who are involved in the dynamics (ballistics) control, the general control, gas dynamics and heat exchange, the load tolerances, the aerodynamics, power propulsion units, radio systems and other service systems.

In a number of cases, as early as the stage of the design start-up, the operational group may obtain consultations from various technologists, operational men, experts in material studies and representatives of other design subsections.

The decisions made by the project design working group are not final. They are channeled periodically to the project leaders to be corrected, who, together with the leaders of various subject matter subsections, form a team which is the decision-making group (DMG). Sometimes the DMG group includes the representatives of neighboring organizations, scientific institutes which are to review the project, the coordinators and leaders of the appropriate agencies.

Within the working phase of the "external" design, the DMG group would include the leaders of the design and scientific institutions, the representatives of the customer and also such /12 organizations and agencies which are responsible for proper operation of the designed spacecraft.

The monitoring by the DMG team of the whole course of the design is done periodically, particularly in selecting alternative variants and in resolving conflicting situations at the junction points, where the requirements of different design subgroups may be in conflict.

The basis of unavoidable conflicts and contradictions is a mutual mistrust, which is quite understandable, as to the calculations of experts in aerodynamics, ballistics, heat exchange and the spacecraft weight.

The disagreement as to the level of safety features, in regard to specific parameters (aerodynamics, fuel supplies, reasonable size of compartments, the radii of free landing maneuvering, safety factors in the calculations of strength) are discussed by the DMG team and the degree of compromise which can be attained on these questions shows how real is the design project.

One of the major tasks in the interaction between design leaders and the operational design team is to determine the reliability of presented results in regard to the selection of optimal variants with the simultaneous evaluation of accuracy of calculated models and the issuance of recommendations to the specific subgroups. All backup parameters inherent in the project must be controlled in a centralized fashion, resulting in a unified procedure which would model some data scattering of the major spacecraft parameters.

In defending their point of view, the narrow profile experts must be skillful enough to propose some cardinally new design solutions, which might be quite unexpected even for the experienced designers who, as a rule, are bound in their activities by the existing constraints and prohibitions imposed on the original design.

The critical approach of the design experts enables them to question the initial data and the opinions of the junior designers, as they proceed with the developement of the starting design concepts and select the goal-oriented targets of the future spacecraft.

The project designer, in addition to the firmness in his opinion and certainty as to the original requirements, must be tolerant with respect to new ideas, he must be prepared to sacrifice some results of the preceding poor designs and must be ready to risk, while searching for the new technological decision. He is responsible for the selection of the configuration, composition and design construction of the spacecraft.

The major responsibility of the leader No. 1 is to establish the main design task. This includes the clearly formulated requirements in regard to the spacecraft and its major systems, the systematic listing of criteria and constraints, the elucidation of parameters which may be varied which define the general appearance and operational mode of the spacecraft which is being designed.

The next responsibility of this leader is to correlate the proper design with the properly selected reasonable (and sometimes optimal) parameters which correspond to the design project, keeping it within reasonable boundaries and allowing for some risk, and finally to prepare the technical assignments and starting data for /13
the subsequent development of the design subassemblies and the spacecraft systems on board.

New, and sometimes not quite ordinary, are the functions and responsibilities of the leader No. 2, who is in charge of computations and optimized search, who utilizes the computers and instrumental graphics to review the variants with detailed calculation of the size and mass parameters of the spacecraft. He is responsible for the mathematical formulation of the optimization of shapes and trajectories, the component synthesis and structure of the spacecraft. The leader No. 2 is responsible in generating a set of applied programs and data base, the maintenance of program module interface, he is responsible for content of the structural "menu" which is to be used in the operational mode of the dialogue of the users, who are to carry out the calculations with respect to various goal-oriented subject matters.

The operational model of the design formalization of a typical spacecraft in the phase of "internal" design work includes, as a rule, the following procedures:

1. To study the target-oriented purpose and likely operational conditions of the future spacecraft. To gather the information about the environment (the planetary environment or near-planetary environment), to study the initial and end conditions of operation. To compile the first sketches of standards and emergency flight programs and of the operational modes of the target-oriented and service systems. The procedure is terminated with a critical analysis of the design project.

2. To look for and investigate poorly known physical effects and phenomena and also some unusual design solutions in terms of the thermal protection, braking and landing systems, new types of fuels and power propulsion units, promising navigation means, the means of radio communication and control, the means of thermal control and power supply systems. To refine the requirements associated with the target-oriented systems and the payload. To conduct the evaluation of ballistic computations, determining the fuel supplies, investigating the navigational questions, the correction of trajectories and radio communication at different trajectory segments. To describe the interaction with other spacecraft, other assemblies and launching systems.

3. The fine points (or refinements) of criterial base by systematic arrangement of qualitative and quantitative requirements and constraints. The selection of project parameters, by the use of

which it may be possible to attain the best or maximized results. The recalculation of parameters which are controllable in the course of project development and during the subsequent project refinement. The determination of limits and nominal starting data which describes the environment, materials and fuel, the conditions of take-off and the onboard system parameters.

4. The rough design of the spacecraft with the sketched placement of the payload and the junction point with the booster systems. The estimate of sizes and the solution of viability equation for a given spacecraft.

5. Refinement of the standard and nonstandard flight programs and the construction of onboard operational system cyclograms.

6. Selection and standardization of the onboard systems and, if possible, the synthesis of optimal composition with the sequential selection, first of the functional systems (radio communication means, means for navigation, control devices and trajectory correction devices) and then the air conditioned systems for viable existence of the crew, the thermal control systems, the power supply systems, the control propulsion units and the main propulsion units.

7. In the case of spacecraft or specific subassemblies, the basic purpose of which is the atmospheric flight, the development of proper configuration by selecting relatively simple geometric components with subsequent approximate evaluation of the braking and carrying parameters, based on the direct calculations or the data from aerodynamic tunnel tests, using the spacecraft shapes of similar type and taking into account several types of balancing modes, combinations of altitudes and flight speeds.

8. Coupling of the basic design with the propulsion units. The placement of control propulsion units and proper selection, on the basis of a prototype, of the number of main propulsion units with appropriate parameters. A more accurate calculation of the projected expenditure of fuel and its volumetric dimension, taking into account the required propulsions of the trajectory correction and fuel reserves.

9. Development of approximate calculated model for the joint /14
definition of configuration, motion and size-mass parameters of the spacecraft.

Establishment of programs and refinement of the algorithm for the combined project calculations.

10. The goal-oriented automated search or selection (frequently by using a computer) of the spacecraft variant and the selection of a number of competing alternative variants.

11. Calculation of the total weight of the design, calculation of the spacecraft weight - part-by-part and combined, for several variants under consideration, preparatory work to generate the project documentation: general appearance of the spacecraft, the schematic sketch design, theoretical drawing, limiting masses, standard flight program, information about the basic technical and flight parameters.

12. Discussion of the selected variants in regard to the general appearance of the spacecraft, with the leadership of the design bureau and at meetings with the coworkers who are developing separate systems. The detection of new and unexpected circumstances and conditions of the spacecraft functioning, the definition of emergency flight operational modes and non-standard situations which, as a rule, will be defined by the basic calculations.

13. Investigation of the accuracy of the obtained parameters for a spacecraft by using the calculated, analytical models which describe the operation of the systems, the distribution of weight and pressures across the surface area, the operational motion and heat exchange. Modeling of random scattering in the initial data and conclusion as to the possible intervals during which specific criteria for each of the selected variants may change. Formulation of the technical assignment aimed at the development of the major systems and formal representation by the design documentation.

Such general arrangement of the project at hand makes it possible to describe the model of the activities of the designers, to define by mathematical symbols the target and constraints of the project at hand, but as a rule this is still not enough for the development of a formal and universal algorithm which could be used for the start-up of a project, in designing any airborne device.

The major complexity in solving the design and search for the optimal solution is constrained because of insufficient data and the absence or reliable models for estimation of the specific criteria.

The mathematical formulation and model of the search for the optimum project design will look as follows:

The designers select two parameters $\bar{X}$ and $\bar{x}$, by altering of which, they may change the spacecraft appearance and the mode of its motion. The vector of formal integral parameters $\bar{X}$ reflects the standard equipment and variation of the systems on board, the types of fuel, materials, construction design schematics, number of propulsion units, emergency support units, etc.

The vector of continuous parameters $\bar{x}$ defines the geometry of the spacecraft (shape of the nose, degree of ellipticity, the half-cone angles, and sweeps of frontal shields, extended part of the body and the tapering down of the body), and the mode of motion along the trajectories (the moments in time for the switch-over to the other types of control, the blast-off time, rebalancing, the activation and deactivation of the propulsion units, the starting angles of attack or banking, the applicable coefficient in the programmed change of the pitch angles, etc.).

Each of these parameters is assigned the limits of change (the constraints of the first kind)

$$1 \leqslant X_i \leqslant K_i; \; b_j \leqslant x_j \leqslant a_j.$$

The starting data which describes the external environmental conditions ($\bar{\ell}_B$) and the parameters which define the efficiency of materials and fuels ($\bar{\ell}_M$) are systematically defined. The limits of change in these parameters are also defined:

/15

$$l_B^{(p),\min} \leqslant l_B^{(p)} \leqslant l_B^{(p),\max};$$
$$l_M^{(q),\min} \leqslant l_M^{(q)} \leqslant l_M^{(q),\max}.$$

There are some goal functions $\Phi_\xi(\bar{X}, \bar{x}, \bar{\ell}_B, \bar{\ell}_M)$, which can be obtained by using the algorithms analytically, graphically or heuristically, to estimate the values of projected parameters F_ξ.

It is quite desirable to improve some of these parameters, and then they become the group of FK_ξ criteria.

The relative importance of FK criteria is known approximately and it becomes possible to compile the approximate advantageous relationships by grouping together the criteria in terms of their importance, and isolating within one group the most valuable criteria which correspond to the goal-oriented effectiveness (the attainment of either longitudinal or latitudinal ranges, of the heights, speeds, availability of resources, depth of penetration into the planetary atmosphere, provision for the optimum probability that the flight program will be fulfilled, decrease of the stress effects on the crew) and the criteria which contradict them, indicating the degree of perfection in the systems and in the spacecraft design (total payload, volume of the payload, weight of the target-oriented systems, the total weight of the spacecraft, cost of the project design, etc.).

The design has some qualitative criteria which make it possible to attain simplicity and technological sophistication in design.

It may incorporate some approved and tested technical solutions, forms and principles of control, it may use the standardized blocks and subassemblies.

The relative value of antagonistic criteria and the indicators of importance of qualitative requirements is being corrected periodically in the model of decision-making, after the opinions of experts from the DMG team have been heard and processed.

In regard to some parameters FG_{η}, there is no point in trying to obtain the best or maximized results. It would suffice that they would be maintained within a specific range (speed thrusts, thermal flows, overloads, temperatures, expenditure of fuel, heat evolution, expenditure of power, total power output, total volume, dimensions).

One is faced with a multiple set of inequalities (the constraints of the second kind) which form in a complex way the area in which the design decisions are to be made.

The constraints of the second kind will have the following form

$$FG_{\xi} \leqslant FG_{\xi}^{(0)}.$$

Let us assume that the goal functions which are used in determining the criteria will be $\Phi_{\xi}^{(k)}(\bar{X}, \bar{x}, \bar{l}_B, \bar{l}_M)$, and that the goal functions which are used to calculate the limiting parameters are $\Phi_{\eta}^{(G)}(\bar{X}, \bar{x}, \bar{l}_B, \bar{l}_M)$.

Among the constraining conditions there are limitations which are in the form of the following equalities

$$\Phi_{\eta}^{(G)}(\bar{X}, \bar{x}, \bar{l}_B, \bar{l}_M) = FG_{\eta}^{(0)}$$

(the constraints of the third kind).

The purpose of one procedure in the search for an optimum design is to determine the spacecraft variant with the optimum values of the criteria

$$FK_{\xi}^{(opt)} = \text{opt}\, \Phi_{\xi}^{(k)}(\bar{X}, \bar{x}, \bar{l}_B, \bar{l}_M).$$

Since one is involved here with the vector criterion $\overline{FK}$, the designers are faced with the problem of defining a multitude of /16 pareto-optimal solutions $\overline{FK}^{(opt)}$ with the simultaneous development of the concepts in which one of the variants will be assumed to be be the best one, in the presence of several antagonistic criteria.

The limitations of the third kind make the procedural search significantly more complicated, resulting in the need to solve the boundary problems which require, for example, the placement of the spacecraft on the optimal trajectory for the assigned end conditions, or the selection of minimal thickness of the heat shield, permissible within the extreme temperature which is permissible in the design.

To satisfy the requirements of the synthesis of optimal design-power propulsion schematic diagrams, or the composition of onboard systems, one must take into account the conditional-logic constraints of the following type

$$\Phi_p^{(G)}(\bar{X}, \bar{x}, \bar{l}_B, \bar{l}_M) < FG_p^{(0)} \bigvee \Phi_q^{(G)}(\bar{X}, \bar{x}, \bar{l}_B, \bar{l}_M) < FG_q^{(0)}$$

(constraints of the fourth kind).

Among the constraints of this type, one frequently encounters in practice the incompatibility of some system variants, fuels, materials, body work and power propulsion systems, the principles of control, as one discretely selects the proper structural variant or arranges for the components, included into the spacecraft design.

The discretely-continuous region of the existence of solutions J has the maximal dimensions which are defined by the vector of discrete variables $\overline{X}(X_1, X_2, \ldots, X_i, \ldots, X_n)$ and by the vector of continuous variables $\overline{x}(x_1, x_2, \ldots, x_j, \ldots, x_m)$. In looking across the whole region J and considering the variants on the basis of a discrete grid with discrete steps, fixing on the scale of continuous variables λ, the total number of variants which are to be reviewed W will be

$$W = \prod_{i=1}^{n} K_i \times \lambda^m$$

The region of permissible solutions is $J_g \in J$. It includes all variants which remained, once certain variants were rejected, taking into account the constraints of the second, third and fourth kind.

The multiplicity of permissible levels of R criteria for each criterion FK_ξ is the reflection of J_g multiple and has the same dimensions.

The goal in searching for the best solution is ordinarily much wider than just one procedural search. It involves the elucidation of relationships between the optimal solutions on the basis of starting data and in the presence of constraints, in other words it involves the development of the relationships of the following type

$$\overline{FK}^{(\mathrm{opt})} = \varphi(\overline{I}_{\mathrm{B}}, \overline{I}_{\mathrm{M}}, FG_{\xi}^{(0)})$$

when

$$\overline{X}, \overline{x} \in J_g.$$

The numerical methods which are used to solve problems of such optimal design search are described in Chapters 3, 4.

CHAPTER 2.

MODELS FOR DEVELOPMENT OF NEW TECHNOLOGICAL IDEAS /17

One of the most complex, and difficult to formalize and model, areas of design engineering is the development of technological ideas. There is a broad area of activity for spacecraft designers who are involved with inventions, because of abundance of new technological solutions, the appearance of nontraditional and nontrivial configurations and design schematics of subassemblies and operational units, and the unusual operational conditions. Attempts to create an optimal and viable design forces engineers to investigate the little known physical effects and phenomena, far removed from the traditional sphere of engineering activities.

Under such circumstances, it is interesting and educational to try to generalize and investigate at least the simplest models which could be used to search for technological concepts by systemizing the approaches in spacecraft design and the known techniques applicable to inventions.

The assistance which could be rendered to the designer by the methods in preparing the inventions and computing technology may turn out to be quite useful to analyze the computed models of functioning, to elucidate the optimal and limiting design parameters, to gather and store information as to the specific properties, features, prototypes, materials, etc., which have either been only slightly used or recently discovered, and which apply to the environment and technology.

This book will try to generalize the questions of modeling of the synthesis operations of new ideas and distinguishing features for the spacecraft which are to be designed.

In the case of readers who are interested in methods and examples of design research, we recommend the book [1] which considers also the utilization of computers for data gathering and variant selection in order to improve the first prototype. The most widely used approach in the domain of inventions in our country is the methodology developed by G. S. Al'tshuller, which handles the problems associated with inventions and is being taught in the schools for creative inventions, within the All-Union Society of Inventors and Efficiency Experts.

The proper search for technological ideas is a rather complex process which is quite individual, which occurs differently among different people, and is supported to a considerable degree by the intuition, experience and creative capabilities, and therefore to formalize it completely in algorithmic descriptions is not possible.

Every sensible man will agree with this unquestionable assertion. The accumulated experience of design work indicates that the bulk of technological ideas and discoveries has been developed without using ready-made recommendations and "manuals on inventions." Attempts to represent the process of inventing as a goal-oriented selection among the potential variants, with sequential improvement of parameters of the first prototype are quite unfortunate.

Quite recently, the successful utilization of a computer for such variant selection in the simple engineering problems gave reason to believe that the computing technology will be able to invent and develop technological ideas.

It is known that many renowned inventors who have proposed the unique design concepts, as a rule, approached their goal in an entirely different fashion, by using atypical ways of thinking and /18 were preparing differently for the "bolt of lightning," which ordinarily follows with a new and unique idea. However, almost all inventors must, perforce, carry on with a very thorough and cumbersome job in analyzing the prototypes from the adjoining domains of engineering, they have to approach sysematically the requirements which are imposed on the design, the design constraints and gathering of all other data which would describe the operational conditions and distinguishing features of the future invention.

"Information is the mother of invention" - this clear-cut formulation must be remembered by all who forget that the development of ideas is unthinkable without preparatory work, without studying similar design solutions, without extensive knowledge about the use of different physical effects and phenomena.

And yet, as a rule by studying and rethinking the known ideas and design concepts, one can end up with ordinary technological solutions which are in essence the modernization of already used methods or devices which are extrapolated to a new area of the engineering activities. Such technological solutions are ordinary called "weak" although the majority of them may comprise the subject matter of the invention.

The "strong" solutions are characterized by their uniqueness and in searching for them, one encounters factors of unexpected turns in thinking. Such solutions are always attractive and indeed, quite unique. The desire for novelty has brought about various methods which stimulate the imagination, which are being used quite successfully. Any means are applicable for the discovery of unusual properties or methods of use, including the techniques which develop flexibility in thinking of professional experts, who are as a rule tied down by the traditional approaches in their engineering activity.

It is not by accident that the development in methods which would improve the search for new ideas commenced with the mastery of approaches in developing the unexpected technological solutions (the methods of "brain trust storm").

The essence of the method of "brain trust storm" is to create first the "arbitrary" set of different ideas and subsequently, to process these ideas critically. A group of people with great imagination is selected for this process. Among the "initiators of ideas" may be professional experts, but is better to include in this group of people the experts from neighboring domains of engineering activity.

The success of "brain trust storm" depends to a great degree on the skillful organization of the exchange of ideas and concepts, and the range of questions in science and technology which are to be considered.

The purpose of the leader here is to ensure psychological creative activization and to get away from the "vector of psychological inertia." The psychological inertia is due to various factors, among which are the fear to propose a ridiculous and foolish idea, the risk to be criticized by a professional expert, the slowing down of imagination because of the absence of skill in developing unusual proposals. The leader must create a friendly creative atmosphere, he must maintain and encourage the airing of unusual ideas and also must restrain and prohibit criticism, so as to overcome the psychological barrier of shyness and develop competitive conditions in the originality of thinking, stimulating the participants in the "brain trust storm" to actively participate and express bold ideas. It is desirable that the participants in the discussions would entertain and expand the proposed ideas. Special types of questions which expand the utilization of the proposed ideas are used to "distract" the inventors from the ordinary point of view on the problem which is being handled.

The subsequent critical processing of ideas is being conducted /19 by a group of "critics" who are the experts in the idea handling, and who are capable of finding a rational concept in any proposal, defining the acceptable way to proceed with further development of such an idea.

The unique feature of the "brain trust storm" method is its simplicity and accessibility and also its universality with respect to the problems which are to be investigated (technological, scientific, organizational).

One of the effective methods in collective creativity is the senectic method (the method of combining different components).

The basis of the senectic approaches in developing ideas is the active utilization of analogies which are inherent in some other domains of technology, in the environment, and in the personal perception of man.

The addition, to the methods which help to develop new trends in finding the proper ideas, are methods which systematically search for ways, and then by being coupled to the methods of the "brain trust storm," create orderly-directional and controllable continuation toward the goal of the invention. For the purposes of progress, one utilizes the method of control questions, which stimulate general ideas about the target-oriented object which is being designed and conditions of its functioning, making it possible to define the imaginary limiting or ideal solutions, which are the "record setters" with respect to some specific properties needed in the future design.

The method of morphological analysis makes it possible to organize analysis of possible multiple solutions. One isolates several basic features of the object which is being designed which forma matrix, combining in one direction the morphological features or properties which indicate the target-oriented design, and in the other direction - the variants of possible technological execution of such properties. By compiling such a matrix, it becomes possible to see in greater apparency the "field" of possible design solutions which are defined by the combination of variants, ensuring that all properties of the future object in question will be incorporated.

The methods of matrix construction which defines the distinguishing features and means to obtain them, as well as the approaches for a variant selection, may be in a general case formalized and may be reduced to the operation with algorithms of integral programming which has been considered in the studies [45 and 48], in the course of formulation of structural synthesis. It seems that it would not be a waste of time to refer to the data reference systems which are based on the modern computers and use them as partners for the selection of interesting and unexpected properties and the structural composition synthesis of the objects which are being designed. The search algorithms which are to select the proper combination of components from a multiple set of components, with the associative correction during random sampling, are described in detail in the study [45].

In our country, we are quite familiar with the "algorithm /20 for the solution of inventor's tasks" which combines and systematically treats a number of activities, refining the whole problem, setting it up in concrete terms and removing technolgical contradictions, expanding the initial solution, so as to make it approach the a priori formulated ideal end result.

In systematic handling of the properties, one utilizes a set of physical phenomena and effects, and also some clearly defined illustrative models, making it possible to overcome the psychological inertia. Among the operators, handling such algorithm, are the operator who is to formulate the interaction between the object which is being designed and the environment in which it is to function, the operator who defines the end goal and selects the general direction for the proper solution, the operator of functional synthesis in the subsystems, the operator of negation, the operator of alternative comparison, the operator of solution transfer, and others.

In spite of the multiplicity of various recommendations in developing the technological ideas, one can combine all rational approaches and formulate the general principles for a model to develop new technological solutions.

In accordance with these principles it is necessary:

to formulate the end goal, to describe an ideal result and to handle systematically all properties, requirements and constraints which affect the definition of the configuration, structure and functioning of the object which is being designed;

not to be distracted by some specific properties of the object which is being designed, attempting to utilize as a prototype a completely unique and unexpected solution, for the time being disregarding the requirements and constraints, developing the "outsiders" view, which is to imitate the dilletants's point of view, who is free of pragmatic expert "prejudices";

to search for analogies in other domains of engineering activity, utilizing new physical phenomena and effects in the adjoining technological areas;

to represent in all apparency the functioning of the object which is being designed, modeling its activity by using simplified analogies which will make it possible for the engineer to "feel" the interaction between the object and the environment;

to set up and review all possible combinations of variants which would have all the basic parameters characterizing the solution which will be generated.

To attain the above-mentioned criteria, it is reasonable to be supported by the modern computer technology. Here one has the real possibility to automate the following operations:

to gather the data about the physical phenomena and effects which are only slightly known to engineers, the use of such techniques in the other areas of technology for analogous purposes. For the accumulation and systematic handling of the information about the physical effects, it is reasonable to develop special data banks which would tie together the information about the required properties, the physical phenomena which realize such properties and also the basic parameters and consequences of utilizing the appropriate physical effect;

to develop operational data generation about the typical engineering solution in the neighboring domains of engineering activity. The storage of typical design solutions may also be organized by using /21 computer data carriers and utilizing the data reference system and developing the images by computer graphics;

to compile the matrices for the goal-oriented features, the proper selection of the most likely variant of the features which are to be generated and which would characterize the ideal solution. The development of unique combinations of variants may be organized by using a special algorithm which utilizes random sampling.

The models for development of new technological ideas are constructed in different ways, depending on the directions of problem-solution and on the degree of novelty in the technological solution which is being handled.

It is reasonable to separate three different levels of novelty and correspondingly, to employ three active approaches.

The first level is fundamentally new, "strong" solutions which correspond to the pioneering inventions which have no prototypes. The approaches which are to be used for search under such circumstances are quite individual and are supported by the psychological creative activity.

The second level are the solutions which feature a degree of novelty, but which do have prototypes in the environment and in the other domains of technology. In such cases it is quite effective to undertake a thorough systematic review of the required features and properties and consider the analogies for the ideal conditional models, with the formulation of "limiting" solutions which would maximize one of the properties of the future method or device.

The third level corresponds to the modification of known inventions, so as to improve some specific properties.

The example which we shall present below characterizes the second and third search approaches which are most frequently used in the practice of design.

We are familiar with the configuration of landing spacecraft of segmented-cone type, with an extensive frontal area, used as the braking device and thermal shield.

The flat-bottomed cone for the payload is within the protected zone and the aerodynamic properties of the device are defined primarily by the frontal part of the cone and its angle of attack and the surface of this part of the cone comprises about one third of the total surface of the device.

The angle of attack turn, during the flight, is provided by the mass balancing, in other words by the shift of the center of gravity of the device axis toward the wind-facing generatrixes of the rear part of the cone. This provides for a good lift-to-drag ratio (on the order of 0.2-0.3) which, in combination with a considerable frontal drag, provides the effective braking in the atmosphere of the Earth and other planets during orbital descent or during atmospheric reentries.

Such are the general shapes of the descending and returnable spacecraft of different kinds (for example, of Soyuz, Apollo, Gemini and others). In some cases, the frontal cone is in the shape of a truncated cone which, in terms of its configuration, resembles a spherical segment. A considerable shift of the center of gravity sideways from the spacecraft axis results in some inconvenience in the component arrangement, makes it more complicated to set up the equipment and also requires some additional ballast.

One of the design criteria, the foundation of which may be found in the book [15], is the ratio of volume, bisected by the plane passing through the resultant, to one half of the total volume.

If the design efficiency factor drops to 0.7-0.5 in the case of the spacecraft having Q on the order of 0.5-7, then it becomes impossible to design the spacecraft of segmented-conical shape without having considerable ballast.

This is precisely the lift-to-drag ratio which is required for the descent through the atmosphere of the planets in the future. In the case of the above-mentioned descending spacecraft, it is /22 required to have simultaneously a more pointed frontal area because of the increased thermal flows, while in the case of convective thermal flows which predominate today during the reentry flights of spacecraft, one requires the truncated frontal area [15, 45].

In order to improve the spacecraft configuration - everything else being equal (retaining the braking method by the frontal surface with moderate curvature) one must change the design parameters, providing for more pinpointing at the area of maximum heating, in the region of the critical point, attempting to preserve the shape "fullness," in other words to preserve the acceptable relationships between the surface area and total volume.

Figure 2.1 shows the devices of segmented-cone and semicone shape, with the computed lift-to-drag ratio of 0.5 and 0.7. The spacecraft frontal surface areas of the resulting semicone shape are transformed in accordance with the above-mentioned requirements. The figure also shows the position of the resultant aerodynamic forces.

The argument in the spacecraft design change may be as follows: it is necessary to attempt, by retaining the general contours and shape of the assigned cone and without changing significantly the frontal surface area, to make it flatter in the area which is far-removed from the critical point and more pointed in the area of maximum heating. In doing this, one simultaneously moves the aerodynamic resultant closer to the axis and improves the heat exchange in the most critical frontal zone. Mentally, one can imagine such frontal area as a resilient and elastic membrane which may be flexed at the critical point.

The same result may be obtained by the target-oriented computer variation, utilizing the numerical optimization methods which describe the parameters of the frontal cone (the half-angle sweep in the plane of the drawing, the degree of ellipticity, radius of truncation, axial inclination of the cone with respect to the velocity vector or with respect to the plane of the central point of the spacecraft). In searching for the optimal solution, the criteria which may be used are the displacement of the center of gravity from the axial position of the rear part of the cone, the design efficiency factor or the conditional mass of ballast.

The frontal areas of the semicone shapes, shown in the figure, have another advantage: the airflow is at zero angle of attack which facilitates the calculations of the gas dynamics and heat exchange, thus making it possible to disregard the calculations for asymmetric flow in the spacecraft wake.

It is not by accident that various modifications of the semicones were the subject matter of inventions.

The study [15] shows the example in searching for a proper shape of the descending spacecraft, returnable after interplanetary flight. Here one considers the stages of transforming the segment-cone and then the semicone configurations by gradual improvement in the contours, taking into account the weight of the thermal shield, the design and component criteria. The semicone shape as shown in Figure 1, in addition to a number of novel components, has a number of standardized technological solutions and has been selected as a rational design for the Earth atmospheric reentry with hyperbolic velocities.

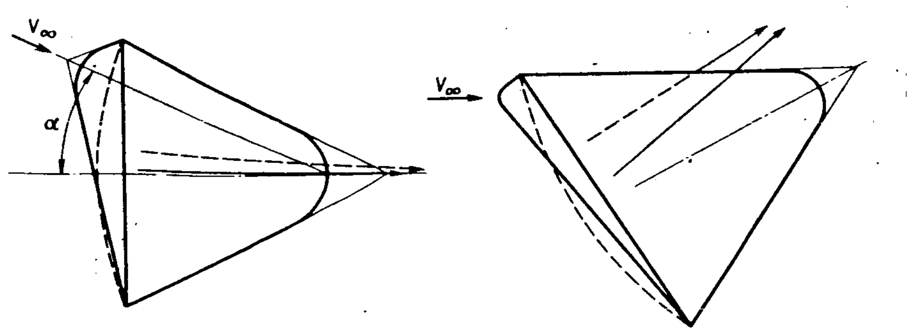

Figure 2.1. Segmented-cone and semicone configurations of the descending device with the available lift-to-drag ratio equal to 0.5 (to the left) and 0.75 (to the right).

CHAPTER 3.

CALCULATION AND MODELING OF UNCERTAINTY FACTORS IN SEARCHING FOR AND SELECTING THE DESIGN SOLUTIONS /23

At the initial stage of spacecraft design it is necessary to make the responsible decisions, associated with the significant material expenditures for the accomplishment of the whole program, and also associated with the scientific research and fulfillment of the national economy tasks. The technological risk associated with the development of the design solutions must be evaluated objectively and one must find the optimum strategy in searching for the proper design.

In the case of spacecraft which are designed for new flights toward the planets, and consequently a new set of operational conditions, of greatest importance are two problems: the problem of proper criterial base (definition, arrangement in terms of rank and comparison in terms of the importance of the criteria, requirements and constraints) and the problem of objective modeling which would define the fluctuation of design parameters on the basis of which the prognostication is possible as to the possibility of the spacecraft development which would correspond to the design.

In trying to generalize a typical situation in which the designer finds himself, who is to select the general features of the spacecraft, let us point out the need almost always existing for a compromise between two contradicting criteria-requirements.

One of these criteria reflects the goal oriented efficiency of the spacecraft and contains the projected factors which would indicate the fulfillment of the flight goal or would prognosticate the proper functioning program for a family of spacecraft. Among these for example, one could obtain the maximum level of fail-free autonomous operation, the greatest probability that the tasks will be attained, the maximum possible range, the largest surface area which the planetary automated device could traverse while carrying on with the investigation, the braking modes during the return flight with the best stress conditions of the crew, the attainment of planetary regions which are furthest removed from the orbits of the mother ship, the biplanetary trajectory flights of the largest number of spacecraft, the maximized data gathering in regard to the atmospheric parameters and in regard to the planetary surface and lowering of the program cost.

The other criterion (or a group of criteria) refers to the relative design perfection and perfection of the systems on board. As a rule, these are the indicators of mass which reflect the weight of means utilized for operation and the payload delivery, and the weight of service systems, namely: weight of the spacecraft, of fuel, of thermal shield, of landing gear, propulsion units and control devices, of the power supply systems, of the systems of thermal control and life support. As a rule, the spacecraft mass is fixed and is defined by the acceleration systems and rocket carriers.

Let us present the examples of contradictions of these two criteria in selecting the general features of the spacecraft.

The return to the Earth of a crew from the interplanetary flight requires the "extended" trajectories with nominal overload and stress which is difficult to attain, because of the requirement to minimize /24 the mass of the thermal shielding which calls for short trajectories and intensive braking.

In trying to find the composition, ways and levels of instrumental operation, of the operation of assemblies and fuel supplies for the autonomous interplanetary spacecraft, one will unavoidably face the contradictions between the probability of fail-free operation and the mass of back-up subsystems and subassemblies.

A lengthy active working cycle on the surface of a planet, with exposure to local temperatures and pressures, is not provided by the minimal mass, rudimentary passive means of protection and in improving this situation, one must use the extra weight capabilities and space which is designed for the useful payload.

It would be desirable to develop a whole series of standardized multipurpose satellites for the plantary data gathering, with the maximized standardization at the point of junction assemblies, power supply systems, the systems of thermal control, the means for stabilization and radio communications. While standardizing these items one could lower the cost of the space program. In the case of any satellite, the price of standarization is the construction weight increase, because of the overweight factors, extra capacity of power output in the assemblies, and the oversized, standardized compartments.

The penetration under the cloud cover of the giant planets by a probe with research equipment and radio signal transmission to the orbiting spacecraft generates much more scientific data than by studying it by remote control, using the planetary trajectories. However, in planning for the descent of such a probing module, one would have to increase considerably the weight of the thermal shield, of the frontal surface area, of the probe itself, of the balloon accessories for braking and controlling the approach trajectory which in turn would influence the available weight left for the research systems of the whole mission.

The efficient activity of the self-propelled planetary mobile unit which could investigate the regions far removed from the landing site of the space module is proportional to the distances traveled, and depends on the amount of research equipment. A lengthy active operation must provide for extra strength of drive, extra fuel, extra power supply units and heating units, thus increasing the power output and thrust parameters of the propulsion units, in other words the expenditure of allocated weight which could be otherwise used for the equipment and local studies of the landing site.

The rocket propulsion units reserved for the near-planetary activities on board of the multipurpose spacecraft ensure a simple development of research orbits for the study of the upper atmospheric layers. The alleviation of near-planetary maneuvering with the fuel weight and power economy makes it possible to conduct the remote probing along the flight trajectory during a short time period, but by using a broad range of research instruments.

In starting up to design the Martian plantary spacecraft, it would make sense to elucidate which is more advantageous, the landing of the simple designed module by using the approach trajectory, disregarding in the course of descent the aerodynamic Q factor and landing in the equatorial regions a huge quantity of research equipment or employing more complicated, in terms of execution, means for orbital placement, with the subsequent "sliding" descent toward the polar cap, of device with a good lift-to-drag ratio for maneuvering, but carrying minimum weight research equipment.

The design parameters of acceleration units or the air-tight compartments in the spacecraft (cross-sectional dimensions, spacing of beams and frames, position of the propulsion supports, thickness of shells) are selected on the basis of two contradictory criteria. The first is the design weight and the second is the rigidity which is evaluated on the basis of relative deformations at the junction of frames and supports for the propulsion units, and the adjustment of the navigational sensitive components. The examples of designing such a model, with the equivalent strength of design and the appropriate operational conditions are considered in detail in Chapter 7.

The requirement of minimum mass in the design and in the thermal shielding of the descending modules contradicts the requirement of centered balancing and it turns out that the excess mass of the frontal surface area and the thermal shield for some types of devices could be used for balancing.

In an analogous fashion, by minimizing the length of cables, along the frames and panels of the instrument compartments, one should take into account not only the weight of the electric conduit bundles, which are armored and fastened, but also the central balancing and the displacement of the center of gravity from the compartment axis must be compensated by some weight addition.

The living quarters of orbital stations must feature multi-layered, light shell consisting of an air-tight frame and thermal insulation, with the material having low specific weight. However, /25
for protection from solar flares, one would require some local thickening of the shell, so as to create the proper protection from the background radiation, or to design radiation shelters.

The propulsion assemblies and panels of instruments or support frames of some assemblies should be designed to be of the minimal mass required. This requirement almost always is in contradiction with the technological factors which dictate the limiting or minimal standard equipment and sizes of parts, of stampings and other equipment which indirectly is associated with a decrease in the cost of the items.

Within the framework of an alternative in selecting the compromise solution by using even two antagonistic criteria, there is no one single optimal variant, in other words, a variant which would in the traditional sense have maximized parameters in terms of each criterion. However, there are variants which are easy to define (but difficult to recognize) which are combined into a family of best possible, or the best, boundary compromises or the pareto-optimal variants.

The pareto optimal variants can be recognized only after comparing them with the multiplicity of solutions already considered. This is precisely why the variant search falls within the pareto-optimal family, or criteria similar to pareto-optimal, and the job must be conducted by an individual who can recognize the most advantageous variants and who simultaneously makes the decision as to the ways of transforming the variants from those which are already selected by the decision-making group (DMG). If, among the multitude of already considered variants (or, presumable, among the multiplicity of permissible solutions), there is no single competing variant with the one selected by the DMG, it must be assumed that selected variant is pareto-optimal.

A known model of typical activities of the DMG team in selecting the pareto-optimal solutions is ordinarily based on the development of stereotype representative which presumably define a constant relationship of the importance of criteria within the idealized hypothetical solution. Such a model is formalized by the so-called linear, additive or multiple convolutions. Within the convolutions, the criteria ratios are modeled by constant coefficients which project a general, preferred direction as a hyperbeam within the space of criteria. The movement is organized by different means along this beam (regardless of the initial relationship between the criteria).

In practice, the model of typical activities of the DMG team undergoes significant changes. The actual recommendations of the group are far from the canonic relationships in terms of the importance of criteria. Depending on the proximity of the attained levels with respect to the limiting one, or on the basis of the degree of attainment of specific quality, as compared to the worst situation, the binary relationships of mutual criteria values will change. Such dynamic state of the opinions within the DMG group is known to anyone who has encountered in practice the problems of vector optimization. Figure 3.1 shows isometrically the coordinate plane of K_1K_2 criteria. Here a typical case of maximized antagonistic criteria is demonstrated.

The K_1 criterion may designate, for example, the mass of goal-oriented systems and of the payload (the total mass minus the weight of the basic frame, of the propulsion units, of the thermal protection systems, of the power supply systems and of the thermal control systems, of the landing gear and of the control systems). The maximum of K_1 criterion (point 4) corresponds to the limiting technological solution, in the presence of which one attains the ideal perfection of the design but the spacecraft is useless because it cannot accomplish the target mission. /26

The K_2 criterion signals that the target mission is completed. It indicates for example, the probability of fail-free operation, the probability that the specific assignments will be executed even during deviations of the flight program, it indicates the cost of standardized equipment, the best parameters in terms of resources, in terms of the range or prelanding maneuvering, in terms of overload action, etc.

The point 2 shows the best level of target-indicated effectiveness, but is unacceptable in terms of the design perfection factor. The boundary regions of possible solutions (the arc from point 1 to point 2) includes the pareto-optimal variants, each of which competes with the neighboring one but under all circumstances, is better than any of those which are in the plane of criteria values which are closer to the beginning of the coordinates.

The dynamic state of opinions in the DMG team can be easily illustrated by the arrows, indicating the direction of improvement in different variants from the point of view of increase in the efficiency parameters and the perfect design parameter. The general idea about equivalence in the relationships with respect to the criteria in the future hypothetical solution can be obtained by analyzing the work of the DMG team, handling a quite ordinary variant, the criteria of which correspond to point 5 in Figure 3.1. In this case, one assigns approximately equivalent weight coefficients and brings about (if it is possible!) the movement along the beam, from point 5 to point 1.

If the DMG team receives the initial variant which is similar to the best values of K_1 criterion (point 4) or K_2 (point 3), then it suggests to use the relationship between the weight coefficients, "favoring" the "underdog" criterion, by determining the advantageous improvement of the criteria in the direction, as shown by the arrows.

In the actual design work, an experienced coworker within the DMG group will request some additional information with a complete set of data about the qualitative factors and criteria of lower rank, which were not previously taken into account. He will familiarize himself with the worst design solutions, wishing to study the early history of variant search and would like to learn the whole complement of variants which have been considered before the selection of a specific, initial prototype, on the basis of which the model of the relative value of the criteria has been constructed.

After obtaining, on the basis of the model describing the values of criteria, one pareto-optimal variant, the expert from the DMG team will inquire about the top values of each criterion and will try his best to elucidate the whole series of pareto-optimal variants or preoptimal ones which are not far from them. /27

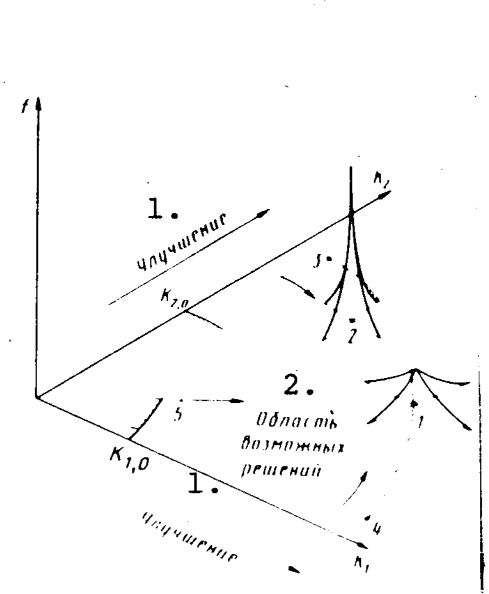

Figure 3.1 Area of possible solutions in selecting the design variants on the basis of two contradictory criteria, taking into account the uncertainty factors.

Key: 1. Improvement; 2. Area of possible solutions

The bulk of recommendations naively consider that in solving the vector optimization problem, it is possible to undertake the goal-oriented transition within the space of criteria from, say point 5 to point 1. Unfortunately, the spacecraft designer is able to solve the so-called direct design problems, in other words he can determine the values of criteria, on the basis of parametric values (of the shapes, composition, of the control of motion).

The inverse design problems cannot be solved since there are no methods available which would enable us to determine unambiguously, accurately and without sequential approximation, the shape of the spacecraft on the basis of a set of parameters.

The most important feature of the comparison of variants by the DMG team experts is the evaluation whether they can be constructed, the degree of reliance that the top quality indicators may be attained, on the basis of internal relationships and the clear understanding of technological risk.

This fact is ordinarily ignored by the commonly accepted models in decision-making, involving the problems of vector optimization. The complications caused by uncertainties in evaluating the appropriate criteria are particularly inherent in the spacecraft design when the new flight conditions are at hand and the operational environment is not clear, when there is some uncertainty in calculations, flight programs, the expected and unexpected situations.

Figure 8 shows conditionally the multiscale relationships in the density of criteria distribution for two variants at the criteria plane (point 1 and 2). The first variant is characterized by the maximized nominal values of the criteria and are on the pareto boundary. However, its optimality is explained by the use of rather promising, but poorly investigated materials, which were used in design and thermal shielding, the utilization of new fuels with rather optimistic parameters, the equipment with a different component base and the principles of navigation and control which are based on recently discovered physical phenomena. However, the density histograms of the criteria distribution are of "smeared-out" character, thus decreasing the level of trust that the experts from the DMG team could have in regard to the record parameters in this variant.

The second variant is further removed from the pareto boundary and the averaged-out values of its criteria, as it appeared, could not excite the DMG team.

This variant has the well-tested variants of the onboard systems and assemblies with analogous designs and prototypes in the past. This design utilizes standardized compartments and subassemblies in which the known materials, coatings and propulsion units were used and the types of fuel are well known. Therefore, the density curves of the criteria distribution in this variant demonstrate only slight deviations from nominal values.

The DMG team is interested not only in achieving the record factors in the design, but rather to learn how real and how well-substantiated are the design parameters, in other words, whether it will be possible on one hand to fit into the available resources, with the appropriate allocation for the payload and on the other hand - to retain at the acceptable level the parameters of the goal-oriented efficiency. /28

On the scale of possible values of the criteria, there are the boundaries ($K_{1,0}$ and $K_{2,0}$ in Figure 3.2) beyond which no single criterion should be located, once the project is brought to fruition. These limits are defined by the lifting capacity of the carriers and of the booster systems, by the minimal standard equipment used in conjunction with the useful load, by the permissible system reliability or probability that the tasks will be fulfilled, as stated in the special memoranda, associated with the cost of the project.

The random scattering of the design parameters results in the need to take into account the probability that the project will not be completed, as the accepted project variant is being used.

Under such circumstances, the model for selecting the project decisions is based not only on the deterministic calculations, but also also on the probability approach in comparing the variants and using the theoretical game procedures of comparison, pertaining to different possible alternatives in finishing the project on the basis of the technological risk criterion.

Before we shall consider the probabilistic-statistical model of the design solution selection, let us analyze the causes, on the basis of which the parameters which will become real as the spacecraft is designed, do not agree with the stated parameters at the initial design stage. There are three such causes: firstly, the errors and shortcomings associated with the calculations, with the design work and with preparatory documentation. Secondly, the imperfection of the initial approximate formulas and simple calculation models, of the structure and processes of spacecraft operation which disregard the technological requirements, the conditions of assembly and system tie-in and also some unusual situations; and thirdly, the scattering of the initial data values which describe the operational environment and the parameters of materials used, of the fuel, power, dimensions, weight and thermal heat evolution from the assemblies.

To model the factors which are defined by the first two causes is extremely difficult.

If one could carry through several design variants up to the detailed experimental stage and thorough design refinement, then the Q factor of the "initial calculations" and degree of reliability that the outlined design is fault-free could be quantitatively measured by the design Q factor, which is determined on the basis of technological risk. The detailed concept of technological risk we shall consider later, but at this time let us assume that there is a procedure for the evaluation of probability that the design project P^+ will be completed. The risk that it will not be completed P^- is evaluated as $1-P^+$.

At the initial stage of design and by using a special statistical modeling which we shall consider below, it was possible to obtain for one of the variants, on the basis of deviations in the initial data (in the environmental conditions and the factors of the design materials and fuel) the a priori values of P_1^+ and P_1^-.

For the same variant which has actually been constructed, it was possible to investigate and model some new deviation in the initial data, transformed by using the statistical models, but now taking into account the half-scale research experiments, the concrete calculations of some experiments, the tests of fuel and material in the assembly, functioning on the test stand, obtaining now the a /20
posteriori values P_2^+ and P_2^-.

As a rule, in practice we will have the $P_2^+ < P_1^+$ relationship. The K_c factor which indicates the confidence level in the initial project estimate, and reflecting the integral number of errors during the project start-up, is obtained from the following relationship $K_c = P_2^+/P_1^+$.

The problem of modeling the uncertainty factors by using the technical risk criteria is based on the sensitivity analysis of the finalized project data (the project criteria) and the permissible fluctuations in the values of the initial data.

Among these, and quite uncertain and difficult to model, is the data about the initial conditions and flight programs and, consequently, the operational cyclograms of the multipurpose devices and systems on board, and the data and calculations for the unexpected situations. Among the initial data we find the magnitude of specific pulses for promising fuels, the effective enthalpy of thermal coatings or information about the percent composition of the planetary atmosphere and scattering of these values is modeled by using the expert estimates, described in Chapter 5.

The scattering in the majority of starting data is modeled by the normal law with the designation of the whole range of permissible changes equal to three root-mean-square deviations. Among such data we find information about the internal conditions of the operation: average frequency of failure (or the operational time before failure) of the instruments and assemblies on board, the levels of fuel supply, the strength and rigidity parameters of the metallic and composition materials, the specific densities of fuels and equipment, the accuracy of sensitive components in the navigational means, the specific fuel expenditures during the transition processes of spacecraft maneuvering. In the same group we find the information about the environment: the atmospheric density and gradient of its change as a function of the planet and altitude, the macrorelief data of the landing site, the parameters of the soil and microrelief, the speeds and parameters of the entry corridors into the atmosphere, the characteristic velocities of switch-over trajectories and corrections, the deviations in the blast-off data and the changes in external temperatures and pressures.

The fluctuations in the initial data frequently makes the possibility to finalize the project sometimes doubtful. For example, with the existing and varied opinions as to the mechanism of the thermal shielding removal, taking into account the protection available (see Chapter 5) and the insufficient knowledge of the thermal and physical parameters of the spacecraft coatings, protecting it from the gigantic but not accurately defined thermal radiation flows, as the spacecraft enters the atmosphere of the giant planets, the mass of the probe with a thermal shielding, penetrating under the Jupiter cloud cover, is being estimated, as has been shown in the study [15], with the accuracy on the order of 30%.

The mass of thermal shielding of the shuttle spacecraft, entering the Earth's atmosphere with hyperbolic velocity, is being estimated because of the above-mentioned uncertainty factors, with the accuracy on the order of 15%.

The optimistic and pessimistic probability estimates of the fail-free operation of the onboard systems in the interplanetary spacecraft, which is to travel in the course of 5-7 years in the region of the outer planets, differ by a factor of 3. The model of probability estimate of the successful operation of systems in such spacecraft, depending on the operations and time of flight, is described in the study [45].

The mass of booster units with nuclear power propulsion (solid state reactor of "A" type within the chamber of the propulsion unit) which utilize cryogenic fuel at the final stages of acceleration during the Martian mission, is estimated with accuracy on the order /30
of 20% and the fuel losses because of evaporation during the two-year time period are estimated to be from one third to one half of of the total fuel mass which will be used. The mass of the same acceleration system with the electric nuclear unit of small thrust, with the present-day general knowledge about the specific pulse and the characteristics of the heat carriers, fluctuates within the range of 40%.

By using the optimistic estimates of the fuel container weight in the booster system design with binary fuel supplies of single-stage type (SALKILD) for the take-off from the Earth, it is, in principle, possible to place the payload into a near-terrestrial orbit which is about one percent of the blast-off mass. In using the pessimistic estimates of the specific square meter weights of the fuel tanks and the shell structure itself, such system would not be capable to attain the specific velocity necessary to place into orbit its net-dry weight.

Quite recently the design of devices which were to descend on Venus required only small weight special shielding devices for the equipment of planetary probing and the relative mass of the descending equipment for landing, using the optimistic estimates, was less than one half of the total mass of the device itself. If one is to employ the worst data about the pressure, temperature and time needed for touch-down, one would have to reduce the mass of the research equipment to ten-fifteen percent of the total mass of the planetary module.

By employing the statistical testing modeling to measure the variation in the figures of starting data, it becomes possible to obtain the distribution or random criteria functions and to evaluate the limiting fluctuations of these figures from the nominal values. On the basis of such distribution functions, one develops the reserves, based on the computed values of the respective parameters, one refines the limiting masses and one determines the probability that the project at hand will be realized, in other words one develops in this fashion the technical risk parameters.

One ordinarily understands by the technical risk parameters the probability of such situation, as the project is being brought about, when the design limitations on the parameters will not be satisfied. The technical risk modeling presupposes the presence of accurate algorithms, the calculation programs and optimization of size-mass parameters, as well as the operational processes.

The models of technical risk estimates are based on the following conditions and constraints.

1. The availability of mass-related statistical prototypes, on the basis of which one obtains a high degree of verifiable probability in regard to the distribution of parameters (which are in the form of test histograms).

2. The absence of relationships between the computed parameters of different design variants, in other words the independence from each other of different criteria calculations.

3. Separation between the meaning of technical risk and fail-free probability of operational work. The technical risk criteria compare the prognostication parameters which were made during the project start-up, which, on the basis of the parameters of a given project design, are verified and approved before the device is being used.

If P_+ is the probability of prognosticated value within a specific range and $P_- = 1 - P_+$ is the probability that the design and task at hand will not be fulfilled (the technical risk), then the advantage of one variant as compared to another (for example, the second variant as compared to the first one - see Figure 2) should be evaluated on the basis of complex event probability $P(2 \rightarrow 1)$ which states that in designing both modules for the second variant, the design criteria ($K_1 > K_{1,0}$ and $K_2 > K_{2,0}$) will be satisfied and in the case of the first variant the ($K_1 < K_{1,0}$ and $K_2 < K_{2,0}$) will not be satisfied.

It is clear that:

$$P(2 \rightarrow 1) = P_+(1) - P_+(1) \times P_+(2).$$

If the constraints of the design are quite rigid for the variants which are being compared, in order to obtain the technical risk parameters for the acceptable probability with acceptable confidence factor becomes a rather time-consuming project, involving a large amount of statistical testing. Under such conditions it is reasonable to compare the variants on the basis of limiting and most inferior, but still attainable criteria, comparable to the assigned boundaries for such criteria.

The models for determination of limiting-attainable levels of criteria are based on the theory of statistical extremal values. Chapter 5 describes the modeling method for maximum values, using the third exponential distribution, as recommended in the special bibliography.

Any model of decision-making in which the technical risk parameters are being used is being utilized in practice in the form of subjective opinions of the experts, and therefore a special survey of the DMG team must be carried out, in an attempt to define the favored risks, as assessed by each individual, estimating the readiness for such risks and losses, as well as the advantages, using the scale of criteria value.

The positive results of such comparison, using the technical risk criteria, depends to a considerable degree on the level of knowledge and know-how of the experts from the DMG team, who are able to perceive and prognosticate the possible changes within the project design.

A cautious approach to the selection of a variant in the presence of an uncertainty factor (which is quite characteristic in the case of experienced DMG team personnel) calls for the utilization of methods which guarantee success, on the basis of the worst resolution of the project design.

Among the theoretical and game methods which are extensively used and which provide for the fail-safe decision-making, is the minimax approach.

In contrast to the classical minimax strategy of selection, the design model for an acceptable variant aims at the three-level search procedure for the optimum design parameters in the presence of the worst operational conditions.

The procedure has three search cycles. Within the first cycle, one compares the device variants and each variant differs in terms of configuration (the aerodynamic shape in the case of the descending module), the components and types of materials and fuels used. The intermediate cycle is used to select the proper operational mode and possible physical parameters of the materials, of the assemblies and fuel, in an attempt to elucidate the worst or almost worst starting data.

The internal cycle is used to consider the possible propulsion design schematics and control parameters, in an attempt to find those which would provide the most efficient design and mode of motion for each set of operational conditions (as assigned in the intermediate cycle) and in the device or module, the appearance of which /32
has been formulated in the external cycle.

If one is to utilize the terms of mathematical design search, as presented in Chapter 1, the minimax strategy of the physical appearance selection on the basis of mass criteria, ensuring the proper flight M_0, will have the following form:

$$M_0^{opt} \approx \min_{\{X_i\}} \sum_{j=1}^{n} \max_{l_k} | \min_{\substack{\{x_\xi\} \\ P \leqslant P_0}} M_j(\bar{X}, \bar{x}, \bar{l}),$$

where ℓ_K is the summary data which defines the external conditions ($\bar{\ell}_B$) and internal parameters of the system design ($\bar{\ell}_M$); M_j is the mass of operational means (the means used for descent and touchdown, the power supply systems, the thermal control systems, general control, control of the power units, general design, thermal protection systems, etc.); $\bar{X}$ is the vector of discrete or integral variables which correspond to the summary systems forming the configuration, composition, types of materials, fuel, variants of component circuitries); $\bar{x}$ is the vector of variables which reflects the size of propulsion units, the design of propulsion units, parameters of motion control, thickness of the thermal shields; P and P_0 are the values of attainable and projected probability that the permissible computed values will be achieved. Such accounting for probability reflects a rational rejection of some unrealistic and inferior operational modes.

The equality for computing M_0 is only an approximate one, since in practice one must take into account the iteration cycles and feedback relationships, as one summates the total mass of the space device systems.

In the case of antagonistic criterion, which is used to define the process effectiveness, that the device will attain the end goal, the minimax approach is applied in an analogous way. For example, the model for the activities of the designer who is concerned about the probability of fail-free operation of the onboard systems P_Σ, is also based on the same three cycles, in accordance with the following strategy

$$P_\Sigma^{(opt)} \approx \max_{\{X_i\}} \prod_{j=1}^{n} \min_{\{l_k\}} \max_{\{x_\xi\}} P_j(\bar{X}, \bar{x}, \bar{l});$$

where P_j is the probability of fail-free operation of separate systems.

For example, in any long range interplanetary module for the optimal backup and formal computation of the integral parameters, based on the methods to be used and safety levels of the instrumental general backup and priority backup - all these criteria are to be reflected in the module design. Each variant of the design composition is to be calculated several times for different combinations of the starting data which describe the standard and nonstandard operational cyclogram for the onboard systems, by employing different flight programs. By employing the worst operational modes for each system one must, in the course of such selection, generate the fail-free and optimal operational parameters for the operation of onboard systems, for the proper orientation of the module in space and for the operation of correctional propulsion units.

The other example which demonstrates the minimax approach in regard to the model which is to be selected in the course of the design work is the manned landing spacecraft, which is to be returned from the interplanetary flight to the Earth, using the hyperbolic velocity and which is to undergo the braking cycle, adhering to all of the most important operational conditions: minimization of the thermal shielding and permissible stress of the crew. One is to select the proper parameters for the shape, for the design and for the types of thermal shielding, and by using the combination of such factors, one is to /33
investigate and define the limiting conditions for such systems in the course of atmospheric entry.

For each operational mode which is being analyzed, one is to select the optimum combination of descent control with the minimum thickness of the thermal shielding.

The third example is quite characteristic for the selection of size and mass parameters as well as the power-related parameters for the descent and touchdown on Mars. The external search cycle is used to select the proper configuration and surface area of the aerodynamic braking properties of the spacecraft with adequate thrust, type of fuel and quantities of it, which is used for prelanding braking. The motion along the descent trajectories and the spacecraft touchdown, once its shape has been defined, is being investigated in the presence of the worst combination of conditions which can be found on this planet, using different models for different atmospheric density conditions. Each control motion is being optimized (by selecting the proper control approach) on the basis of minimum mass of the propulsion unit and of the fuel, employed during braking and touchdown.

CHAPTER 4.

SYNTHESIS AND MODELING OF SPACECRAFT COMPONENTS

The mathematical, informational and functional models of the /33 spacecraft components are being used in the course of structural synthesis and while investigating the operational modes and therefore, the design principles here depend on the above-mentioned tasks. We shall consider below the models which would describe the components and structural synthesis. The modeling of operational modes which is to be used with the joint modeling of the system operation will be described in the following chapter. The calculated model, in determining the parameters of the spacecraft systems, makes use of the relationships which exist between the systems and the calculations and refinement of the power which is to be consumed, the heat which will be evolved, the volumes, masses and general fuel reserves. Therefore the structure of such a model is tied in with all system calculations by employing the general multiple algorithms for the determination of power-related and size-mass parameters.

Among the systems on board, one is to isolate such systems, the functions of which are target-oriented, in other words those systems for which the spacecraft was designed.

Among these are the general aggregates of the payload, the equipment for scientific and measuring functions. In combining the systems into a separate group (the target-oriented group system) they are to occupy a special place within the model of general design. The general spacecraft mass, size, heat evolution and power consumption of the system within this group is to be handled separately, independent of other systems and of each other, and the requirements which are imposed upon these systems are aimed at the goals which are to be attained by the spacecraft, and must be formulated as the project is initiated. The goal-oriented systems impose some specific requirements on the parameters of the other systems, and together with a number of other programmed functioning operations, initiate the synthesis calculations for the systems on board. The goal-oriented systems are being supported by a set of auxiliary systems and the parameters of the latter depend both on the target-oriented systems and on each other. Among these are the sensitive components of the navigation means, the control systems, the radio communication and orientation systems, the systems for the life support of the crew, the air conditioning and power supply systems. Such auxiliary systems define the power requirements, the internal heat exchange, the size /34 of compartments, but have little effect on the general configuration of the spacecraft and its structural design.

In modeling and selecting the proper components for the majority of spacecraft, the designer has at his disposal a multiplicity of auxiliary system variants. These variants are defined by different combinations and types of systems of different purposes. The variation of different types and possible use of each of such systems is the result of competing prototypes, of assemblies, systems and subsystems.

The work of the prototypes is based on specific physical principles of functioning, or on the use of one specific fuel. The design differences may have different degrees of standardization and incorporation of standardized subassemblies.

The problems of optimal back-up and stand-by qualitative factors in the competing designs feature different levels of back-up and support by the subassemblies and subsystems, which may operate in different manners.

The spacecraft flight through the atmosphere, outside of the atmosphere, or movement on the planetary surface is provided by a group of service systems for which the target-oriented and auxiliary systems are the useful payload. The size and mass parameters, the configuration and structural design of the spacecraft, are being formed during modeling of the service system parameters, among which are the general design, the design of the propulsion units with appropriate fuel supply, the landing systems and thermal shielding.

As a rule, in synthesizing the onboard systems, the designer is to solve on a large scale the optimization problem of the assemblies and instrument parameters. He has to represent these systems in a general form as being the separate components of the spacecraft, by modeling the relationships between these components at the higher hierarchical level, and comparing the possible variants which may be used on the basis of general parameters, so that the system becomes operational. In some cases, one is to select the optimal size and mass parameters of the power supply system, of the air conditioning and other systems for viable existence of the crew, and if these systems are using common supplies of fuel, he can change the power output of these units, by varying the relationship between the weight of the fuel used and the power consumed by the assemblies which provide for optimized operational conditions. For example, in synthesizing the general component composition for the orbital or interplanetary spacecraft, one must conduct a compromise selection between the mass, power and fuel expenditure in the case of electrochemical or mechanical current generators, with the correspondingly appropriate mass of the buffer batteries, which are to function during the peak power consumption. The way to calculate the thermal control systems in such spacecraft is to determine the most advantageous combination between the expenditure of the vaporizing materials and the power output of either the compressor or heat exchange devices.

The common criteria at the upper level, while synthesizing the spacecraft components, are the mass parameters and the failure-free operational parameters (in investigating the reliability model). The correlated criteria of lower rank are the power consumption, heat evolution, the amounts of fuel, total volume, maximum size. By employing these parameters, one establishes the relationship between the parameters of the whole system and at the specific stages of the computational process, each of the above-mentioned intermediate criteria is being converted (if it is possible) into the mass parameters of either components or subassemblies which generate the power, which remove heat, which provide for placement of communication lines /35 between the spacecraft compartments.

The problem of calculating the parameters of the systems on board is an iteration problem and it can be separated into a number of sequential stages, each of which is devoted specifically to the determination of single-purpose parameters, with the appropriate recalculations of the system parameters which were calculated earlier. These stages are interconnected by the informational feedback, making it possible to refine the mass calculations, the calculations of the volumes and amounts of fuel and the calculations of the necessary power.

The synthesis procedure aims at the sequential stage-by-stage calculations and at each stage one is to calculate the parameters of the single-purpose systems, by organizing the selection among the available variants within the systems of the same purpose (taking into account the variants of the systems of different purposes which were selected at the earlier stages).

For example, in the case of one of the types of spacecraft, the system review commences with the target-oriented systems and the useful payload. Then one models the parameters of control systems and radio communications systems.

After that, one calculates the parameters for the systems which provide for viable existence of the crew. During the subsequent stage, it is convenient to consider the power supply systems, after which one should handle the air conditioning systems. The whole procedure is terminated by reviewing the systems and appropriate calculations of the size and weight of assemblies with the appropriate parameters, within the framework of the spacecraft geometry, and finally one estimates the weight and fuel supplies needed for the propulsion units and maneuvering.

At all stages, one is to calculate and adapt the total volume parameters ℓ and mass M, the total power consumed E, the evolution of summary heat Q and the amounts of fuel G.

The problem of optimal system backup is also formulated on the basis of the reliable model mass and therefore at each stage the obtained figures are multiplied by the probability factor of the failure-free operation within the assemblies and subassemblies.

The graphic procedural model of the design calculations and spacecraft composition synthesis is shown in Figure 4.1.

In the left hand side of the drawing one can see a chain of

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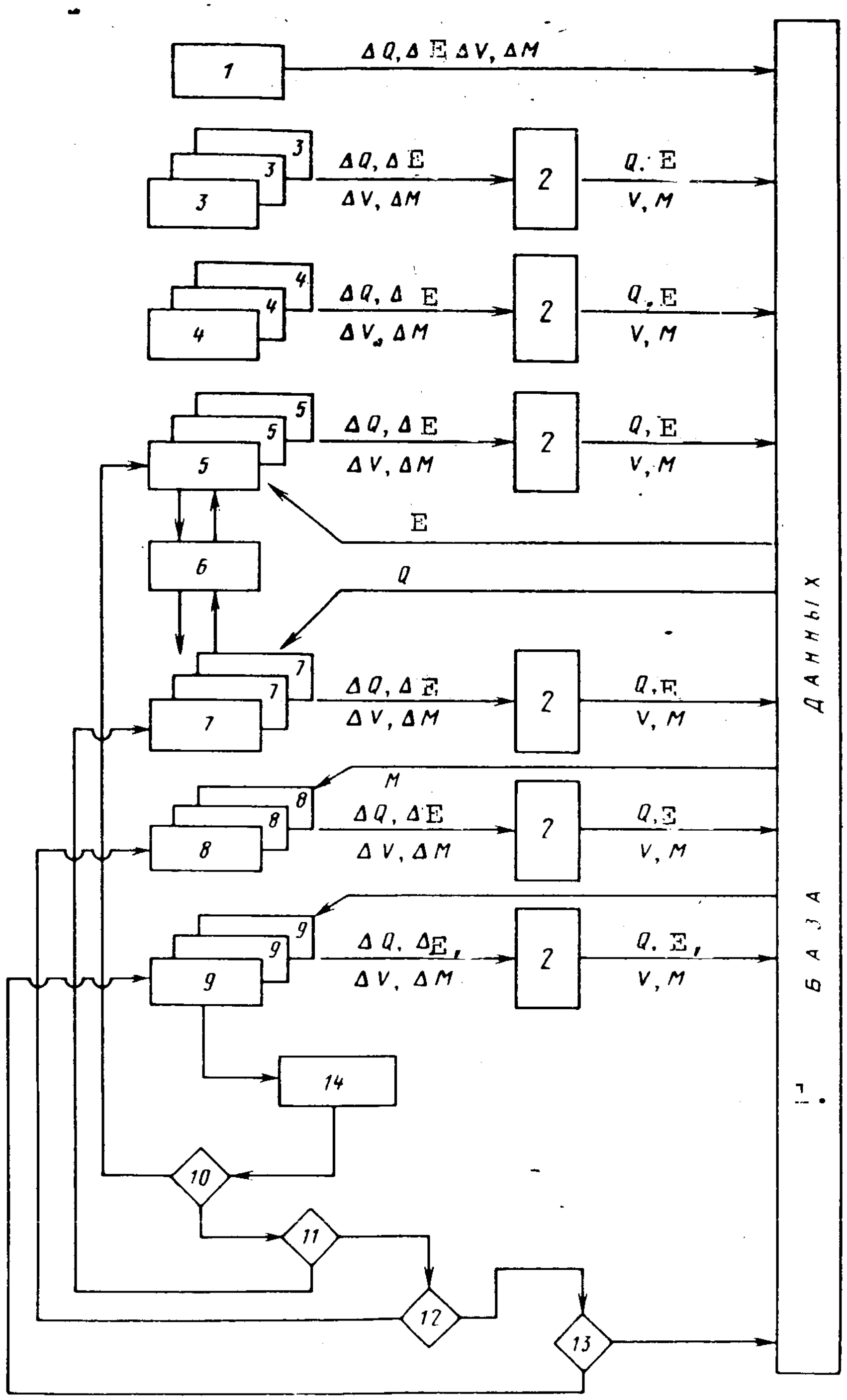

Figure 4.1. Block diagram of the algorithm which defines the synthesis of the spacecraft system composition.

Key: 1. Data base

problem-oriented modules which handle the calculations of separate systems. For each system one can see several modules which correspond to different system variants and which differ by the physical principles of functioning of the aggregate and schematic designs, having different levels and means for the operational backup in the instruments and subassemblies.

In the right hand side of the drawing one can see the calculations and synthesis of the systems on board, with a conditionally unified data base containing the informational model of the spacecraft composition.

As one proceeds with the calculation of each sequential system, one periodically upgrades, within the model, the data in regard to the size and power parameters on the basis of several current non-optimal variants which could be used in coupling several systems. The module 1 assigns the useful payload and starting data.

The module 2 models the combining of several variants within the system of the target-oriented group, coupling it to the instrumental systems which have been calculated earlier. Simultaneously, the module 2 compares several current variants of the components as projected for the assembly by combining the optimal and non-optimal variants of the assembled part, preselected at the previous stages, with the system modifications which are to be /37
coupled at this developmental stage.

The module 6 is to model the optimal balancing, while selecting the proper relationships between the buffer current source from the chemical battery and the power generators of different types (as they apply to the calculations of the general power supply system). The buffer battery provides for the peak operational modes during the time of flight periods when one cannot use solar batteries and electrochemical and radioisotope power generators. (The preparatory work prior to autonomous flight after the energy storage period on board of the main carrier, the navigational operations and trajectory corrections, the orbital flight times, while on the dark side, etc.)

The same module (module 6) is to model the situations associated with the activation of the air conditioning devices, during the peak loads of the system, which are using the fuel (water, alcohol, ammonia, ammonium chloride, etc.). For a given cyclogram of the activated systems on board, one is to select the optimal relationship between the size and mass of the air conditioning and heating devices, of the evaporating devices and compressors with the solid-liquid heat exchangers and external radiators for heat dissipation.

The module 3 calculates the control systems and radio communication systems, the systems of power consumption and heat evolution.

The module 4 develops the parameters of systems which ensure the viable existence of the crew, including the subsystems and assemblies of oxygen supply, air purification, the appropriate pressure and air composition within the compartments, the maintenance of appropriate conditions inside of the space suits, the provision with foodstuff and water, the removal of waste and maintenance of medical control. The calculation models for these subsystems develops the combination with other systems of common supplies of the fuels (of oxygen, water, hydrogen peroxide) and also develops the figures of required energy consumption, the levels of heat evolved and the general parameters of volume and weight.

The module 5 calculates the parameters of various types of systems which are used for power supplies, selecting simultaneously the optimal relationships between the buffer power storage units and constant current sources. The total volume, weight, heat evolved and power consumed by the instruments within the system itself are added to the values of analogous parameters in the other systems. The size, weight and type of selected electric power supply variant depends on the available resources, on the required peak power output and on the total mean power consumption by the onboard systems.

The module 7 is the model of the thermal control system which is employed within the general synthesis procedure of the spacecraft component composition. By obtaining the information about the total heat evolved and taking into account the capacities of its own aggregates, such calculation model system develops the requirements which are imposed on the power consumption and generates the information about the required volume, mass and stored fuel. In addition, in some extreme operational modes, in order to generate additional heat it is possible to activate either the isotope or electrochemical heating units, or vice versa, one can activate the system of the evaporating units, on the basis of which the size and quantities of fuel can be optimized.

The module 8 calculates the quantities of fuel and power required by the propulsion units which are used to orient the spacecraft and /38 maneuver it, on the basis of the operational program, on the basis of the moments of inertia and mass of all systems.

The module 9 determines the mass of the construction design for several typical design schematics, on the basis of the fuel containers required and the spacecraft compartments, computed for the appropriate storage systems for the fuel and instruments.

The iteration cycles with comparison procedures 10, 11, 12 and 13 are to refine the required power supplies (by taking into account the systems for air conditioning which are necessary for the power supplies and propulsion units themselves), they refine the amount of heat which is being evolved (taking into account also the thermal conditioning and propulsion unit instruments), they refine the calculation of the quantities and weight of fuel which will be used during the spacecraft orientation and maneuvering procedures (by taking into account the design construction mass) and also the required design construction mass (taking into account some additional volumes, occupied by the fuel tanks which provide for excess power needed and not taken into account earlier).

The last feedback closes up the iteration calculations, providing the proper convergence for solution of equations, related to the spacecraft viable design.

These equations accomplish a balanced presentation of the volumetric and size-mass model, with the information about the acceptable levels of required size of the fuel tanks and the design compartments which are to accomodate the fuel tank.

The volumetric and size-mass model which is handled by module 14, or by means of a special procedure which is incorporated within the spacecraft composition synthesis. The greatest uncertainty in the volumetric model is in the estimates of the required size of the living quarters.

In the manned spacecraft, the size of the living quarters may reach seventy percent of the total available space. In addition, a considerable quantity of design materials and weight are being expended to develop such living quarters. The existing inaccuracy in the estimates of the living quarters is inherent at the present time in the case of the spacecraft designed for autonomous operation, for example, the orbital stations for long stay in space, the interplanetary complexes and planetary laboratories. By analogy with the surface vessels, the space which is allocated per one crewman is increasing during the time of flight quite rapidly, in the beginning, slowing down at a later time and then gradually approaching the level of 8-12 m^3. The level of this space is based on the minimum space required for viable existence of one person during his short-term stay in the cabin. These standards, on the basis of various recommendations, fluctuate between 0.8 and 1.5 m^3. The level of specific space (per one crewman) also depends on the number of people within the crew because the fraction of constant space which takes into account the space required for control, for sanitation and other services, will decrease per person with the increase of the number of crewmen. Analysis of standard space as recommended in the other branches of industry, and also some generalized data about the space parameters which were developed within the framework of the spacecraft design studies, recommend the following formula:

$$V_{sp} = 1{,}45\left(\frac{T}{n}\right)^{0{,}3} + 0{,}8.$$

The formula may be used to estimate the required free space available for the living quarters (in m^3/man) with the accuracy of ±30% when the crew consists of three men and flight duration is less than 400-600 days and nights.

The space volume parameters required for instrumentation in the instrument compartments and at the control areas are ordinarily defined on the basis of the specific densities ρ_{red}, which is ordinarily defined by taking into account the free area in the vicinity of the instrument's various supports, the equipment junction point between the instrument and other communication means. /39

The instruments on board traditionally are subdivided into three groups in terms of the specific density. The first group includes the heavy, or densely packed assemblies such as the fuel tanks (except for the light, cryogenic fuel), the propulsion unit assemblies, the units for balancing control, the radiation shielding components and the high temperature active and passive heat shielding, the compartments with parachute systems, the radioisotope current and heat sources, the buffer batteries, optical instruments and light sources. The specific density of these assemblies exceeds 0.4-0.5 t/m^3, reaching 0.8-1 t/m^3.

The second group includes primarily the electron instruments, the electromechanical assemblies within the control systems, the systems of radio communication, air conditioning, electric power supply, thermal control and also the read-off and executive devices at the control panels.

The specific density of instruments and assemblies within this group, taking into account the areas of their placement, fluctuate between 0.2-0.4 t/m^3.

The third group includes the light components, with $\rho_{red} < 0.2$ t/m^3. These include the inflatable and collapsable sections (reducers and sluices) containers with storage of light cryogenic fuel, components of mechanical assemblies, requiring space for storage (collapsable antennas, solar batteries, manipulators, thermal control radiators and enclosures with the hydropneumatic lines and electric conduits).

In a general case, the volumetric formula of the spacecraft will have the following form

$$V_u = K_{des}\left(V_{sp} n_{cr} + M_{p\ell}\frac{1}{\rho_{p\ell}} + \sum_i \frac{M_i}{\rho_{red\,i}} + \sum_j \frac{M_j}{\rho_j} + \Delta V_{des}\right),$$

where V_u is the total useful volume; K_{des} is the coefficient which takes into account the desired volume, occupied by the design components (by the power-related components, thermal insulation, the spacecraft enclosure); n_{cr} is the number of crew members; $M_{p\ell}$ and ρ_{red} are the mass and specific density of the interior sections in which the payload is located, M_i and ρ_i are the mass and specific density of instruments and assemblies on board, M_j and ρ_j are the mass and specific density of fuel compartments, ΔV_{des} is the size of special compartments within the design (reducers between the fuel tanks, the air intake lines, etc.).

The typical mass model of the spacecraft combines the payload mass ($M_{p\ell}$), the mass of life-sustaining system ($M_{\ell ss}$), the mass of control assemblies (M_{PSS}), the mass of the control units on board (M_{onb}), the mass of general construction design (M_{des}), the mass of fuel (M_f), the mass of propulsion units (M_{pu}), the mass of landing or touchdown systems (M_{land}) and in determining the total mass M_Σ, can be ordinarily written in the following form

$$M_\Sigma = M_{p\ell} + M_{\ell ss} + M_c + M_{rad} + M_{ac} + M_{PSS} + M_{onb} + M_{des} + M_f + M_{pu} + M_{land}.$$

This commonly accepted form of expressing the mass may result in an erroneous conclusion in regards to the mechanism of the mass calculation. The mass function, if one is to judge it on the basis of its construction, appears to be separable and additive function, in other words it appears to be consisting of a number of independent components which are being added. Even some studies related to the optimization of the spacecraft design parameters have been conducted by assuming that the mass criteria are additive. If the sum of design parameters of each subsystem is to be designated by x_i, then the complete formula in determining the mass will be represented as a function for calculation of the mass of separate systems, namely: /40

$$\begin{aligned} M_\Sigma = {} & f_{p\ell}(\bar{x}_{p\ell}) + f_{LSS}(\bar{x}_{p\ell}, \bar{x}_{LSS}) + f_c(\bar{x}_c, \bar{x}_{p\ell}) + \\ & + f_{rad}(\bar{x}_{rad}, \bar{x}_{p\ell}) + f_{TCS}(\bar{x}_{p\ell}, \bar{x}_{LSS}, \bar{x}_{TSC}, \bar{x}_c, \bar{x}_{rad}, \\ & \bar{x}_{PSS}, \bar{x}_{onb}) + f_{PSS}(\bar{x}_{p\ell}, \bar{x}_{LSS}, \bar{x}_{TCS}, \bar{x}_c, \bar{x}_{rad}, \bar{x}_{PSS}, \\ & \bar{x}_{onb}) + f_{onb}(\bar{x}_{p\ell}, \bar{x}_{LSS}, \bar{x}_{TSC}, \bar{x}_c, \bar{x}_{rad}, \bar{x}_{PSS}, \bar{x}_{onb}) + \\ & + f_{des}(\bar{x}_{p\ell}, \bar{x}_c, \bar{x}_{rad}, \bar{x}_{LSS}, \bar{x}_{TCS}, \bar{x}_{PSS}, \bar{x}_{onb}, \bar{x}_{pu}, \bar{x}_f, \\ & \bar{x}_{land}) + f_f(\bar{x}_{p\ell}, \bar{x}_c, \bar{x}_{rad}, \bar{x}_{LSS}, \bar{x}_{TCS}, \bar{x}_{PSS}, \bar{x}_{onb}, \bar{x}_{pu}, \\ & \bar{x}_f, \bar{x}_{land}) + f_{pu}(\bar{x}_{p\ell}, \bar{x}_c, \bar{x}_{rad}, \bar{x}_{LSS}, \bar{x}_{TCS}, \bar{x}_{PSS}, \bar{x}_{onb}, \\ & \bar{x}_{pu}, \bar{x}_f, \bar{x}_{land}) + f_{land}(\bar{x}_{p\ell}, \bar{x}_c, \bar{x}_{rad}, \bar{x}_{LSS}, \bar{x}_{TCS}, \bar{x}_{PSS}, \bar{x}_{onb}, \\ & \bar{x}_{des}, \bar{x}_{pu}, \bar{x}_f, \bar{x}_{land}). \end{aligned}$$

The mass of systems of one purpose as a function of the parameters of the systems of another purpose is not something new and it is familiar to the designers who reject the algorithms of linear or dynamic programming which are being proposed to optimize the components in spacecraft.

The sequential list of presented masses in the formula is not a random one. It reflects an accepted order in calculating various modules which are responsible for generating the parameters in each of the above mentioned systems. In having such precise sequential order, the amount of feedback data which refines the size, mass and power parameters, is at a minimum but even having such sequential order, it would be funamentally wrong to isolate each separate module in calculating the mass of the best variant within each system, so as to connect it with the sequential system for the best set of modules in the spacecraft.

The systems connected with the common fuel storage and which differ in terms of variants have different volume requirements which dictate the change in the mass of separate variants within the systems which are next to the best in terms of their parameters. For example, the parameters of the power propulsion unit masses or the weight of propulsion units for spacecraft orientation depend on the selection of the type of control systems and on their parameters. The assemblies, selected in terms of the optimal mass, associated with the air conditioning, with oxygen supply and with air regeneration, are as a rule not the best when coupled to power supply and air conditioning systems which require large expenditure of fuel, and have excess of energy in the assemblies to compensate for large quantity of electric power consumption and heat evolution which manifest themselves in compact and light life-sustaining devices.

For analogous reasons, the power supply system variants which are /41
large in volume and mass and which utilize for example the fuel components, turbine generators or diesel generators may turn out to be, unexpectedly, not very advantageous when coupled to inefficient separate life-sustaining systems and system of thermal control which utilize the exhaust products (water, ammonia, etc.) or when coupled to the propulsion units which utilize the same standard fuel. In employing the gyroscopes for control and orientation of some spacecraft, it is reasonable to stay away from gas flow stabilization and orientation devices. In such case, there will be no need to correlate the propulsion systems with the control parameters, but the required power output will be increased and consequently, the total weight of the power supply assemblies which we have considered earlier will also have to be increased. The radioisotope current sources which are used to heat the spacecraft compartments are characterized by low power yield per each kilogram of its mass. They make it possible to lower the fuel mass, in employing the periodic orientation by using solar batteries. The gas propulsion system of prelanding braking during the planetary touchdown loses, in terms of its mass, when compared to mechanical energy quenchers, incorporated in the landing

supports, but may be used for take-off by changing the parameters of orientation and control systems. By employing radar assemblies with large antenna diameters, one can decrease the mass of the power systems and increase the cross-sectional size of the enclosure, but this is associated with larger fuel expenditure, increase in the required thrust during travel and lower expenditures of fuel during the pretouchdown maneuvering, the correction trajectory maneuvering or approach maneuvering.

The heavy and energy-consuming regenerative assemblies for the life-sustaining systems in the future spacecraft for the short interplanetary flights are inferior to systems which utilize stored oxygen, water, foodstuff and absorbers of detrimental matter, but may turn out to be quite acceptable if one considers a considerable volume of stored and exhaust materials, since the increase in the volume also implies an increase of the size of the compartments and consequently of the total mass of the fuel and construction design. The standby and ready-to-operate instruments of the control systems increases their mass, with the concurrent increase in the mass of the power supply units, of the units of thermal control and in the instruments in the compartments. However, such increase may turn out not to be justified in the case of power propulsion units which utilize auxiliary fuel supplies for the correction of trajectories, associated with the malfunction in control systems.

The comprehensive approach in selecting the composition of the spacecraft systems requires an all-around accounting of all interactions within a singular automated procedure of structural synthesis, aimed at specialized methods and algorithms of the directional variant selection.

The integral parameters x_i which may be varied in such methods are correlated with each specific i component, within the structural composition (in the systems, assemblies, hierarchic separation levels, types of materials, fuels and propulsion design systems). The change of each i parameter from 1 to k_i indicates the periodic replacement of the variants within the system which is used for the same purpose and may even be associated with the change of the whole algorithm /42
which is used to calculate the spacecraft parameters.

As one proceeds from j parameter to (j+1) parameter, in other words as one proceeds with consideration of one system variant vs. another system variant of the same purpose, the change in the power and mass parameters which has not been planned (even in the presence of fixed variants of the other systems) is quite likely, which makes the composition synthesis problem rather multiextremal, excluding the possibility of organized preselection and prognostication of the behavior of goal-oriented functions, which are to be used to calculate the mass of the spacecraft.

The setting up and mathematical treatment of the typical problem in synthesizing the composition and structure of the spacecraft as a nonlinear integral programming is presented in the study [45]. The same study classifies the methods and algorithms which can be used to solve the problems of structural synthesis.

Let us note the common fundamental differences in these methods from the classical numerical optimization methods which have been developed for the nonlinear programming with continuous variables and unimodel goal function.

The basic differences are as follows:

utilization of the man-computer procedures to sort out the variants;

rejection of attempts to find the best variant in terms of each criterion;

instead of searching for the absolute extrema, one is to find the multitude of less then optimal variants;

evaluation of the levels of criteria for optimally best and worst variants. The estimate is being conducted here on the basis of statistical theory of extremal values (see Chapter 5);

replacement of the step principle in advancing the parameters with organized or stochastic incremental increase in the variables by the zonal survey of the multitude of variants, using the group tests which apply to all multiple structures which are being investigated;

organization of stochastic procedures in surveying the multitude of structural variants with the possibility of probable estimate of efficient results in selecting the best solutions and simultaneous prognostication of the required time expenditure in finding the variants which are superior to those which have been selected.

The concrete realization of such principles in the algorithms of structural synthesis may be accomplished by the known approaches employing stage-by-stage cyclic sorting of variants during the synthesis process itself, investigating composition or structure (using the concepts of dynamic programming), the employment of group random sampling coupled to the algorithm of statistical estimates of extremal values or the coordinate-by-coordinate search for the best variant, refined by the procedure of complete variant sampling in the Hemming vicinity, in the domain of parameters around the points which correspond to the less than optimal variant or to the pareto-optimal solutions.

The questions of convergence and efficiency of different algorithms which are used for structural synthesis, as well as the approaches of using them in the optimization problems, pertaining to the spacecraft composition and components of specific systems of a spacecraft have been considered in the studies [45, 48] which we recommend to the reader who is interested in the engineering handling of applied problems associated with nonlinear discrete programming.

CHAPTER 5.

PRINCIPLES AND EXAMPLES OF SIMULATED MODELING OF THE SPACECRAFT FUNCTIONING PROCESSES

The simulated modeling is being used quite extensively in developing the project assignment (in the course of the "external" design phase work) and to substantiate the assumptions that the device, with the selected design parameters, will function properly. /41

In the first case, one utilizes approximate models of motion and system interaction and in the second case - the detailed calculations of flight control, with modeling of the gas dynamic flows, heat exchange and using unit-by-unit functional schematic diagrams and operational cyclograms for the onboard systems.

The simulated modeling, with the advance of the electron-mechanical analogues and instrumental and assembly prototypes is being replaced by simulated service tests on stands and experimental test sites.

The task of simulation modeling is as follows: by using the ready-made scenario or by employing random interferences using the operational conditions under different circumstances, to study the effect of the environment, the deviations from the standard parameters of the design materials, of the fuels, of the sensitive control components during each stage of the flight.

By using the simulated modeling data, one can refine the design, one corrects various aspects of the design project and various technical aspects of development in regard to the specific operational systems.

The task of simulated modeling is quite specific for each type of spacecraft and therefore the way the problem is set up and the preparatory work related to the starting data and its variation and consequently, the prognostication of the future flight situations, depend on the qualifications, creative imagination and level of the know-how of the research engineers.

In terms of the general approach to the design problem (see Chapter 1), each step of simulated modeling is multiple testing of actions and responses:

$$\overline{\ell}_A, \overline{\ell}_M \rightarrow \overline{FK}, \overline{FG}.$$

As a rule, the major operational mode, while simulating the operation, is a periodic consideration of fligth situations with either random or goal-oriented assignment of the maximum limiting deviation of the starting data which reflect the external and internal conditions.

One ordinarily makes use of the statistical modeling methods in handling the initial data which results in the stochastic game assignments with corresponding processing of the results pertaining to the random tests.

The need for a multiple, straight forward functioning operational mode is associated with the fact that the goal functions which are being realized in the algorithms and computer programs, after a number of calculation processes, allow for the reaction to the change in the initial data but do not allow for the reproduction of the general picture of the whole operational process, with the starting action $\bar{\ell}_A$ and $\bar{\ell}_M$ which is based on the flight results (on the basis of $\overline{FK}$ and $\overline{FG}$).

The analytical approach in the study and generalization of the load, heat exchange and motion modes is frequently counterpoised by using methods of simple engineering testing with the computer time overload and chaotic piling up of bulky data and random testing. However, the method of statistical or sampling modeling is used more and more frequently, in spite of the probabalistic nature of the obtained results. The design engineers are attracted by unlimited possibilities to review the flight situations, with more and more complex logic and computational functional model, taking into account the data feedback and combining the problems of loads, motion control and the assembly and instrument functioning.

The main design requirements which dictate the use of simulated modeling are associated with the ability to:

prognosticate the possible flight situations, taking into account the response of the design systems which are used to control the spacecraft;

analyze the limiting computed loads and actions to which the spacecraft is exposed;

obtain and train the experts and special services, who are to handle the future flight of the spacecraft which are being designed.

The usefulness of training by employing the simulated modeling has been substantiated by concrete cases from space flight practice. For example, during an emergency situation on the Apollo-13 spacecraft, when during the lunar approach, the systems of power supply and the life-sustaining systems have failed because of the oxygen tank explosion, the ground computing center, jointly with the astronauts, has developed the suggestions and proper measures which were to be taken, so that the spacecraft could return to Earth, altering the operational program of the systems on board.

We shall present below the known advantages of simulated modeling, as compared to tests on the stands or during flight conditions:

the simplicity of obtaining the limiting flight conditions, with the investigation of the "destruction" of the object which is being designed;

the ability to reproduce accurately the flight conditions in the course of a repeat experiment;

the ability to discontinue and then start again the process which is being modeled;

the ability to investigate the phenomenon during the short time and brief intervals by employing the time delay;

the operational efficiency of experimental calculations which are independent of the readiness of the experimental stands, of the weather conditions and the situation with the service personnel.

The shortcomings of simulated modeling are the extensive computer time, because of the considerable number of required operational steps and the appreciable length of each experiment.

One can well understand the arguments of the research designers: "once we reject the analytical relationships with the idealized simplifications and allowances, and once we embark on a direct reproduction of the operational processes, it is reasonable to provide for modeling of physical phenomena, preserving the complexity, the number of relationships and equations, so that one could introduce the numerical modeling calculations of several processes, together /45
with precise integration with respect to time and space.

Let us present the examples of simulated modeling for the operational spacecraft:

1. The analysis of each mass exchange and motion within the planetary atmosphere, during the orbital descent or the descent from the approach trajectory.

2. Study of the planetary landing dynamics during the stage of prelanding braking and touchdown.

3. Study of interaction between the orbital spacecraft and the modules above the planet which gather the information about the planet.

4. Computed modeling of gas dynamic flow parameters, taking into account the physical and chemical processes within the impact and boundary layer.

5. Modeling of cargo delivery systems for the orbital stations.

6. Determination of deformations and transitory dynamic processes in the elastic structures, containing fuel tanks, in the course of evolutions during the trajectory control.

7. Analysis and visualization by using the computer graphics of the kinematics and dynamics of planetary vehicles while in motion.

8. Reproduction by calculations and by means of screen displays of the interaction between the separating spacecraft and modules or the modules and spacecraft at the moment of docking, having the flexible, deforming connections.

9. Study of the dynamics of the module orientation, trajectory corrections and orbital maneuvering, to develop the working orbit and transfer of the module from the landing to flight trajectory.

One of the major advantages in using the method of statistical testing is the ability to evaluate during a finite time period the worst parameter values, inherent in an infinite number of flight situations. This evaluation is of probabilistic nature and in determining the level of extremal external actions, it is reasonable to use the statistical theory method for extremal values.

The evaluation of limiting parameter values is based on the sum of separate particular extrema. In order to obtain them, one is to organize the random group samplings, with the selection of the worst maximum values within the groups $FG_{\xi}^{(k)}$.

In such case, by using the statistical modeling of particular maximum values and employing the third exponential distribution, the level of maximum attainable figure $FG_{\xi}^{(max)}$ is defined on the basis of the distribution function

$$\Phi(FG_{\xi}) = \exp\left[\left(\frac{FG_{\xi}^{(max)} - FG_{\xi}^{(k)}}{\sigma}\right)^{\eta}\right],$$

where σ is the root-mean-square deviation of the particular maxima, η is the factor which corresponds to the shape of the distribution law.

The above-mentioned parameters and the magnitude of the absolute extremum $FG_{\xi}^{(max)}$ is determined by the method of moments, simultaneously with the procedure of statistical testing.

At the present time, the large expenditures of computer time necessary for a thorough statistical modeling (at high confidence level in evaluating the limiting parameter values) slows down the development of the methods of simulated modeling. The alternatives are the tests on experimental stands. /46

One can project that with the appearance of large computers and development of thorough computing models, with the decrease of cost of the computer time, the increase in the number of projects which are to be investigated and greater complexity of specific tests, the gradual replacements of the tests on stands by the computational, simulation procedures is almost unavoidable.

The computational modeling is capable to investigate (albeit, by sampling) a large number of operational modes). The perturbation in regard to the starting data (the factors of environment, flight conditions, system parameters, materials and fuels) may be described in a different fashion, embracing all possible ranges of the data change. For example, the uniform distribution is quite characteristic to prognosticate the variations in the flight programs in the case of multipurpose orbital research spacecraft. Or, one can calculate the planetary approach velocity on the basis of a specific lift-off time. Again, it is also possible to calculate the weight-to-lift ratio in the case of the specific booster systems.

Such random factors as the position inside of the atmospheric entry corridor, the parameters of materials which are used in the design and in thermal shielding, the information about the composition and density of the planetary atmosphere, the data about its relief and soil, the speed and direction of the wind, the pressure and temperature data, the acrued operational instrumental time before failure, the characteristic speed of corrections and of the trajectory and orbital switch-overs are distributed with the greatest density of probabilities near the middle values, making it possible to use the canonic distributions and in the simplest case, to employ ordinary or truncated normal distribution law for the starting data used.

Some starting data, which strongly affect the characteristics of the flight process, are characterized by the asymmetric distribution laws in the probability densities. Among these are the thrust and specific pulses associated with the propulsion units and fuels which are being developed, the information about presumed accuracy of the navigation and radar instrumentation, the thermal-physical indicators of the heat shielding efficiency. The modeling of distribution of such data is based on expert opinion. The whole range of values for the quantity in question is being split into intervals and the estimates within each of the intervals are developed.

The expert opinion must be generalized by using the assumptions of the very same experts as to the nearness of their evaluation to the levels of average values. In the case of Delfa method [72] the experts are given the mathematicl expectation and two quartiles. The quartiles and mathematical expectation split the range into four parts, each containing equal number of estimates. After several iterations, the expert opinion changes only slightly and one could use the approximation methods for the distribution function in order to introduce the information into the statistical modeling of the deviations in the parameters.

Let us first consider the separate methods of modeling the spacecraft parameters, the modes of their motion and system functioning and then cite the examples of simulated modeling of the flight and touchdown. /47

The presented modeling methods include the description of the heat shielding removal, with the thermal flows which are neutralized by the destruction products of the shielding itself, they take into account the effect of integral overload factors to which the crew is exposed and the reliability analysis of the mass model of the onboard system functioning by employing different methods and operational levels of standby.

The studies [15 and 45] consider the questions of optimization of thermal shielding in the case of the future descending space modules. The simulated mechanism of the heat shielding coating removal must be present even in the very first design estimates of the descent trajectories in the case of such devices, because the quantities of the coating which is being removed control the intensity of its decomposition and the mass of removed thermal shielding must be compared to the mass of the design itself or the spacecraft payload.

The total mass of thermal shielding in the case of the spacecraft which return to the Earth from interplanetary flights and which enter the atmosphere with hyperbolic velocity on the order of 17 km/s, is about 25 percent of the total mass of the spacecarft itself and in the case of a module which descends through the atmosphere of Jupiter, the thermal shielding mass and thermal insulation reaches one half of the module total mass.

In selecting the configuration of the above-mentioned modules, by using a computer, one encounters one of the most complex and interesting problems which involves the optimization of the mass of thermal shielding which is being removed which is ten percent of the total mass of the spacecraft which return to the Earth and more than a quarter of the total module mass in the case of the atmosphere of Jupiter. The shape of these devices is defined on the basis of minimum mass of thermal shielding, keeping in mind the requirements in terms of the operational and braking parameters which ensure the acceptable overload mode.

The thermal shielding of descending devices which enter the atmosphere with hyperbolic velocities is exposed to intensive radiation thermal flows, associated with the contact glow layer, and the thermal flows which are associated with the heat transfer within the layer through the wall. The maximum magnitude of these flows reaches 8000 kcal/$m^2 \cdot s$ (during the entry into the Earth atmosphere) and 40,000 kcal/$m^2 \cdot s$ (during the entry into Jupiter's atmosphere).

The spacecraft as designed is protected because of the heat absorption during the destruction of the external layers and the thermal resistivity of the nondestructable, internal layers, which are heated through and which accumulate the heat, transferring it to the cabin shell.

The intense decomposition of the thermal shielding absorbs the heat of the incoming flow, compressed by the impact layer. This heat is absorbed, phase-converting the coating materials, it is absorbed by erosion and by the linear removal of the external layer as well as by the vaporization of the binding material, through the layer of carbonized filler. The more vapors and particles of destroyed material impact the wall coating, the better is the radiation shielding from the impact layer (in accordance with the absorption spectrum in the visible and UV ranges). Analogously, the more intense is the entry /48 at the boundary layer of relatively colder decomposition products of the thermal shielding, the better they will block the heat transfer by the gas heated by the shock wave (in the course of intensce compression and high temperatures near the wall). All these facts make it mandatory to have a good feedback within the design mechanism of all thermal shielding calculations. The calculations of the coating removal, which are conducted disregarding the attenuation of the thermal flows by the layers of the thermal shielding material destruction products, will produce under the above-mentioned conditions the entry error on the order of 1.5-2. On the other hand, the accurate calculation of all effects of radiation screening and blocking of the convective thermal flows from the impact layer is associated with time-consuming thermal-physical calculations of many hours, involving the numerical modeling of the flow fields within the boundary layer. Such modeling calculations are possible if one utilizes the high efficiency computers and employing the specific and typical combinations of altitudes, flight velocities, chemical composition of gaseous mixtures, geometric parameters of the cone blunting and the frontal angular flow changes. The specific functioning of thermal shielding material is taken into account while calculating the kinetics of the thermochemical reactions which accompany the dissociation and recombination of the molecules within the pyrolized gas which forms at the wall as a mixture of the incoming flows and products of coating decomposition. The tasks of simulated modeling aim precisely at such studies. However, in the case of design calculations, one could utilize the engineering methods which are based on approximation of the accurate analysis results, made selectively for typical combination of gas-dynamics and trajectory parameters.

The majority of such engineering methods utilize the concept of the degree of thermal flow attenuation ψ and the degree of radiation screening and blocking of the convective thermal flows is assigned, depending on the differences between the relationships which define the amounts of coating which is being removed and the mass within the incoming flow $\rho_\infty V_\infty$.

The formulas which are presented below approximate the results of the accurate calculations, conducted for the case when the phenolic hydrogen was injected into the boundary layer. The effective enthalpy in this case is assigned on the basis of limiting conditions of heat absorption during all phase conversions of the thermal shielding material.

The degree of attenuation (of blocking) of the convective thermal flows is estimated by using the following formula:

$$\psi_c = \frac{1{,}5}{1 + 38\dfrac{\dot{m}\, k_m \cos\vartheta}{\rho_\infty V_\infty}},$$

where θ is the inclination angle of the local normal to the flow, k_m is the coefficient which takes into account the percent quantities of binders within the coating which is being expended.

The proposed formula produces the increase of convective thermal flows with small expenditures of the coating which corresponds to the general physical picture of the whole phenomenon, since in the presence of small injections of coating vapors, the fraction of their cooling capacity is small and the turbilization at the wall area increases, facilitating the mingling within the boundary layer and the intensive thermal conductance toward the external frontal area of the thermal shielding destruction. /49

The following formula is being proposed to determine the degree of attenuation of the screen radiation thermal flows:

$$\psi_r = 0{,}39\,(1 + \psi_c) = 0{,}39 + \frac{0{,}585}{1 + 38\dfrac{\dot{m}\, k_m \cos\vartheta}{\rho_\infty V_\infty}}.$$

With the strong entry of vapors and particles of the material which is being destroyed into the area between the filler density increase and external frontal removal of the filler, the radiant heating from the impacted layer and the glow gas behind the impact layer are not completely attenuated. This fact is reflected in the presented formula, obtained for the case when the screening is accomplished by the phenolic hydrogen vapors, absorbing only part of the radiant energy, in accordance with the spectral absorption range.

These types of relationships correspond to the conditions for obtaining the analytical expression, associated with the expenditure of the coating mass. The coefficients are selected on the basis of the minimum of sums of squares of deviations at the nodal points, obtained by accurate calculations. The automated search for minimum is conducted by the method of random sampling.

The equation for estimate of the expenditure of the coating mass has the following form

$$g\dot{m} = \frac{\psi_r A q_r + \psi_c q_c - \varepsilon \sigma T_r^4}{I_{eff}},$$

where A is the coefficient of reflection of the radiation flows by the coated surface, ε is the degree of blackness of the coated surface, σ is Stefan - Boltzmann constant, T_d is the operational temperature of the external layer of the coating which is being decomposed and g is the force of gravity.

By solving jointly the equation of the mass expenditure and the equations which are used to estimate the degree of thermal flow attenuation, we will obtain the summary relationship which takes into account the feedback data during the design estimates of the thermal shielding mass removal.

The final solution will have the following form

$$\dot{m} = \frac{\sqrt{a^2 + b^2} \quad a}{76 I_{eff} g k_m \cos\vartheta},$$

where $a = I_{eff}\, \rho_\infty V_\infty \quad 14 A q_r k_m \cos\vartheta + 38 \epsilon\sigma T_d^4 k_m \cos\vartheta;$
$b = 152 I_{eff}\, g \rho_\infty V_\infty k_m \cos\vartheta\,(0{,}975 A q_r + 1{,}5 q_c \quad \epsilon\sigma T_d^4).$

In selecting the shape of the spacecraft by using the criterion of the thermal shielding mass minimization, the design calculations must be conducted on the basis of the limiting operational modes of motion through the atmosphere, based on the preservation of acceptable overload and overstress. This question is particularly important in providing the permissible integral overload to which the crew members are exposed, who are somewhat weakened during the weightlessness stay, and who are exposed to considerable and lengthy stresses as the spacecraft enters the Earth's atmosphere with hyperbolic velocity.

On the basis of experimental data, in the case of the overload or overstress n, there is a limiting time τ_{lim} in the case of the /50
stress to which a well-seasoned and fresh astronaut is exposed and a fatigued human being is exposed.

The limiting pulse of action will be defined as follows for n overstress:

$$I_{lim}=n\tau_{lim}(n).$$

The limiting pulse is one of the most important factors of integral stress action. The studies [15] and [45] show approximate $I_{lim}(n)$ relationships for a well trained and fresh crew and for the crew weakened by the period of weightlessness.

However, such action pulse should not be viewed separately from the other factors of overload and stress. Under the circumstances, when the overload magnitude or the gradients of its action $dn/d\tau$ approaches the limiting values for short duration permissibility, without irreversible changes in the organism, the comprehensive integral overload criterion must take into account the interaction between the limiting accelerations and the pulse activities. The accumulated action of the overload factors and the early history of the load application, in its simplest case, may be taken into account by lowering the threshold pulse action, on the assumption that earlier the organism was already loaded by the forces of inertia and was exposed earlier to the gradient increase of such forces.

At the moment of atmospheric entry, the maximum action of the gradient $(dn/d\tau)_{max}$ is compared to the limiting and permissible $(dn/d\tau)_{lim}$, after which one develops the measure of the gradient action

$$M_{gr}=\left(\frac{dn}{d\tau}\right)_{max}/\left(\frac{dn}{d\tau}\right)_{lim}.$$

If $M_{gr}>1$ (which in reality is unlikely) the measurement of action related to the subsequent overload factors loses its meaning. When $M_{gr}<1$ the measure of residual viability will be defined as

$$M_v^{(1)}=1-M_{gr}.$$

The dimensionless criterion of the action of intertia forces (the measure of maximum overload action) n_{max} is determined analogously, on the basis of the following relationship

$$M_o=\frac{n_{max}}{n_{lim}\cdot k_{att}},$$

where n_{lim} is the limiting value of the singular and short term force of inertia which does not result in irreversible changes in the organism, k_{att} is the coefficient which takes into account the change (attenuation of the limiting value of the force of inertia because of the preliminary loading of it by the gradient increase.

In the simplest case $k_{att}=M_v^{(1)}$.

After the passage of the overload peak, the organism is weakened and the measure of its weakening, as well as the degree of being ready to perceive and accumulate another overload pulse, should be taken into account by using the measure of residual viability, after the first overload peak: /51

$$M_v^{(2)}=1-M_o.$$

The limiting pulse of integral overload action must be corrected by taking into account the loads at the moment of the peak action. The degree of this correction is difficult to determine, but approximately, it may be assumed that the limiting magnitude of the pulse decreases in accordance with the lowering of the residual viability of the organism, in other words

$$I_{lim}^{(3)}=I_{lim}M_v^{(2)}.$$

In the case of the organism which is weakened by the stay in weightless environment, one should introduce the correction coefficient k_w which decreases the magnitude of the limiting pulse action

$$I_{lim}^{(4)}=I_{lim}M_v^{(2)}k_w.$$

The value of k_w obtained in the study [15] is

$$k_w=(I-\frac{455}{I_{lim}})\cdot 0.852.$$

During the braking process on the return trajectory and a short time interval $\Delta\tau$, when the action of the constant overload n is approximately constant, the pulse of integral action is defined as a product of the overload and time and then the dosage of the limiting integral action may be represented in the following manner

$$d=\frac{n\Delta\tau}{I^4_{lim(4)}}.$$

If the load or stress is variable, by integrating the dosage along the trajectory and taking into account its transformation at the moment of the passage throught the first overload peak, we will obtain

$$M = \int_0^T \frac{n}{l_{lim}(n)\left(1 - \frac{455}{T_{lim}(n)}\right)0{,}852\left\{1 - \frac{n_{max}}{n_{lim}\left[1 - \left(\frac{dn}{d\tau}\right)_{max} / \left(\frac{dn}{d\tau}\right)_{lim}\right]}\right\}}\, d\tau .$$

If the obtained dimensionless quantity of the integral overload action (the measure of effect) is less than unity, one may assume that the crew bears the return in a satisfactory manner.

Such approach in constructing the indicators which define the overload action is a simplified and mechanical one, but it may be used in design calculations, it does not require any extensive computer time in searching automatically for the optimal descent trajectories. The specifics of project calculations in such problems, the description of variable parameters and constraints which are to be taken into account, and also the statistical models of complex activity of the load or stress factors on each crew member, which are close to the actual conditions, are presented in the special monographs which define the specific points of spacecraft design.

Let us consider the models which are used to analyze the reliability of the onboard systems at the initial phase of design and the minimum mass of the device. /52

The reliable mass models of the onboard systems which are used in the problems of optimal backup are universal. They are applicable for the design work on the spacecraft of different types.

The typical reliability model describes the probability of faultless operation of separate instruments and subsystems by employing the following assumptions:

1. The component base of all subsystems reflects the current level of technology.

2. The spacecraft is essentially composed of macroblocks or components, the average acrued operational time of which, before it fails, $T_{av\,i}$ or the frequency of failure λ_i are known as well as the average time required for repair $\tau_{av\,i}$ and frequency of repair ν_i. The question of "internal" backup redundancy of each of such macroblocks is not being considered.

3. It is assumed that the frequency of failures and repairs and also the reliability of on and off operations of the macroblocks R_i, of standard and emergency type, are known to have a high confidence factor which has been substantiated by numerous tests.

The essence of work with the reliability-mass model in synthesizing the spacecraft system components involves the determination of a set of variants with limiting compromise relationships between the mass of the backup systems and reliability of their operation. The problem illustrates clearly the general engineering approach in working with two antagonistic criteria, which we have considered in Chapter 3. The goal function reflects the probability of faultless operation, probability that the flight program will be executed or the probability of safe end of the flight. The mass of standby systems ascertains the design perfection. Periodically, by using one of the antagonistic criteria as a limitation or constraint and by considering several variants with different levels and ways of employing the backup systems, one can find the limiting pareto-optimal solutions, each of which will have a minimal mass for a fixed level of reliability. The approach for such selection are the numerical methods of structural synthesis and the calculation of the onboard system mass is conducted by taking into account the mutual effect of such masses, the examples of which were presented above.

In the calculations of the faultless operation probability, one makes use of the block-by-block functional diagrams with the system detail with respect to the devices, instruments and blocks.

The probability of faultless operation (disregarding the relationships with the other blocks) in such standard instrument (or block) may be expressed in the following manner:

$$P_i = R_i \exp(-k_{res} \lambda \tau_\Sigma),$$

where τ_Σ is the flight time, k_{res} is the ratio of the actual time in operational mode to the total flight time and R_i is the probability of successful activation.

The study [45] includes the formulas for typical situations in which one is to evaluate the probability of faultless operation of several systems in having the ready standby equipment, or the probability of failure of certain parts of simultaneously operating /53 aggregates, as well as the probability of successful operation of all systems which are the "cold reserve" redundant systems.

If the "cold reserve" of one aggregate or instrument is accomplished by another aggregate which is not identical to the basic one (operating on different principle), the probability of successful operation of such group of aggregates in the case of absolutely reliable work of all switches from the emergency component to a standard component may be determined by using the suggestions from [45] and utilizing the following formula

$$P = \frac{\lambda_2^2}{(\lambda_2 - \lambda_1)^2} e^{-\lambda_1 \tau} + \left[\frac{\lambda_1^2 - \lambda_1 \lambda_2}{(\lambda_2 - \lambda_1)^2} - \frac{\lambda_2 \lambda_1 \tau}{\lambda_2 - \lambda_1}\right] e^{-\lambda_2 \tau},$$

where λ_1 is the frequency of failure of the basic aggregate and λ_2 is the frequency of failure of the standby aggregate (one utilizes two standby aggregates).

In the case of single-purpose n instruments with absolutely reliable switching devices, the probability of successful operation of at least one of these aggregates, on the assumption that the others are in the "cold reserve," is determined by using the following formula

$$P=\frac{\lambda_2\lambda_3\lambda_4\ldots\lambda_n e^{-\lambda_1\tau}}{(\lambda_2-\lambda_1)(\lambda_3-\lambda_1)\ldots(\lambda_n-\lambda_1)}+\frac{\lambda_1\lambda_3\lambda_4\ldots\lambda_n e^{-\lambda_2\tau}}{(\lambda_1-\lambda_2)(\lambda_3-\lambda_2)\ldots(\lambda_n-\lambda_2)}+$$
$$+\ldots+\frac{\lambda_1\lambda_2\lambda_3\ldots\lambda_{n-1}e^{-\lambda_n\tau}}{(\lambda_1-\lambda_n)(\lambda_2-\lambda_n)\ldots(\lambda_{n-1}-\lambda_n)}.$$

One frequently encounters in practice the case when two different subsystems, designed to execute the same functions, opperate simultaneously. One first activates one subsystem (with the failure rate λ_1) and the other subsystem is on a "hot standby" but does not operate (the failure rate of λ_2). As the first subsystem fails, the second is activated to full power output and the failure rate here will be λ_3.

If one knows the probability of successful switching off of the emergency system and the activation of the reserve system R_{12}, then the total probability of successful fulfillment of the goal functions using such twin complex will be defined as:

$$P=e^{\lambda_1\tau}+R_{12}\frac{\lambda_1}{\lambda_1\lambda_2\lambda_3}[e^{\lambda_3\tau}\quad e^{(\lambda_1+\lambda_2)\tau}].$$

In the real operational conditions of the onboard systems it is possible to repair some failed components, and if one is to take into account the hierarchy of repair, keeping in mind also the real probability of failure of some devices which shut off the malfunctioning components and activate the backup components, it becomes possible to model the emergency situations only by using multiple modeling. Such simulated modeling which is conducted by statistical testing requires considerable computer time but it produces at the same time the possibility to account for the probability that the goal functions will be fulfilled, to determine the probability of successful flight with definite losses and the probability that the common task which is to be executed by several devices will be fulfilled.

The first example of solving a common problem of simulted modeling is to investigate under different conditions the braking capabilities of the manned spacecraft, descending through the atmosphere, returning from the interplanetary flight and entering the Earth's atmosphere with hyperbolic velocity. The studies simulate approximately the control of motion, the gas dynamic flows, the mass heat exchange and the effect of overload or stress on the crew. One models the terminal principle of control, ensuring the landing at the assigned longitudinal range. At the major descent trajectory segments (the atmospheric entry segment, the segment of leveling off, the pretouchdown maneuver) one solves the navigational problem, ensuring the constant requirement of maintaining the assigned range before touchdown. On the trajectory segment for touchdown maneuver, this is the only requirement. At the moment of atmospheric entry and leveling off, the range is of secondary importance, and is to be accomplished only in such case when it does not interfere with the current and more important tasks. Among these are the proper entry into the calculated entry corridor (avoiding the limiting overloads and ricochet trajectories) attempting not to exceed the peak radiation and convective flows (because of thermal shielding peeling), provision of the close to minimal mass removal of the heated layers of the thermal shielding, provision not to exceed operational temperatures within the spacecraft compartments and preservation of acceptable integral level of stress to which the crew is exposed. /54

The descent control is a situation event since at different stages of motion, one is to satisfy specific predominant requirements, having always in the back of one's mind that one has to land at a specific point.

The principles of control (rolling and axial turnovers) as well as the aerodynamic configuration (moderately pointed bicone), which are established and described in a number of studies, are assumed to be fixed.

The programmed complex for modeling of return flights includes the interconnected modules to calculate the aerodynamic parameters by numerical integration of spacecraft surface pressures, the calculations by integration across the frontal surface area of the linear and mass removal from the coating, the numerical integration with respect to time of the equations of motion with the prognostication of proper zeroing-in and estimate of thermal flows within the specific regions of the device, the calculation of required thickness of thermal shielding at the specific points on the surface and determination of the fuel expenditures by the propulsion units for control and banking.

The perturbations in the values of starting data is described by the normal law. One also investigates the following data scattering: entry velocity into the atmosphere, entry angle into the atmosphere at the height of 120 km, gradient within the exponential law of the atmospheric density change as a function of altitude, angle of attack of the device for standard balancing at the moment of atmospheric entry, correction for nonstationary flow conditions, referenced against the computed lift-to-drag ratio, corrections of coefficients which define the apparent overload as a function of apparent speed, corrections to the attenuation function of radiation and convective thermal flows because of removal of the thermal shielding destruction products toward the boundary layer, corrections of the coefficients which take into account the effect of the stress factors which affect the astronauts, the effective enthalpy of the thermal shield coating, the coefficient of thermal conductance in the thermal shield and the degree of blackness of carbonized layer.

The typical return trajectory into the Earth's atmosphere appears as follows.

The spacecraft is first captured by the atmosphere and switched over to the acceptable overload and thermal conditions. On the trajectory capture segment, one observes the linear dependence between the apparent overload and the apparent speed, making it possible during the flight along the upper entry corridor boundary in the rarified atmosphere to maintain its position without leaving the corridor because of the lifting force, directed downward. During the motion along the lower boundary of the entry corridor and dense atmosphere, one has the ricochet phenomenon with the switch-over to the permissible overload level as a result of the lifting force directed upward.

In the course of maximum overload passage, three alternatives are possible: the ricochet with the emerging flight out of the atmosphere; the intensive braking with overload, close to the trajectory maximum overload, corresponding to the minimal thermal shielding mass, and the motion of leveling off, with progressively decreasing overload. In the latter case, a small increase in the /55
thermal shielding mass is compensated by the acceptable integral overload action which is the major criterion of the second flight segment. After braking, down to the circular velocity level, the danger of leaving the atmosphere, of the overloads and of intensive heating with large removal of thermal shielding and overheating are assumed to be overcome, and from that moment on,the control of motion is associated only with the attainment by the spacecraft of the assigned flight distance.

The sorting out of trajectories during statistical modeling pursues two goals. First is to model the scattering of the thermal shielding mass values and the measure of integral overload action. The second one is to elucidate and analyze the limiting atmospheric entry modes (the ricochet phenomenon with the speed of emergence greater than 11 km/s), the destructive isooverload immersion with a measure of integral action $M>1$ (the intensive heating and separation of thermal shielding and overflight of the touchdown area).

Assuming that the average time of the modeling process of the return flight, by using a powerful computer of BESM-6 type, is equal approximately to a real descent time (12-15 min), we must interrupt the operational cycle and data accumulation, providing during one operational session the calculation of 20-30 trajectories with a thorough analysis of 3-4 limiting flight modes.

The typical program complex which utilizes FORTRAN language for the computer of BESM-6 type integrates the motions of the center of masses with the calculations of banking turns. The integration is accomplished by Kutta-Merson method with a constant external step of 1 second and variable internal substeps. At the prelanding maneuver trajectory segment (after braking down to the circular velocity), each 10 s, one carries out the prognostication with accelerated integration by using the improved Euler method to handle the equations of motion of the mass centers. At the entry and leveling off segments, one conducts at each step of integration with respect to time the calculations by using the Simpson method of the aerodynamic parameters, proceeding with the integration across the whole spacecraft surface area of the pressures and mass expenditures. After calculating the descent trajectory, one estimates the thickness of the thermal shielding layer at 5 specific points, by solving the boundary problem using the trial run method.

The second example of studying the flight modeling is to analyze the possible penetration modes of the space vehicle into the atmosphere of the giant planets, for example, of Jupiter. The study [15] describes the technical aspects of such a problem. Here one speaks only about the delivery of the research assembly into the dense atmospheric layers, under the cloud cover. The delivery is accomplished by launching the module and braking and the descent device. It may include the shell with thermal shielding and the balloon system for descent, once the speed is down to 100-300 m/s. The atmospheric entry in the area of the equator of Jupiter, for example, not using a steep trajectory and taking into account the planetary rotation, makes it possible to lower the approach speed down to 50 km/s. Such speed will bring about huge specific thermal radiation flows which exceed $3 \cdot 10^5$ kW/m^2.

Just like in the preceding example, during each step of integration in time, one is to calculate the expenditures of the thermal shielding mass, taking into account the screening effects and blocking of the thermal flows. In the case of the device of a simple shape with symmetric frontal cone, one is to determine the aerodynamic parameters and mass decrease by direct calculations, without any numerical integration, on the basis of the pressure distributions and removal of the coating. During each moment of flight, one defines the alteration in shape because of the linear mass removal and erosion at the truncated surface area. The load at the center of the module (the ballistic coefficient) is being corrected on the basis of the current mass and diameter of the device.

The deviations in values of the majority of starting data is being modeled by employing the truncated, normal law.

For each calculation test run, during the entry mode, one randomly changes the following components of environment, of the starting conditions and of the thermal physical processes:

the gradient of change in the exponential law of the external environment density increase during the atmospheric immersion;

the percentage content of hydrogen in the hydrogen-helium atmosphere;

the atmospheric entry speed because of the deviations in the approach trajectory (within the range of 10 percent);

the atmospheric angle of entry; /56

the function of attenuation of thermal flows which are screened by a layer of the thermal shielding products of decomposition;

function of blocking the convective thermal flows by injecting into the boundary layer the pyrolized gas, consisting of vapors of the binder in the thermal shielding coating;

the degree of blackness of the starting and carbonized coating;

the magnitude of the air-to-drag ratio because of nonstationary conditions of flow and pressure pulsation.

The thickness of heated thermal shielding layer is estimated for the single-layered coating simultaneously with the trajectory calculations by integrating with respect to time at the critical point the differential equations which connect the speed of the temperature front movement into the interior of the coating and the linear material removal for a specific thermal conductivity.

The deviations from computed values (5,000 kcal/kg) of the effective enthalpy are described by asymmetric distribution function which is being formed by a separate procedure.

The realization of the descent trajectory takes about one minute of computer time, if one is to use the computer of BESM-6 type, making it possible to investigate in great detail the process of braking by constructing after 2,000-2,500 different models, the final histograms which demonstrate the scattered values of the thermal shielding mass for the device in question.

Simultaneously, one estimates (by using the method of statistical extremal values) the worst attainable mass levels of the heated and removed thermal material. These quantities are compared with the quantities obtained in the typical samplings of the limiting descent modes (the movement along the lower boundaries of the approach entry corridor, combining this information with the pessimistic model of the density change as a function of altitude or braking in relatively rarified atmospheric layers with the approach along the upper corridor boundary by using the optimistic model of the density change with respect to the altitude).

The third example of simulated modeling reviews the controlled descent and touchdown of a heavy automated spacecraft which is to land on Mars.

The spacecraft systems include the following: the frontal braking screen, the propulsion unit for prelanding braking, the kinematic supports and shock absorbers in the landing gear and the propulsion units which control the descent. The rarified atmosphere makes it impossible to utilize only the lift-to-drag ratio technique to slow the approach flight, and therefore one must actively brake the speed by using a propulsion unit at the velocity level of 300-700 m/s.

The mass optimization which is based on the dimensions of the shield and fuel storage require a well developed aerodynamic braking assembly which provides the velocity quenching from 6 km/s at the approach trajectory down to 300 m/s at the arbitrary altitude of 3-3.5 km. Such speed reduction is accomplished by movement through the atmosphere, using the sliding trajectory descent and banking control of the device, the frontal shield of which, because of asymmetry in shape or mass balancing, provides for the lift-to-drag ratio on the order of 0.4-0.45. If the controlled descent is not optimal, one can employ either the accelerated trajectories with isooverload braking or the extended ricochet trajectories with repeat atmospheric entry. In the first case the planetary approach takes place too early and the device has no time to slow down to the required speed and in the second case - the device will ricochet with a considerable speed and it is impossible to quench its speed, even after a repeat steep descent. The detailed methods of motion calculations and also the questions of the descent trajectory optimization and the description of dimensional-mass parameters of the above-described devices are presented in the studies [15, 45].

The optimal descent trajectory is close to the isoaltitude flight mode, with the leveling off after the passage of the maximum overload during atmospheric entry. The switch-over from the isoaltitude mode is accomplished at the flight speed on the order of 1200 m/s. The active braking trajectory is of low angle type, with the rotation of the velocity vector just before touchdown. The propulsion unit thrust for prelanding braking corresponds to 1.5-2 terrestrial units.

The goal of the descent control is to attain at the end the assigned conditions, in other words to achieve at a given velocity and in the presence of flat trajectory, the least possible flight speed. The speed reduction helps to save fuel for the pre-touchdown braking maneuver or for the subsequent take off of the module which is of returnable type and in which the propulsion unit is combined with the touchdown propulsion unit. /57

The goal of the touchdown control is to minimize the fuel expenditure and create acceptable conditions for the spacecraft contact with the planetary surface.

The modeling of descent and landing is conducted simultaneously and therefore the situation control is brought about at two different trajectory segments, with different perturbations and flight conditions. In considering concurrently the two major problems of descent and touchdown, one is to solve during the flight the problem of assigned range of low accuracy (on the order of 1000 km), necessary for the radio communication during touchdown between the control center and spacecraft, employing direct communication and also of landing in the area of the planet with acceptable macro-relief.

In contrast to the preceding problem, the overload constraints and thermal flows play no role and therefore are not to be taken into account within the permissible entry corridor.

Among the perturbations which are to be modeled are the information about the planet, which is of indefinite nature. Among these are the density at the conditional zero altitude and the exponential indicator of the density change as a function of altitude, the height of landing site, assigned at the beginning of the landing trajectory with respect to the conditional zero level, defined for the descent trajectory segment, the free-fall acceleration near the plantary surface, the radius of prelanding maneuvering in selecting the site (the nominal value here corresponds to the conditional time of "suspension" which is 60 s, with the horizontal movement of 4 m/s).

The perturbations which reflect the deviations of the design parameters and the data associated with the near planetary trajectory are as follows:

characteristic speed of approach trajectory correction;

velocity and the atmospheric angle of entry at the conditional altitude of 120 km;

frontal drag and lift force (along the polar of the device);

balanced angle of attack with respect to the nominal (change of the angle of attack is caused by the change in centering);

angular speed by banking control;

specific pulse in space and thrust of the propulsion units for prelanding braking;

coefficient of the pulse loss during the reverse activation of the propulsion unit in the course of touchdown;

specific pulse and the control of propulsion unit thrust;

the degree of nozzle expansion in the propulsion units for prelanding braking;

fuel expenditures of the propulsion units which are used to orient the spacecraft and to stabilize it by employing the pitch and yaw during descent and landing.

In the computational trajectory model, one determines the motion of the centers of masses. The motion around the center of masses is modeled only in terms of banking to account for the fuel expenditure during the descent.

At the prelanding trajectory segment, one studies two types of the vector of thrust control: the gravitational turn (thrust versus speed) and braking with a constant trajectory inclination angle, prior to the prelanding maneuver.

The moment when the landing propulsion units are activated is prognosticated toward the end of the descent trajectory by using the approximate calculations of the active braking trajectory segment. Almost all perturbations are described by the normal law, with the assignment of limiting deviations and under the conditions that the whole range of deviations will be $\pm 3\sigma$.

The range of deviations of the atmospheric entry speed is determined by the difference between the minimal and maximal velocity for the specific range of take-off data. Between the limiting values, one assumes that the distribution density of the obtained speed modeling is uniform.

At the starting phase of modeling by the methods of statistical testing, one employs the test run by using the limiting deviations in regard to the information about the planet. The total number of descent and landing modeling modes may be quite considerable since one calculation of the trajectory takes between 7 and 18 seconds, by using the computer of BESM-6 type.

One considers separately the emergency flight modes associated with the possible aborted landing and returning to the near-Martian orbit. There are two advantageous trajectory segments during the descent and landing, if one is to abort the latter: the first segment is from the moment of atmospheric entry to the middle of the iso-altitude level and the second segment is from the beginning of the active braking to the touchdown. On the first trajectory segment, because of rotational lift force, it becomes possible to ricochet, developing an intermdiate orbit by using the power of the landing propulsion unit. At the second trajectory segment it is advantageous to pitch the spacecraft from braking maneuver to acceleration and the subsequent speedup, by using the power from the propulsion units which are used for the return flight. At the end phase of the second segment, it is advantageous to complete the standard prelanding braking program and start at the moment of "suspended" mode before touchdown. /58

The fourth example of simulated modeling is devoted to the study of dynamic loads, acting under different conditions of spacecraft landing.

The goal here is to ensure the landing under different operational conditions, which are defined by the totality of external actions: the inclination of the landing site, the density and hardness of the soil, speed, banking pitch and yaw evolutions during the contact with the soil of the different supports in the landing assembly. The simulation of external conditions is conducted in two stages: first - with the singular "runs" to analyze the sensitivity of the spacecraft to the limiting deviations of the starting data and then - in the mode of sampling statistical testing using all multiple cases of load exposure during the operation. One conducts separately the study of destruction of the basic design and failure of support assemblies in the presence of possible and random deviations in the strength of materials and different parameters of the soil.

Among the assemblies of landing support, one models dynamically the active support system and the behavior of the spacecraft as its supports make contact with the planetary soil.

The majority of random deviations is modeled by using the normal distribution law. One such session may take from 10 minutes to one hour of the computer time if it is the computer of BESM-6 type, or the unified computer system of ES1040-ES1060 type.

By employing the numerical integration, one solves the problem of spatial motion of the center of masses and the elastically interconnected point and distributed masses with respect to the center of the mass.

One applies the external loads and inertia stresses at different junction points of the mass systems and the effects of the control organs, the mounting of the propulsion units, the factor of the payload, the instrument compartments and fuel tanks. During such evolution of the elastic system of masses, one defines some additional inertia loads as a function of distribution of the spacecraft design masses.

The integration of motion of the mass systems is conducted with the step from 0.01 to 0.0001 s. In the time period prior to the contact and in the case of bounces, the integration step may be increased and during the touchdown, the step is fractionated even more, for the accurate reproduction of the contact dynamics.

The macrorelief of the landing site may be modeled not only by different inclination and slope orientations but also by the random combinations of positions of the spacecraft on the slope of craters of different size and depth.

The successful touchdowns are classified on the basis of a number of criteria, for example on the basis of the condition of the landing supports, on the basis of physical contact between the majority of landing supports or the surface of the contact ring, on the basis of the spacecraft axis and its deviation from vertical, once the spacecraft is at rest or once the spacecraft landing supports are back to the projected position.

The most complete study of landing mode is to model the dynamic design construction, taking into account the fluctuation in the quantities of fuel in the tanks.

The technical parameters of landing assemblies, of the soil and of the touchdown itself have been considered in great detail in the study [15] where the reader can find also a description of the dynamic design schematic of the landing assemblies.

The fifth example of simulated modeling is characteristic for the design studies which investigate the composition of the spacecraft systems, with the proper selection of ways and levels for backup instrumentation. The optimal backup instrumentation systems in the spacecraft is impossible without modeling their operational efficiency, taking into account the failures, repairs, replacements and possible switching over to the other flight programs which would make it possible to solve only a limited number of the tasks assigned to the mission.

Of particular importance is to study the dynamic operational modes of the standby onboard systems, in the case of autonomous and lengthy flight modes of the space, orbital, interplanetary and /59
planetary devices.

The complete simulation of emergency situations presupposes not only the accounting for all the standby equipment, but also the analysis of functional standby availability, whereby, by changing the programs and flight conditions, one can attain the same goals by activating the other spacecraft systems. On the basis of such simulated modeling, one develops recommendations to be applied to the design project in order to improve the functioning of such systems (the increase in the fuel supplies, the provisions for parallel-sequential operation of the propulsion units, the increase in the available thrust, the load decrease on the carrying surfaces, the increase in the available power from the power sources on board, the increase of the chassis base, etc.).

The criterion in such problems is the probability that the missions and flight programs can be fulfilled, the probability that the flight will be successfully completed and that a part of the goal functions will be accomplished, with the provision of possible losses.

The thorough analysis of alternatives which ensure the alleviation of emergency situations is supported by the functional-logic diagrams which describe different consequences of the assembly and sub-assembly failures. In using such approach one cannot, unfortunately, avoid some idealization in the estimates of the fail-free instrumentation and equipment, as one considers the mass model of the spacecraft. It is ordinarily assumed that the average time before all elements within a system will fail and also the average duration of the repair cycles for specific instruments is guaranteed by the research development at a high level of confidence, provided by a lengthy experimental study and refinement. It is also assumed that all operational and standby units are designed by using the same component base. The spacecraft, by using such approach, is being assembled by using the readymade macrocomponents and the question of "internal" standby in such enlarged component is not being considered.

In regard to each instrument, aggregate, block or assembly, the design parameters usually are known and are compiled in terms of the specific data base about the mass and size parameters, about the energy consumption and heat evolution, about the average operational time before failure and frequency of failures, as well as the average time required to repair and the frequency of repair.

Simultaneously, in accordance with the types of systems and aggregates on board, the specific and advantageous methods of standby equipment provisions are provided for. For example, in the case of specific instruments of the control system, of the radio communication system, of some types of propulsion units, of onboard computational systems, it is advantageous to have available the "hot" standby equipment or redundancy systems. The "cold" standby equipment is used for the support batteries and current sources which are used for the viable existence of the whole complex, for the assemblies of thermal control and for some special instruments in radio communication.

The "hot" standby systems in the spacecraft are called for because of the operational requirements to have them ready within a short and strictly defined time interval, which corresponds to the trajectory corrections, the radio communication sessions with the Earth or with the other spacecraft. One knows and utilizes on the basis of ballistic calculations the mathematical expectations and root-mean-square deviations for the components which characterize speeds, inherent in the operational orientation cycles, in the acceleration, in braking and trajectory corrections.

One also utilizes the earlier mentioned functional-logic schematics of typical consequences in the case of an aggregate or a system failure, and also a probable choice of the flight programs, with the cyclograms which activate the major instruments and aggregates.

The problems of optimal standby equipment with orientation toward the automatic shutoff in case of emergency and switching over to the standby components, having a high or absolute reliability in switches, disregarding the cases when it is impossible to repair the instruments which have failed, may be resolved by using the analytical relationships.

The accurate and thorough evaluation of possible emergency situations is being conducted by modeling the events which cannot be analytically calculated. Amont these are the shutting off of several emergency assemblies and activation of several standby assemblies which are in the "cold" standby reserve, having a specific probability for the emergency switch-over control. It is impossible to describe analytically the actual picture of repair with a complex hierarchy of sequences, called for to reestablish the operation of the instruments, under the conditions when one sometimes has to rearrange the sequence of repair or temporarily post- /60
pone the repair during the stressed moments of flight or the overload of the crew by the current jobs.

The statistical modeling of situations on board is based on the following assumptions:

the number of failures and repairs is assumed to be stationary, of Poisson type;

the interruption of repair work on the basis of the flight condition will not affect the future accrued operational time or the functions of the repaired instrumentation;

the failure of components in one system will not affect the operation of the components in another system.

On the basis of the mathematical expectation, presented earlier, and root-mean-square deviation of the operational accrued time before the typical instruments and aggregates fail is equal to $1/\lambda_j$ and the repair time $1/\nu_i$.

The statististical modeling of repair work on board of the orbital station makes it possible to prognisticate such situations which are difficult to assess as the accumulated requests for delivery of standby equipment, the supplemental refueling of the onboard systems, delivery to ground laboratories of the assemblies which are to be repaired, the utilization as the standby assemblies and subsystems of the delivery spacecraft, with the usage from such spacecraft of the stored equipment and fuel for the orbital maneuvering operations.

The problems of simulated modeling which we have considered, just like the other problems of modeling of structural composition and shape of the spacecraft, would be unthinkable without using the powerful computers with peripheral devices.

We can recommend the special literature (for example [2]) for the readers who are interested in specifics of using the computer systems.

CHAPTER 4.

CONSTRUCTION OF MATHEMATICAL AND INFORMATIONAL MODELS DEFINING THE CONFIGURATION AND SHAPES OF SPECECRAFT

The mathematical and informational models which describe and reflect the spacecraft configuration differ in terms of their complexity, reflecting the multiplicity of shapes and design requirements - from the primitive and simplest geometric bodies which are not designed for the proper gas dynamic flows, which are oriented toward the flight outside of the atmosphere, to the most complex shapes of binary variable curvature, with overlaps, overflows, channels and smooth transitions between the lateral sections, the profile of which aims at the subsonic, supersonic and hypersonic flight modes. /60

Naturally, it would be unreasonable to employ complex approaches of topological nature and time-consuming methods of kinematic shape development in employing complex configurations, by selecting the contour arcs so as to describe the simple body shapes. It is reasonable to subdivide the configurations of devices, in terms of the complexity of the shape formation, into groups and then to apply for each particular group of shapes which is used in the practice of design the method of mathematical modeling of enclosures and the principle of data model construction, containing information which describes and reflects the surface shapes, after which the defined methodology can be generalized.

It is convenient to single out three such groups of configurations.

The first group includes the shapes which are formed by combining the elementary bodies, spheroids, cones, cylinders, toroids, parallelepipeds and plates. The shapes of such type are quite characteristic for the blocks and segments in orbital and interplantary devices which undertake autonomous flight outside of the atmsophere and which are protected during the atmospheric flight by /61
the enclosure containing the payload. The mathematical description of such shapes is not complicated and is related to the characteristic longitudinal and lateral cross sections, with the assignments in the schematics of the general radii, of the diameters, extensions and angular dimensions.

The second group includes the configurations which are designed to move through the atmosphere while accelerating or undergoing braking, when the flows at the hypersound velocities define the heat and mass exchange, the overload operational modes and entry into the assigned area. As a rule these shapes of descending or shuttle type devices enter the atmosphere either from orbital flight or from the approach trajectories. The shape of such devices is described by the theoretical drawing, combining the streamlined surfaces (the frontal surface or screen, the nose cone or main rocket body) and the protected surfaces in the rear part (bottom part of the rocket with the payload).

The surfaces here consist of the sectors which are parts of complex geometric bodies, among which one finds elliptical truncations, the cone and spherical segments, scarfed elliptical cones, cone aprons and aprons and blisters, tetrahedrons, bodies with sectorial elliptic cross sections and linear change of ellipticity lengthwise. The shapes of this second group are described by piecewise-analytical method. An example of a mathematical model of the bicone configuration will be presented below.

The third group is represented by the bodies of "lifting body" type, with smooth outer shapes and variable, arbitrary cross-sectional profiles.

The information model contains the data for control or base points which control the contours of cross-sectional support and the coefficients making it possible to transform the contours smoothly in space, by going from one cross section to another. The mathematical model of such shapes includes a number of conditions and recurrent relationships on the basis of which one can calculate any point on a cross section, between the two neighboring assigned cross sections.

The interaction between mathematical and information models is brought about by using the metric position calculations and using in one common algorithm the data base outlines of the device and also the library which includes the program calculations for the parameters of arcs and their cross sections and parameters which transform these arcs and place them along the body. The purpose of such follow-through algorithm is to obtain the third coordinate on the point outline on the basis of two other, assigned coordinates. It will be shown below that it is sometimes possible to describe small and smooth surface areas (which have no breaks in the curvatures) by one analytical expression in the form of a bipolynomial, the coefficients of which may be represented in matrix form.

However, the whole outline of the body with arbitrary cross sections may be assigned only topologically, in other words by combining sequentially selected contours on the assumption of continuous "line flow" which connects these contours. The major lines which are drawn on the theoretical layout are the ditox lines, central lines and percent lines which connect the linear coordinate points of the same relative angular dimension with the points on the neighboring cross-sectional planes. The transformation of the arc contour of one cross-sectional dimension into a similar arc contour of somewhat similar cross-sectional dimension is being conducted in the simplest case by the physical outline-kinematic method, changing smoothly the /62
parameters which characterize curvature, the end coordinates and orientation of arcs. The outline surface is being formed by continuous displacement in space of the transformed arcs in the contours, thus creating instead of the two-dimensional point information model of cross-sectional configuration, the mathematical model of continuous configuration.

Let us consider two examples of mathematical modeling of configurations of the second group, which is a combination of complex bodies, described by the piecewise analytical method.

The first example describes the bicone configuration of descending devices shown in Figure 2.1. The advantages of having shapes of such type, rather than those which were used earlier and which were of segmented-cone form, were presented in the studies [15 and 45]. The frontal surface of the bicone device is essentially a scarfed elliptical cone, the configuration having the angle of attack which is close to zero.

By operating in the Descartes coordinates and having longitudinal axis of the cone X, and the vertical in the plane of symmetry Y, drawn through the cone apex, assuming that θ is the half-cone angle in XY plane and designating by ε the cross-sectional ellipticity, one can obtain the expressions for integration with respect to the circular angle Φ and the longitudinal coordinate X in order to calculate the surface areas and volumes exposed to the flows, the aerodynamic forces and moments (with respect to pressures), of the thermal shielding mass removal.

The size of elementary surface area dS on the cone may be calculated as

$$dS = \sqrt{2A}\, d\Phi dx,$$

where $A = \epsilon^2 k \operatorname{tg}^2\theta + \epsilon^2 \cos^2\Phi + k \sin^2\Phi$

(where $k = [y_0 + (x - x_0)\operatorname{tg}\theta]^2$), y_0 and x_0 are the coordinates at the junction point between the cone segment and elliptical blunted segment.

The scarfed elliptical cone defines the variable limits of integration. One of such symmetric limits is calculated by using the following formula

$$\Phi = \frac{\pi}{2} - \arcsin\frac{1}{\sqrt{k}}y.$$

The second example is characteristic for the piecewise configuration of the "lifting body" type, by selecting the components the cross-sectional contours of which consist of two semi-ellipses with a common base. The ellipticity of each semi-cross section changes linearly along the axes of each body.

In polar coordinates with the longitudinal axis X, the radius from the center to the contour ρ and the rotational angle within the cylindrical system ϕ we will obtain the equation for each body

$$\rho^2\left[\frac{\cos^2\varphi}{b^2(x)}+\frac{\sin^2\varphi}{c^2(x)}\right]=1,$$

where b(x) and c(x) functions are defined by the size of semiaxes of the ellipses in the initial cross section of each body (b_0 and /63 c_0), the length of each body ℓ and the same values at the end cross section (b_1 and c_1), in other words

$$b(x)=b_0+\frac{b_1-b_0}{l}x;\ c(x)=c_0+\frac{c_1-c_0}{l}x.$$

The computational relationships and expressions which are used to calculate the aerodynamic characteristics by numerical integration and to compute the mass removal from the surfaces of the above-described bodies are presented in the studies [15, 45, 48].

The difficulties in working with the mathematical and informational models of smooth contoured bodies with arbitrary cross sections are explained by the fact that the transformation of a contour into another one, in the presence of inflections, bends, linear segments and degenerating turns represents a problem which cannot always be solved from the point of view of the continuities in derivatives (in obtaining the smoothness of different order). More often than not, one must provide for smoothness in terms of the first derivatives (the smoothness of the first order). In some cases, it is necessary to provide continuity not only in the first, but also in the second derivatives (the smoothness of the second order) on the profiles of carrying surfaces and on the flow-facing parts of the body, along the flow lines. One of the important requirements is to liquidate the bends and microwaves (oscillations) which have not been planned and the absence of them will be manifested by the sign constancy in the second derivative along the contoured lines which connect the neighboring cross sections.

All mathematical methods which describe the outlines of the group which is being considered may be arbitrarily subdivided into three large classes. In the methods of the first class, one utilizes the standardized special functions which reflect conditionally the contour of the typical cross section completely with the parameters which transform, in a necessary manner, the cross sections with respect to the distance. The conditional depiction of the contour is obtained because of the limited number of typical shapes which may, with some degree of error, be described by the special functions. The second class includes the methods which are based on the piecewise description of contours and intercontour lines of parameter-carriers by having a set of joint arcs of different curvature. The third class may include various methods of canonic description of contours, joined by polynomials with different degrees of smoothness.

The typical representative of the methods of the first class is the method of special contour which is used primarily to reflect, employing several reference points, the characteristic rectangular-oval contours or joint semicontours.

The closed contour is described by the equation of the following type

$$y = \pm \sqrt{\frac{1 - z^2}{1 - mz^2}},$$

in which case y and z are the dimensionless coordinates which are measured in reference to the size of a standard square embracing this contour. The m parameter which changes as a function of distance, proceeding from one cross section to another, reflects the contour shape, changing it from a square (as $m \to 1$) to the circumference (when $m=0$). In the range $3 \leq m \leq 1$, the contour is of convex type and when $m < -3$, the points of bending and concave sectors will begin to appear. The detailed description of the method of special /64
contours and its utilization and refinement is described in the study [49].

The longitudinal cross sections of the carrying components can be conveniently described, as suggested in the study [48], by the exponential dimensionless special functions of the following type

$$y = a\tilde{x}^{b} e^{c\tilde{x}},$$

in which $\tilde{x} = \frac{x}{x - x_m}$, and therefore, in having $\tilde{x}$, which can be transformed from 0 to $+\infty$, the real x axis of the profile changes from 0 to the maximum chord size x_m.

The a, b and c quantities, defined by the parameter carrier functions from z coordinate with respect to the sweep of the aggregate, change the position of the point of maximum profile thicknesses, the magnitude of this thickness in the semiprofile (from the central line) and the conditional radius of rounded-off segments at a given distance of the blunted tip. By changing these parameters, it is possible to obtain pointed, supersonic profiles or profiles with extended backward points of maximum thickness, and the tip which is bent upwards (the hypercritical profile).

As one can see, in the case of the methods for the first group, one requires small amount of data as to the maximum size dimensions of the cross sections and limiting deviations (the maximum thickness and profile fullness). The continuously assigned parameter carrier functions guarantee the absence of inflections, bends and discontinuities between the cross sections. The informational model which provides for the methods of the first class presents no difficulties and includes orderly-organized set of points which define the limiting size of the cross sections and reference points which characterize the behavioral expansion of the parameter carrier functions, in terms of the distances and sweeps.

The utilization of the methods of the first class is constrained because it is impossible to describe accurately the cross-sectional contours with inflections, with the junction points between the straight and oval surfaces, with the protrusions because of the limited number of parameters which define the characteristic appearance of the outline.

More effective description of the zonal outlines is used in practice by employing the local analytical special functions which are being formed by using the piecewise interpolation of the outline-forming surfaces and also the hypersurfaces of spatial functions which reflect the standard air pressures or temperature profiles. Quite interesting and promising are the fracional bipolynomial functions which are assigned in matrix form. They are represented quite fully in the studies of matrix stereometry by K. M. Nadzharov.

The bipolynomial outline-forming function has the following form

$$y = \sum_i a_i x_i^\alpha \sum_j b_j z_j^\beta;$$

It is characterized by intersecting terms with the maximum power parameter ($\alpha+\beta$). The a_i and b_j coefficients are determined by a special program, ensuring the averaging-out of the interpolation function between the reference points, if the basis rests on several nodal points and the total number of such points is equal to the number of coefficients in the bipolynomial. The program selects as such nodal points the most advantageous representatives from the /65 representatives from the multiple set of starting points and there is absolutely no need to have orderly positioning of these points within a specific matrix or network. The informational model of zonal outline description includes the data sets of the initial points, among which one can find the extreme points, outside of the zone, which define the junction points of the bipolynomial "remnants" which is the result of the neighboring zonal overlaps.

The most widely used and the most common method of the second class (for the piecewise joining together of the contours having arcs of different curvature) is the method of curves of the second order. Its universal employment, including the production mannuals of various services, is explained by the apparency of geometric construction and convenience of describing the contours which are formed by the designers who are working with the straight line segments, with the circumferences of different radius and with convex arcs.

These arcs are fixed at the junction points with a specific determination of tangential lines and also the specific discriminants or radii, and coupled centers of circumferences. The change in the conditions of arc fixation as one proceeds from one cross section to another is approximated by the functional relationships, the graphic depiction of which is obtained by the same piecewise method from the arcs or straight line segments.

The canonic equation of the curve of the second order in its explicit form will appear as follows:

$$y = A_1 (x - x_0) + A_2 \pm \sqrt{A_3 (x - x_0)^2 + A_4 (x - x_0) + A_5},$$

where A_i coefficients are defined by means of the coordinates of the arc end and the intersection point between the tangential line at the end of the arc and also by the magnitude of the discriminant. The magnitude of the discriminant is determined on the basis of graphic depictions, defining the segment which is bisected by the curve from the base of the median and its relationship to the median within the triangle, formed by the chord and tangential lines at the extreme points of the arc. The special cases of the curves of the second order are parabolas (the discriminant is equal to 0.5), the circumferences or ellipses (the discriminant is less than 0.5). In operating with the circumference arcs or straight line segments, the variation "freedom" is reduced and the calculations and preparing of the starting data is simplified.

The advantages of piecewise-arc description of the cross-sectional contours, aiming at the design arrangements with clear-cut mold-drawing, is that one obtains initial information with the absence of oscillations, deflections which have not been planned and pulsation of the contour curves.

The information model which accompanies the method of curves of the second order contains coordinates of the arc junction points, the data about such junction points, the coefficients of the curve equations for the cross-sectional contours and for the lines of parameter-carrier functions. The volume of such information is small but the data is organized by using different data banks with sharp differences in the size of coefficients (the difference of 5-7 orders of magnitude).

The major drawback of the mathematical methods of the second class and their informational models is the requirement of time-consuming mold preparatory work and the non-automated geometric tying-in of the contours with the lines of the parameter-carrier functions and also the unavoidable errors, as the data is introduced into the data bank and into the programmed computer. The inaccuracy in geometric construction at the arc junction points indicates that the inflections in the contour which were not planned will begin to appear. The curvature discontinuities at the arc junction points are undesirable because of the streamlined flow requirements, they complicate the calculations of the trajectories of motion and the automated processing involved in the outline-forming detail description. /66

The methods of the third class, in terms of a number of distinguishing features, are opposite to the methods of the second class and their main advantage is the uniform information model and the efficiency of data input which does not require accurate geometric tie-in. Herein is also the major drawback of these methods - the possibility of having uncontrolled oscillation, inflections and contour bending which have not been planned.

Among the methods within this class are the classical spline functions assigned explicitly and in the parametric form, the local splines, the approximation splines, bisplines, polynomial functions which are assigned in terms of the outline grid with a number of specific differntial boundary conditions, the methods of aerodynamic contouring, the methods of reference arcs and others. The worst approximation of such junctions with smooth polynomials is in the case of contours in which sharp curvature changes are taking place, at the junctions of different arcs and straight lines and at the points of the contour inflection. The bisplines enable the designer to assign and transform the contour by employing the control points within a polygon which embrace the future outline-forming curve. More often than not, the spline functions are used to describe the limited sections of the typical joint aggregates (junction points, overlaps) and also to assign the outline grids, which define the curvilinear surfaces of the components which are being processed. One is also familiar with the applied areas of the spline function utilization to approximate complex spatial surfaces in modeling the standards of the external loads or the position of lines in the models of gaseous dynamic flow experiments.

A special place among the methods of the third class is occupied by the methods and algorithms of the piecewise contour description of the conditionally joined polynomials, supported by the special constraining arcs which prevent the appearance of unplanned inflections and pulsations of the outline-forming curves (the method of drafting the reference arcs).

The major concept in these methods is to acquire the advantages which are found in the methods of the second class (the apparency of presentation, absence of overlaps at the curvature junction points) without losing the advantages of the spline function (the efficiency of automated data preparation).

The general idea in this method can be explained if one is to consider the simplest case of drawing a contour curve through four points. The first three points are drawn precisely to depict the curve of the second order, for example, a parabola. The same curve is drawn through the second, third and fourth points. Within the interval between the second and third point, the outlining curve of interest is determined by using the weight coefficients. These coefficients reflect the preference in the vicinity of the second point on the parabola, which approaches from the side of the first point, and near the third point on the parabola which moves away toward the fourth point. At the center of the interval, the curve of interest along the y axis is averaged out with respect to y axes of two limiting parabolas. The mass coefficient may reflect the linear tactics of the y axes preference, and then the curve of interest will be a cubic parabola. By varying the type of mass functions, /67 one can construct practically any requirement for the positioning at the interval junction points, but in practice the differences in y axes coordinates which limit the parabolas is small and precise adherence to the conditions of smoothness at the nodal points is not required. Instead of parabolas, one could utilize the other limiting curves of the second or third order: the circumferences, fractional rational functions, hyperbolas, etc. The type variation of the limiting curves is the subject of selection, in finding the optimal form of the surface outline. In the process of such optimization, one selects from the total number of nodal points the most acceptable ones, which will play the role of the base points, through which the outline contour will pass in most precise manner. The remaining points become reference points and the contour outline will pass between them, by employing different approaches and criteria of averaging-out and also the criteria of smoothness and contour rigidity.

The spatial "filling" of intervals between the nodal points of the grid at different cross-sectional points of the contour and the percent lines are ordinarily being drawn by using the same mass coefficients which enable one to change the coordinates smoothly, depending on the degree of "nearness" to each neighboring line of the contour.

As one can see, the informational model includes the data about the nodal, reference and base points, grouped in terms of the cross sections, percent lines which tie together the cross sections, the bitox lines and intermediate lines. Simultaneously, such model defines the possible limits in the change of coordinate nodal (base) points since in optimizing the surface outlines, a small displacement of the starting contour points is permitted.

In a general case, the data base which provides the informational model of the surface outline is supported by a bank of geometric data and a packet (library) of programs which are used in the calculations of the outline points and lines, the conditions of placement and some auxiliary functional relationships (the values of the second and third derivatives, the local curvature radii, the control points, discriminants, the total length along the contour outlines). Such data bank of geometric information has the hierarchic multilevel logic structure, with sequential transition from the aggregates to the contour grids, next - to the contour curves of specific cross sections, to the curved lines of the parameter carriers and then to the specific arcs which are represented by the nodal, reference and base points. At the specific levels of the archive data bank catalogue, one has the encoded data about the conditions of point fixation at the boundary intervals and about the coefficients of the outline-forming arcs.

The size of such data bank may be arbitrary and for each group of cross sections, one must provide within the relation structure of the data bank the generation of data which ties together the information about the coordinates which define the outline, with the information about the arc fixation in the contour curves.

The recording and data selection is based on the requests of various users and is available on recall from the program of position calculations on the surface of specific aggregates. In such case, the user operates with a variable number of hierarchic levels of the data bank catalogue, gathering together the information sequentially about specific parts of the spacecraft (the aggregates, zones, surfaces, curves and arcs). /68

The specific users (technologists, design planners) are interested in selective informaton, obtained by using the special reference requests, as for example, one might be interested to obtain the coordinates for a specific aggregate, for a zone of local surface (the aggregate cross section, the junction points on the enclosure, the base of a bevel, etc.).

The informational reference system of the data base provides for a successful search for information in regards to different aggregates and zones, employing several modifications of the component and the data base information about the typical aggregates and cross sections which is stored within the same data volume, with the possibility of direct access.

The informational model and the data base with the program library and reference systems which support it is the supplemental information to the theoretical drawing, and is subject to continuous correction, reediting, update, control with testing and verification of the conditional state of such data. During the verification processes, one could utilize the control requests which are accompanied by the data base which is recorded on the magnetic tapes (the control sums, the conditional total length, the intervals and sums of partial derivatives). The copies of such data bank volumes reflect their periodic correction (the "grandfather," "father," "son" variants).

The programmed accompaniment of the informational model data base, which defines the configuration and outlines of the device, ensures that:

the proper recording and data correction is made, with the periodic data density reduction in the direct access information;

it is possible to obtain the reference information about the structure and content of the catalogues and archives;

it provides the interface between the libraries of the mathematical program description of the outlines;

it provides for functionally viable generation of information about the coordinates, defining a specific point in the outline;

it provides for the information which reflects the numerical model of the contour outline by using the computer graphics;

it provides for the control of conditional state of such information in regard to the outlines.

The physical structure of the data base which defines the model outline is of multifile type - it features a flexible distribution of reference and geometric information recorded on different tapes in a simple and complex form. Each simple recording, accompanied by special features which encode the recording number, belongs to a specific geometric object within the hierarchy of the physical shape which describes the module and the corresponding control summary data.

One of the major areas for practical use of the mathematical and information models pertaining to such outlines is the variety of position geometric calculations with visualization of results by using the compter graphics, all of which is done by the technical personnel, designers and chief designers. Among the data which can be gathered from such models are the points of line intersection /69
for the surface outlines which define the general position or linear surfaces (of "table cylinder" type), the points of line intersection on the outline-forming surface, the points which are at a specific distance from normal and from the lines on the outline-forming surface, the points on the tracing of the plane which defines the general position and which are located at a specific distance from the assigned points and the points of triangulation lines which connect the assigned points on the outline.

Each operation of such position calculation is conducted automatically by using the minimization algorithm for the errors of closure in terms of the distances between the points and incidentally different geometric objects. The general presentation of such a problem and typical procedures for the position geometric calculations are presented in the study [48]. It is required that the informational model and program library which serve the position calculations responds in a fast and unambiguous way to the requests, stating to "produce the third coordinate on the basis of two coordinate points, inherent in the outline."

The functional viability of position calculations provides that the processing time for such response would not be more than one or two hundredths of a second. For example, the development of a numerical model for a simple component outline, with the calculation of the normals, of the linear bevels and nodal point grids requires more than two thousand iteration trials with the request each time about obtaining the third coordinate on the basis of two assigned coordinates. To organize the position calculations, the informational model is requested from the operational computer memory and all prognostications of positions of contour outlines are obtained without using any external data carriers. In the case of an aggregate or a body with twenty cross sections and thirty percent lines, the data volume within the operational memory of such information model, containing the approximation programs, should be 250 kilobytes.

CHAPTER 7.

MODELING BY USING A COMPUTER AND COMPUTER GRAPHICS OF THE DESIGN LOAD-BEARING CONSTRUCTIONS, THE STRUCTURES AND SHAPES OF AGGREGATES, ASSEMBLIES AND SPACECRAFT COMPONENTS

The modeling of design load-bearing constructions and geometric /69 representation of the design components began intensive development with the appearance of computer graphics and computing systems of high speed and large memories, accessible during functional work.

Among the numerous results of using mathematical models in design problems, one should single out in the first place the revolutionary changes in the engineering approaches of synthesizing the constructions of equal strength, based on the detailed verification calculations, using the method of the end components and utilized in designing information models, describing the spatial shapes of the components, their drawing projections and geometric depictions for the manufacture without any master drawing, using the program-con- /70 trolled machines.

The traditional approaches in modeling the load-bearing designs of the shell components and compartments, with the separation into finite components have even recently been oriented toward the verification calculation of the finished design structures. In order that such models, which were earlier used for analysis, could be made the instruments of a synthesis, two problems had to be resolved: it was necessary to decrease the processing time of each verification calculations by two orders of magnitude and to develop the iteration converging procedures for the goal-oriented mass redistribution, adding the load to the underloaded components and strengthening the overloaded components in statically uncertain construction design.

A part of the shell assemblies (support beams, attachments, the mechanisms for the antenna unfolding, solar batteries, radiators, landing devices) is represented by a statically definable construction design.

The other part of the compartments which have a simple symmetric shape and which contains typical shell components (the fuel tanks, spacings between the tanks, instrumental compartments, the compartments for the crew) are calculated by using special methods and determining directly the minimal thicknesses of the multilayered, wafered or reinforced shell panels, and the surface areas of typical side frames, stringers and beams.

The thin-walled pieces of the shell which are continuous and which have variable cross sections and load-bearing side frames, cutouts and discontinous cross sections are separated into the elementary end components (the panel segments for the side frame, the arcing pieces of the side frame, segments of stringers and beams) with the subsequent problem-solution to determine the loads in these

C-2

components, in the statically undefined system.

Let us consider the example of modeling the load-bearing part of the shell, representing the fuel compartments, within the framework of the design mass optimization. In handling the optimization problems, one should take into account the factors of uncertainty, presented in Chapter 3. In such case, one employs the minimax strategy of trial-and-error search, against a background of the worst computed cases. Within the external cycle, one chooses among the load-bearing and design-arrangement schematics. Within the intermediate cycle, one reviews the computed variants and for each load-bearing assembly, one selects the worst computed case. Within the internal cycle, for each variant of the load-bearing and assembly design schematics, and for each computed case, one determines the optimal mass, in other words one selects properly the minimal possible cross section for a typical configuration of the side frames, stringers, thickness of the bottoms and of the compartment shell.

Within the framework of the combined optimization, the formal, integral variables are correlated with the assemblies and compartments and the specific values of these variables define the specific modification of a given assembly or compartment. For example the possible integral or non-integral tanks, the tanks with a common bottom or with intertank spacing, the tanks of spherical, cylindrical or toroidal shape, the multitank system or the system with the interconnected tank compartments, the tanks with external or internal thermal insulation or without it, tanks with laminated design of the walls or with internal longitudinal or lateral load-bearing reinforcements. The multitude of the above-mentioned combinations is supplemented by various construction materials.

During the sequential review of the developed assembly approach, one organizes the sequential change in the parameters of the load-bearing components. The spacing of non-load-bearing side frame components are being varied as a first step. The defined variant of the side frame components is being split into the subvariants, having different spacing of stringers. And finally, after taking into account the multitude of computed cases, all the above-mentioned variants are verified by the appropriate iteration procedure in regard to the permissible sizes of the typical cross /71 sections for the side frame components, stringers, thickness of the walls and thermal insulation.

The limitations which are produced after the iterations which define the size of the design components are defined either by the technology involved (minimal thickness of the walls, of the shelves and other profiles) or by the flight time, by the fuel mass and by temperature (the thickness of thermal insulation) or by the limiting strength (thickness of partitions in the intertank compartments and thickness of the bottoms), or by critical stresses during the loss of stability (the size of subsections and cross-sectional dimensions of the side frame components and stringers).

For the assembly of intermediate complexity, the described iteration process is reduced for one structural variant to 10-20 trials, with the required accuracy in regard to the limiting stresses, and thicknesses are within the error range on the order of ten percent.

Among the external conditions which affect the calculations of the design mass are the maximum longitudinal and lateral overloads, the relationships in distribution of fuel among the compartments, the specific fuel densities, pressures in the tanks, the time-related overload profiles, the specific fuel expenditures and the constraints imposed on the size of units.

Since not all dimensions of the design components may be calculated and selected on the basis of strength-related calculations, one employs the statistical relationships to estimate the mass of the reinforcements, damping partitions, the reinforced structures at the junction points between the units, the cable network, the means which are used to attach the thermal insulation and functional hatches.

The calculations have shown that in determining the stressed state of the shell and of the side frame in a typical fuel compartment, it would suffice to compute three cross sections and associated load-bearing factors, corresponding to the standard bending moments. These are the cross sections at the end and in the middle of the compartment.

The optimization of one variant of the load-bearing and design assembly is being conducted in a packet data mode with the computer time expenditure by using the computer of either BESM-6 or ES-1060 type on the order of 10-20 minutes.

The model of the load-bearing structure, while in the process of design, is supplemented by a special information model, describing the geometric features of components with the appropriate drawings.

The design requirements dictate three avenues for further development of the modeling methods, defining the shape of technical components.

The first avenue is related to the mathematical description and visualization on the screens by computer graphics, of the geometric images and skeletal load-bearing design schematics, of the kinematic depiction of mechanisms, of the configurations of assemblies and components.

The second avenue provides for the construction of typical components of the images and symbols for conditional depiction of cross sections and drawings projections, pertaining to the components of general machinery and equipment profile.

The third avenue is the molding work with mathematical and informational modeling of contours and of the surface outlines. The models of such type have been considered in Chapter 6.

The mathematical models which provide for the development of the second direction are in the form of progammed means which are called, not quite appropriately, the design languages. These means are in the form of program libraries with the requests directed from the central control procedural languages, employing the languages of highest level, most frequently - the FORTRAN. Each geometric component of a drawing is generalized and identified by the language operators at higher level. Among such programs developed by the Institute of Technological Cybernetics, Academy of Sciences of the Byelorussian SSR, and oriented toward FORTRAN language, is the FAP-KF system and also REDGRAF (USSR) system, DIGRA (GDR) system, EUCLID (France) system and ART (USA) system.

The depictions drawn by these means are modeled by a set of elementary geometric objects (segments of arcs and straight lines, /72 curves of the second order, points of fixation) with the traced out orientation of contours, formed by these objects.

The modeling of interacting conditions between the developed contours with the elucidation and depiction of the conditions of intersection, tangential contact, the equidistant displacement, the assigned angular rotation with respect to the poles and axes are represented by the special language programs which are used to depict the component drawings. The most complex part of modeling of such images are the methods of recognition of the interior points of the object, the closed loop evaluation and connectivity in the developed geometric object.

The auxiliary operations necessary for the machine drawing - the calculations and definition of the line dimensions, shadings, markings of the special exterior lines, signs and texts, the perimeter calculation of the contours, of the surface areas and of the cross-sectional moments of inertia - supply the machine-building drawing with all required data, as per the Unified Design System.

In obtaining the image drawing, the information model may become more complicated if one wants to obtain the coordinates of the plotter pen positional motions and the size of such a model is hardly acceptable for storage of the drawings and basic data. There- /73 fore the informational model of the drawing must represent a set of generalized parameters which describe the elementary geometric objects and the conditions of their placement and correlation. In such case, the storage and data base of drawings and images and the tie-in of these images with each other are possible, and can be used as an assembly drawing, employing a set of separate component drawings.

Figures 7.1 and 7.2 show the drawings of typical upper eye components, used to attach the support units to the spacecraft shell.

The mathematical and informational models of drawings were obtained by the design language operators FAP-KF (FORTRAN).

The mathematical and informational models which are designed for automated reproduction of a drawing are conditional at the present time and of primitive nature. They are tied in to the types of cross sections and content of the drawn projections which the user has and cannot describe the component configuration as a whole.

The generalized spatial representation of the shape of a complex component or of an assembly of intermediate complexity, by using computer graphics and employing the mathematical and informational model, remains for the time being a task which cannot be handled by the available programming means.

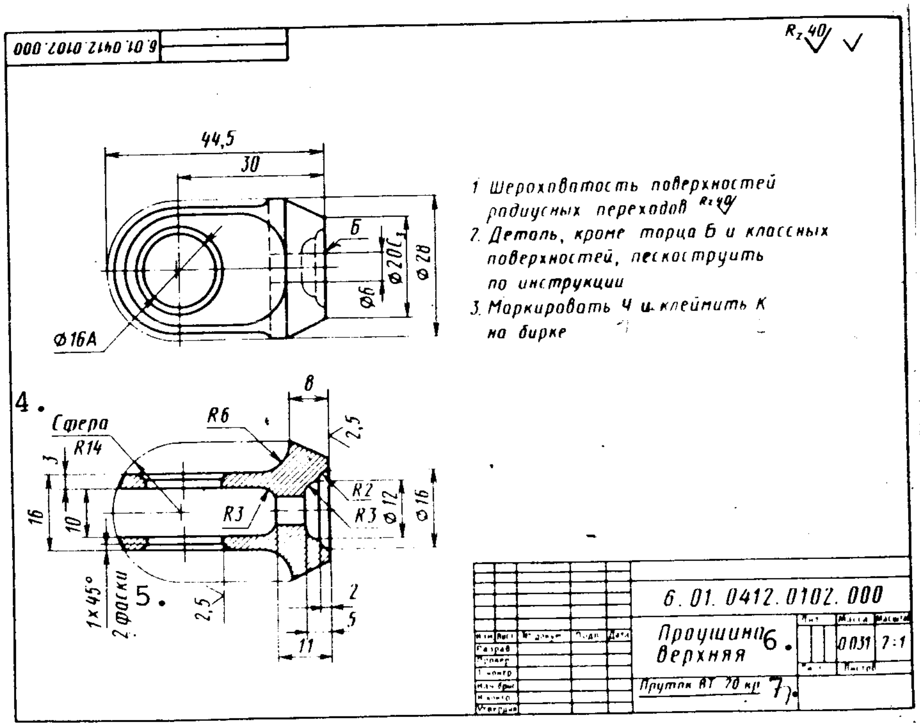

Figure 7.1. Computer graphics drawing of a component which is used to attach the support rods within the major support beams.
Key: 1. Surface roughness of radial tapers; 2. The component except for the end butt 6 and fine grade surfaces should be sand blasted as per instructions; 3. To tag T, and stamp S on the label; 4. Sphere; 5. Chamfers; 6. Upper eye; 7. Rod 7. Rod AT, 70 cr.

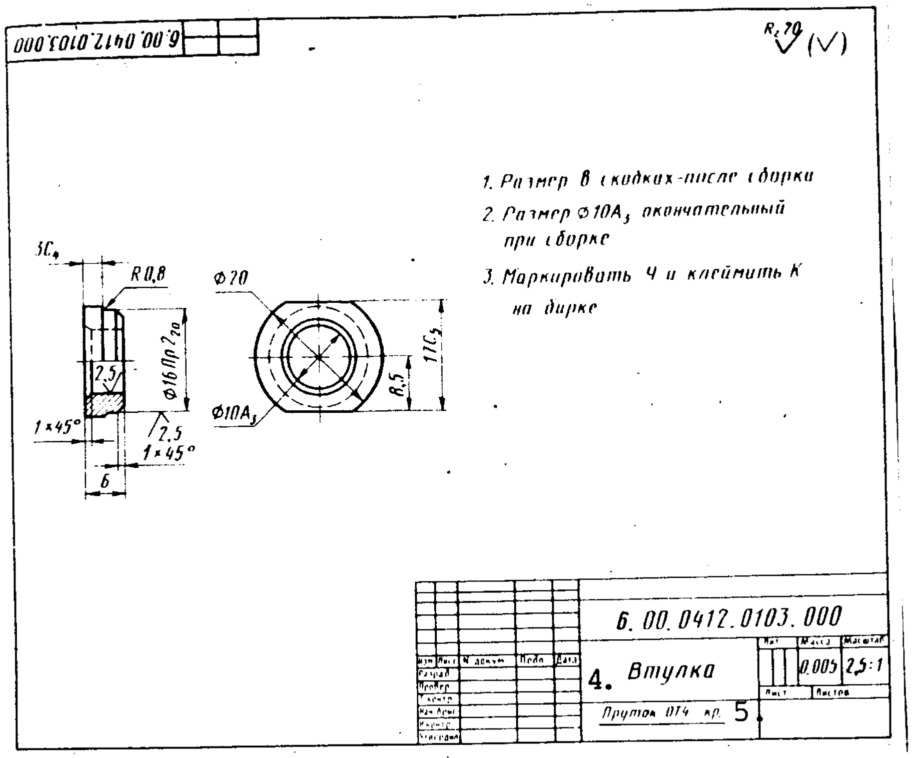

Figure 7.2. Standard support component - drawing made by computer graphics.
Key: 1. Size 8 (after assembly); 2. Size ⌀10A, final after assembly; 3. To tag T, and stamp S on the label; 4. Bushing; 5. Rod OT4, cr.

The geometric depiction of a component of simple shape may be described by using the mathematical modeling, aiming at its storage in the computer memory. Among such components, of great interest are the mass-produced items, which are handled by the programmed control machines, since it becomes possible to manufacture components without master drawing, and transmit the informational configuration model of such component directly from the designer to the technician by using instrumental carriers. /74

Among the components of the spacecraft shell there are several outline-forming parts and some surfaces which earlier were calculated by using the molding approach and nowadays may be determined by placement calculations, supported by mathematical and informational models of the external contours (see Chapter 6). Such component types may be the brackets and fittings for the attachment of instruments or panels to the load-bearing shell, some shelves or parts of the side frame, the wafered paneling of the shell, the lids and reinforcements of hatches, the honeycomb plates and junction segments of side frame components.

There are several methods for mathematical modeling of the configuration of such components and each of these methods is supported by its own type of informational model, which in turn defines the logic structure of the data base.

The most widely used method of mathematical modeling of the outline-forming components aims at the separate description of specific ribs and faces by the piecewise-analytical functions, with separate representation in the informational model of the coefficients of equations which correspond to the outline-forming surfaces and curves, as well as the condition of correlation of such surfaces. The coefficient data follows in a certain order which corresponds to the numbering system of the surface outlines.

By using the canonic description of all surfaces, the algebraic-logic method of external shape description is applicable and can be used to evaluate the position of surfaces with respect to the areas of interior volume of the component in question (60).

The algebraic-logic model of the component outlines is convenient to evaluate the volumes of the segmented component parts which can be encoded in the blanks and is based on the procedure of the point definition within a volume, bisected by a set of surfaces. The model which utilizes the algebraic logic, in describing the surface of a component, indicates indirectly the shape of its interior volume, transforming the procedural conditional-logic relationships into algorithms which can be used by a computer and by employing the appropriate signs, define for the points of interest that they do belong within the volume of the half-space, bisected by the surface in question. The informational model which is constructed on the basis of algebraic-logic relationsh ps is not too applicable for depicting the component shape by computer graphics because of the complexity and long time required to describe the boundaries between the conjugated surfaces.

The mathematical models, developed in recent years, which can use the direct volume component development for the objects in question and which are based, for example, on the methods of matrix stereometry (which have been extensively developed for the shape developments in the assemblies of thoroughly detailed devices by K. M. Nadzharov), are reduced at the final end to the piecewise-analytical functional description, with the time-consuming isolation of the shape boundaries and with the introduction into the informational model of the equation coefficients. /75

One of the few but important advantages of the piecewise-analytical description is the economy aspect of such informational model. The complexity of mathematical description in the case of the designer-users and technicians, the time-consuming and non-standardized conversion of analytical models into visual programs by using computer graphics make the piecewise-analytical modeling methods in describing the component shape not very attractive for design and production purposes.

The alternative method to those which we have considered above is the finite-dimensional point-frame description, with the grid representation of surfaces and fine edges, represented by a set of nodal points. The nodal points on the grid are approximated by a set of gauge-patterned surfaces, extended over the shell by a set of patterned arcs or joined by polynomials with different order of fixation. The outlines or bevels are modeled by linear surfaces of "table cylinder" type. Ordinarily, they are composed of normals at the grid boundary points, approximating the surfaces which are equidistant to the unit outline.

The geometric depiction, developed by the point-frame method, is a conditional model of the component configuration since it may not correspond to its drawn image. It includes for example the points and lines which are outside of the component contours, to be able to assign the trajectories for the processing device or it may show conditionally the intersection of faces at a sharp angle, without rounding off which is obtained by milling. The separate surfaces, represented in such point-frame digital model are fictional ones, since the real processed surface is located in space with the equidistant displacement, equivalent to the thickness of the spacers or coating. The displacement is provided in calculations by computing the segments of normals at the grid nodal points or by assigning a fictional diameter of the spherically shaped die.

The informational model which corresponds to the point-frame geometric component image is of hierarchic, multilevel type and includes within itself the oriented groups of surfaces and bevels, split into separate grids and boundary lines, which in turn include the spaced arcs. The elementary geometric object of lower level is the nodal or contour point with three coordinates, tied in to the constructed planes of the component.

The drawback of the point-frame method of modeling the component configuration is the bulkiness of the informational model, with the need to store more than a thousand numbers for the component of average complexity. The advantages are in the simplicity of data input into the processing programs, and visualization programs, with display by computer graphics. The finite-dimensional linear grid representation of surfaces makes it possible to conduct directly, by using a digital model, the calculations of volumes and surfaces, it enables us to accomplish the expansion and projection on the plane of curvilinear sheet components, to do the trimming and shape refinement because of local design complications (the appearance of grooves, cut-ins, holes and flanges).

The point-frame model of the component configuration makes it possible to position freely in space a component, making it apparent in terms of its general view and projection, and displaying the cross sections of the component on the screens of graphic devices, all of which makes this method of modeling quite promising for the dialogue type of program devices used in geometric design. /76

The shape development of components in this case occurs in the dialogue interaction mode, with the transmission of geometric images from the plotting board to the display screens and employing the automated drawing devices with subsequent recording of digital models into the data bank. In using the dialogue programming of geometric design, the user, operating within the framework of the structural "menu" which prescribes the specific stages of the detail outline transformation, employs the standard procedures of data transmission, regardless of the type of graphic device. The graphic depiction program set for the dialogue geometric design aims at the invariant part of the mathematical and programming support, with the latter executing the geometric spatial constructions and using the languages of higher level, avoids the symbolic codes of concrete displays or automatically drawn depictions, using for the visualization procedures the specialized programs which employ the post processors and the code symbols of coordinate plotters and graph plotters.

The example of means of programming of geometric modeling in the dialogue mode may be the graphic programming set DISPL-2, developed by the Special Design Office of IMU Institute of Cybernetics, of the Academy of Sciences of the Ukrainian SSR, for the ARM-M complex.

The user of such program set may operate concurrently by using several devices of computer graphics if the data, transmitted from one device to another, can be memorized. The major programs in such file are the control programs which define the initial, final and intermediate stages of data transmission, the programs of image transformation as one switches to the coordinate system of the computer graphic device, the programs to change parts of the images on the graphic display screen, the programs of ready-made symbol visualization, the visualization of parts of the text and line segments, the programs for specific image depiction on the screen (brightness, line width, scale). To diagnose the errors at the moment of recoding and transmission of data, the program set is equipped with the control codes. It is also possible to stop the geometric design process and return to the initial position.

The study [48] describes the dialogue process of shape formation by using the programs from DISPL-2, where the component is assigned by the point-frame method. Such process aims at the multicomputer complex with the utilization of a central data bank to store the information about the component models developed and to conduct at the central point the time-consuming point position calculations on the surface outlines.

The standard model of the geometric design process is based on the stage-by-stage refinement of the component configuration. This refinement is accomplished by proceeding from a blank to a simple prototype, joined to the neighboring components. Then one proceeds with the intermediate deisgn stage, superimposing the standard components with specific holes, sheared surfaces,chamfers. Finally, one completes the setup of the geometric image in such form which is applicable for the subsequent automated manufacture. /77

The start of mass production of components on the basis of such data and without master drawings, with the components being processed on the program controlled machines, imposes specific requirements upon the informational model which is developed by the computers. Such model replaces concurrently several traditional documents, namely: the drawing of component and its assembly, the molding process or table of nodal coordinate points for the surface outlines, the data coding map necessary for the program-calculated processing, the data input into the automated control systems which define the mass and centering parameters, the technological documentation which accompanies the project indicating the standard size of the blanks and the variant which is used for the processing and assembly.

The informational model of the component which includes concurrently the design and processing steps, includes the information about the geometric shape, encoded in a set of design parameters (type of materials, type of component, specific tie-in of the component to a unit, subunit or area, as per number of the drawing) and also the technological data which is contained in a special accompanying code (number of blanks of accepted standard type, specific variant of positioning the processing tools and dies).

The information about the geometric image is compiled into a hierarchic system which combines the coordinate points summary data. In contrast to the design data these are simple numbers, recorded just once, or even contained in one number, and it includes different design and technology information.

To develop the informational model of design technology with a concrete data base is possible by using the data bank hierarchic structure. Such structure reflects the multilevel principle of putting together an aggregate and separating the device into units, subassemblies, subunits and separate components. The hierarchic principle of the logic structural data bank construction aims at a multilevel catalogue, at the lower level of which one generates the basic information about the component which is submitted to the data archive. In the catalogue, each recording is connected only with the data at the higher level, controlling the system address functions which are subjugated to the levels of lower rank.

The application for work with the information design models which are known within the Component Manufacture Control System, developed for the Automated Control Systems, is not always justified because it is impossible to address directly the data base, extracted from the mathematical modeling programs which define the component shape, it is not always justified because of large operational memory consumption and because of the difficulties for user to master the complex logic structure with the excess service program capabilities.

The users of the design and technological data base request it for reference or conduct the recording themselves of the geometric model and of service data. The planning and dispatcher services limit their functioning by obtaining from the catalogue the generalized parameters of blanks and the number of components of different types. The technologists and programmers who are involved in the mechanical processing, by using the geometric component images, develop the control program for processing equipment and the means for geometric monitoring of components and equipment. For the users /78
of the dialogue systems of geometric design, the data base is the general systemic nucleus of the whole process in component development. The leaders of enterprises and of separate subservient organizations assume that the major purpose of the data base is the interface between the automated design systems and manufacturing process. It is possible by using the means of functional computer systems of ES type, which employ the FORTRAN language, to develop the design and technology data base, and organize direct access files by using the disk devices and employing external memory.

The informational design and technology model contains data which is represented in the traditional design work by a number of important drawings and production documents and therefore, the control methods, the methods of upgrading, correction and protection of the data bank information must be clearly defined by the appropriate instructions and must be adhered to by all users of the automated component development process. Under such conditions, one can single out the role and function of the administrators of the data base who are to review and control all information, maintaining the archives in operational state, with the periodic dissemination of copies and documentation of the state in which the data base recording is found.

The transmission to production channels of the informational component models is accomplished by means of the computing center channels of communication or by means of the magnetic tapes. Each act of data transmission is accompanied by a protocol with the identification of the recorded copy content, using the summary data, the summary information about the components and some other control requirements. The transfer protocols are approved by the data base administrator's signature and by the people in charge of the calculation and design offices. From that point on, they acquire the legal force which is analogous to the traditional design documentation.

The constant upgrading which is associated with the refinement of the design concept and correction of the errors results in the need to take into account at the same time several informational models which differ in terms of the date when they can be obtained. In the case of the same components, one has simultaneously the theoretical standard informational model which is currently and continuously upgraded and refined by the design bureau. Secondly, the production standard informational model which corresponds to the theoretical one, at the moment of the last upgrading, and thirdly, the production current informational model which takes into account the errors at the point of junction and manufacture of the concrete assemblies and components, incorporated in each spacecraft.

The information about deviations from the production current data model from the theoretical standard models must be periodically upgraded within the framework of the integrated automated process, supported by the System of the Automated Project Design (ADS) and by the ASTPP. They unify the work of the design bureaus and of the experimental and manufacturing enterprises.

CHAPTER 8.

INFORMATION MODELS OF THE PROCESSES AIMED AT THE ACCOMPLISHMENT OF THE PROJECT

The purpose of informational models which accompany the design project as it is being brought about is, first of all, to provide the leadership and design developers with the objective information about the status of the project fulfillment and about the results of the projected work, and secondly, to prognosticate the possible design parameters. /79

To fix the status of operational work and the corresponding time schedules is the major task of the coordination and automated control system services and quite a few articles and standardized recommendations have been published at home and abroad.

Let us consider the second aim in the development of informational models which are to service the design organizations and for which the qualitative prognostication of the technological flight parameters is a proof of the positive results and quality of their work.

The analysis of current parameters of the object which is being designed is based on the data base which accompanies the project and on the applied program sets which define the dimensions and mass as well as the technical parameters, in order to achieve the goal-oriented effectiveness.

The data base which accompanies the project aims at the users in the design offices, at the coordinating services and as a rule, includes the following data:

the data on aerodynamic characteristics, obtained by calculations and by the upgraded experimental results;

the data on loads and different computational cases;

the data on mass, centering parameters and inertia parameters;

the data on the technological flight parameters.

The logic structure of the data base, the means used for diagnostics and safety must provide for functional upgrading of information about the design project, utilizing as a standard a set of programs and calculation retrieval systems which are available at the design offices, as well as the specialized program complexes for the calculations and verification of strength, dynamics and heat exchange.

The informational and reference system of the chief designer is supported by the data base and is equipped with the managerial, problem-oriented language which ensures the assembly of computational sequential programs, with data input and output supported by the data base. The prognostication of trends in the change of technological flight parameters requires a periodic processing and determination of the current state for a given project design, with evaluation of possible limiting deviations of the computed and experimental data, with respect to the project criteria. The methods of determining the parameter scattering on the basis of the initial data fluctuations may be made a basis of the altorithm which could be used to model the possible and limiting deviations in the criteria, on the basis of the optimistic, average and pessimistic estimates.

The data base which is used in the project design must satisfy a number of specific requirements, of which the following are the major ones:

1) maximal simplicity and convenience of use;

2) the ability to work with the data base by recalling the applied programs; /80

3) minimal size in the operational memory allocated for the servicing programs;

4) flexibility of logic structure, with the ability of adapting it toward the change of the design tasks and the nomenclature change while exchanging the modules;

5) the ability of adapting for a more comprehensive, in terms of designation, identification and dimensions of the same data which is produced in the form of results and ability to retrieve the applied programs developed by different and separate users;

6) correlation between the hierarchic structure of the data in splitting the spacecraft into the subsystems, with the ability to sequentially conduct calculations of different design parameters.

The developed and known data bases (OKA, KAMA, INES and others) developed to handle the automated control system problems are not very well applicable for specific design calculations of the spacecraft, because of the complexity of use, because of the overload of operational and reference programs, because of large required memory and the absence of means which make it possible to utilize the data from the applied programs, written in FORTRAN and PL-1 languages.

One of the important attributes of the program availability for the data base under consideration is the algorithm for the recognition of the informational insufficiency, with subsequent effective diagnostics and development of recommendations for the defect removal, in preparing the data for calculations with respect to each design assignment.

A simple and widely used approach, making it possible to correlate the separate programs is a timely development of special individual junction point subprograms which accompany the entry and exit of each module. These unique post and preprocessors can retransform the results and initial data, in accordance with the unified method of information representation in terms of the data base.

The data base for the technological flight parameters controls and generalizes the most important information related to the project which contains the basic tactical and technological parameters and information about the spacecraft use. Therefore, it is equipped by special safety means and by a special control means. The control and diagnostics are also needed in conjunction with the importance of the decisions which must be taken by the leadership of the project design which is already in the process of manufacture.

The safety provisions are of three types: equipment, programs and procedural (introduced by the appropriate organizational measures).

Among such measures are the development of special operational modes in the machine shops as the data from the computers is being transferred, the documentation in accordance with the special instructions and statutes, the storage of the data carrier in special safes with regulated access, the designation of individuals responsible for the data introduction and data retrieval within the subject matter departments of the Special Design Office. The safety of equipment which is much more difficult to bring about consists in introducing into the telecommunication lines the encoders and decoders and in protecting the video images generated by the special devices. The program means for protection and safety bring about the encoding and decoding of data as it is being fed to the terminal printout or video display. It provides for blocking and code words during the /81
data recording and while retrieving it simultaneously from several archives or from the hierarchically higher data bank catalogues. It provides for the printout of the most important data in a special form, it defines the data recording sessions and keeps track of the personal retrieval by the users who are using the program library and data base.

One of the most well developed, in the practice of design, is the mass and centering model, which is supported by the data base pertaining to the mass, centering and inertia parameters. This model is an integral part of the technological documentation development since the control of the mass is accompanied by a number of drawings, and the analysis of relationships between the current and limiting masses is the main instrument for the determination of project design conditions.

The control operations of the mass-centering and inertia parameters of spacecraft are accompanied by the sketch and operational design. They become a part of the general design work, beginning with the mass and dimension calculations as the general features of the spacecraft begin to develop, ending with the control of finished execution of production, as required by the drawings, pertaining to the mass of components and balancing of the assemblies.

Hence - the major tasks of control:

1) verification of reliability of the computed model used at the initial stage of design;

2) well-founded distribution of the mass limits in terms of the components at the beginning of design work, in accordance with the general layout of the whole aggregate, and then during the subsequent operational stage - in accordance with the draft separation into the assembly and subassembly units, into separate units and components;

3) prognostication of the actual masses of the spacecraft components with a flexible distribution of the mass safety reserves during the project design and during construction;

4) functional determination and issuance of the centering and mass-inertia parameters to the responsible users;

5) analysis of the current distribution of mass-centering and inertia parameters on the basis of the draft data and manufacturing data and preparation of measures to facilitate the construction;

6) control of the current state of development and refinement work on drawings, with subsequent readiness to submit them to the production departments;

7) assembly and processing of data about the actual mass, inertia and centering parameters while the aggregates, assemblies and components of the spacecraft are being manufactured.

The most difficult is organization of such control at the initial stage of design (to optimize all aggregates in terms of masses) and at the end stage - in monitoring the production process (to estimate the actual mass).

In the first case, the automation of calculations runs into the need to combine the obtained results of optimal mass distribution in terms of the construction design load components with the approximate and not very convincing estimates of the masses for the components and parts which are not of load-bearing type, plus the appropriate mounting devices, all of which is done by analogy with the estimates of similar construction designs.

In the second case, the mass distribution, in accordance with /82
the specifications of the drawings for the assembly items which reflect later on the combined product, is in contradiction with

the technological shop manufacture of components which must take into account the methods of manufacture, thermal processing and the presence of standardized components which are incorporated into different assembly units.

The mass control must produce the preliminary estimate of the mass-centering and inertia parameters. It must assist the designers in calculating the mass of components and of assembly units, it must control the assembly itself and accumulate and store the computerized data, it must ensure the processing and analysis of information, it must generate operationally the data with respect to specific assemblies and with respect to the finished product as a whole at all stages of design, construction and manufacture of the spacecraft.

The data base of mass-centering and inertia parameters which embraces the activity of the design bureau operates as a rule with the data pertaining to the development of the spacecraft frame components, taking into account only in rough terms the design of ready-made products (propulsion units, instruments, various pieces of equipment, landing devices and service equipment). It is therefore necessary to develop the interrelated operative data base which aims at the unified type of computer, with centralized data processing, generated by the manufacturing enterprises and from the factories in the neighboring fields, which belong to different agencies.

One of the tasks of control which is to be solved jointly with the systems in selecting the optimal load factor for various design components, and the subsystem which is involved in the design of other subassemblies, is the automated search and retrieval of data for the best way and levels of unification and standardization of the design components, various cables and different pieces of equipment.

In the course of construction, the information from the data base of the mass-centering and inertia parameters undergoes changes which corresponds to the refined data about the whole project. In the beginning, the operations are conducted with limiting masses which, as the design documentation develops, are supplemented by the refined drafts, developed separately and replacing the limiting masses. Later, as the spacecraft is being manufactured, one must take into account the actual mass and also the centering and moments of inertia on the basis of the obtained results, by weighing and balancing the different assemblies.

PART II

FUNDAMENTALS OF MATHEMATICAL MODELING AND DESIGN WORK FOR SPACECRAFT TESTING /82

The theoretical studies of the basic operational modes of motion of the spacecraft during its active flight, involving the flight trajectory separation, in terms of the orbital flight, descent flight and parachute descent through the planetary atmosphere, its soft touchdown and further functioning on the planetary surface is one of the most important stages of the design work and refinement operations on the ground in the case of space technology prototypes which appear to be promising. In the second part of this book, in addition to the above-mentioned questions, we shall consider /83 the fundamental principles of the project design in testing the spacecraft: the theoretical methods of planning, the scientific experiment with automated components, the approaches used to process the results of spacecraft tests by the methods of mathematical statistics and probability theory, so as to randomize the external effects and take into account the errors in the initial data and in the obtained results.

Particular attention is devoted to the evaluation of reliability and maintenance of its operational status, by employing the test equipment mounted on special stands. Some aspects of these questions were analyzed in the studies [15, 27].

CHAPTER 9.

MATHEMATICAL MODELS OF SPACECRAFT MOTION WHILE PLACING IT INTO THE ORBIT, THE SEPARATION AND DESCENT THROUGH THE PLANETARY ATMOSPHERE

Once the spacecraft is launched as per program, depending on its purpose, one may isolate a number of stages and phases: the active flight through the Earth's atmosphere, when the propulsion unit is functioning, the booster separation, detachment of the front cowling, activation of various components, the detachment of some parts, flight through space, entry into the planetary atmosphere, descent, touchdown and operation on the planetary surface (Figure 9.1).

During the stage of active flight, the operational conditions are characterized by a program of the thrust change in the propulsion units, by the inertia forces, by the trajectory parameters, the meteorological conditions in the area of blast-off, by the aerodynamic forces, taking into account the random wind action, by acoustic effects and vibrations of a broad spectral range (5-2500 Hz) which are associated with the pulsating thrust, fuel and internal oscillations of the spacecraft itself.

During this stage of the spacecraft functioning, the maximum loads on the components are toward the end of the operational stage, when the current level of the thrust is at a maximum and the spacecraft mass is at a minimum (n_x=4 ... 10).

Of special importance is the moment when the first stage booster thrust is cut off and the device undergoes maximum, sign-changing oscillatory motion. The active load in this case may be higher by a factor of two and more than the figures determined, disregarding the dynamics of the moment of cut-off.

The flight vibrations are of complex nature and are characterized experimentally by monotonically increasing longitudinal overload-frequency factor $n \simeq 0.35\nu^{0.46}$.

The duration of vibration (τ) for a typical spacecraft is 600-750 s.

The levels of load factors and frequencies define unambiguously the oscillation amplitude (in millimeters):

$$A = \frac{250n}{\nu^2} .$$

At high vibration frequencies, the oscillation amplitudes are small (ν=2000-2500 Hz, A=1 μm) and there is no particular danger for the spacecraft. Of great importance in the case of mounts of separate units are the frequencies which are similar to their internal frequencies (5-200 Hz). In this case, one must add to the maximum static load factors the vibration load factors (n_v=1 ... 2). The safety factor is at a minimum (f=1.2 ... 1.3). /84

In the case of the frontal cone, for calculations one assumes the conditions of the maximum velocity drag (35.0 kPa) and the angle of attack which, because of the flows, may reach ~6-8°.

At the stages of separation, activation of the components and various component detachment, the operational conditions are characterized by activation of the pyrotechnical means, unfolding to a dead stop of various functional components, the activation of docking and mating components and others.

The separation of stages is ordinarily accomplished by using the explosive charge devices or accumulative charges. The activation of these devices generates high pressures and other forces during the time period which is in milliseconds which creates in the adjoining areas the short-term overloads, reaching several hundred units. The unfolding and activation of collapsible paraphernalia on the spacecraft (of the antennas, solar batteries and scientific and research instruments) many of which are of folding type because of size, is accomplished by spring-loaded devices, by the torsion devices and other gadgets.

Because of frictional differences in hinges and lack of flexibility in cables which connect various sections of the mechanisms,

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the unfolding devices may acqurie increased angular (and linear) speeds of motion (ω=1 ... 1.2 rad/s).

The kinetic energy of moving components $K = \frac{I\omega^2}{2}$ (I is the moment of inertia of the moving components) must be compensated by the deformation energy of support components.

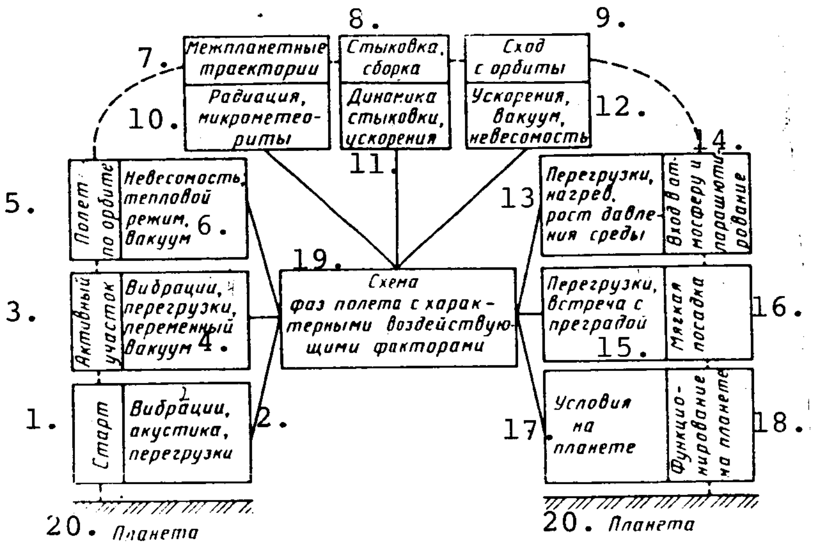

Figure 9.1. Schematic diagram of different phases of the spacecraft flight with characteristic active factors.

Key: 1. Blast-off; 2. Vibrations, acoustic, load factor; 3. Active flight; 4. Vibrations, load factors, variable vacuum; 5. Orbital flight; 6. Weightlessness, thermal mode, vacuum; 7. In the planetary trajectories; 8. Docking, assembly; 9. Departure from the orbit; 10. Radiation, micrometeorites; 11. Dynamics of docking, acceleration; 12. Accelerations, vacuum, weightlessness; 13. Load factors, heating, pressure increases; 14. Atmospheric entry and parachute activation; 15. Load factors, obstacle encounter; 16. Soft touchdown; 17. Planetary conditions; 18. Operational functioning on the planet; 19. Schematic diagram of different phases of the spacecraft flight with characteristic active factors; 20. Planet

By assigning the deformation shape on the basis of energy equivalence, one can determine the stresses within the design components. In addition, a part of the kinetic energy may be absorbed by the braking devices (springs, spacers). In such cases, /85 the safety factor is increased by the increased computed angular velocity (f=1.2).

At the stages of the spacecraft descent through the planetary atmosphere and during the touchdown on its surface, the operational conditions are characterized by the functioning of the parachute or parachute-propulsion systems, by the atmospheric descent trajectory parameters, by the thermal stresses and by the changes in barometric pressure, by the meteorological conditions at the site of the descent and touchdown and by the specifics of soil at the touchdown area, as the spacecraft makes contact with the planetary surface.

The generalized safety coefficients which connect the real stresses with the computed ones by using the design mechanics include the random components, which may be theoretically analyzed.

The isolation of random components in the form of cofactors and determination of these components as a function of the design parameters make it possible to lower the construction design mass.

In the static problems, the safety factor can be easily related to the probability of fail-free functioning of the design:

$$\Phi = \Phi\left[\frac{(f-1)N_a}{\sqrt{D_a + D_{cc}}}\right],$$

where $\Phi(x)$ is the integral or probability, N_a is the acting load, f is the safety factor, D_{cc} is the dispersion of carrying capacity of the component and D_ℓ is the dispersion of acting load.

For the actual factors of the load variation and of the carrying capacity, the probability of fail-free operation, corresponding to the safety factors used (f=1.3 ... 1.5), is in the range between 0.96-0.999.

The analysis of dynamic models is more complex. The functional cut-off of the propulsion unit after the maximum overload, the landing load factors of the descending device, the stresses during docking and separation of the devices - these are typical cases of dynamic load exposures.

The transition processes which occur after a rapid load application are related to a rigid design construction of the spacecraft and are characterized by the factors associated with the load increase β, which define the extreme points of stresses as compared to the random static loads.

For the spacecraft, its shell and onboard systems, it is of practical interest to analyze the lateral oscillations as the dynamic forces are applied.

In considering the spacecraft in its first approximation as a single mass console system, mounted on a flexible support, one can estimate the enhancement factor of the lateral loads by using the following formula

$$\beta = \frac{1}{\sqrt{\left(1 - \left(\frac{\omega}{p}\right)^2\right)^2 + \left(\gamma \frac{\omega}{p}\right)^2}},$$

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where ω is the frequency of one of the lateral carrier oscillations, p is the frequency of the spacecraft internal oscillations and γ is the damping factor.

It is clear that the enhancement factor associated with the lateral loads β is a random quantity. In addition to the assumed or known frequencies of lateral carrier oscillations ω and the spacecraft internal oscillations p, of great importance and effect on the scattering of β values is the damping factor γ. The design-inherent damping is to quench the oscillations by ways of friction of the contact surfaces, by impacts within spacings and cyclic deformation of the material, all of which is connected to the logarithmic decremental oscillations ($\gamma=\lambda/\pi$). In the case of typical design constructions $\lambda = 0.15 \ldots 0.25$. The methods of damping which are used in some systems and assemblies (the hydraulic systems, sensor) cannot be used for the spacecraft. The load factor at the center of the masses $n_0\beta$ is determined on the basis of the lateral load factor at the junction point with the oscillation carrier n_0. Assuming that the load factor changes linearly, one can determine its value at any point.

In the region which is close to the resonance region ($\omega=p$) the overloads may reach high levels (n=4 ... 5) which is not permissible.

In designing the spacecraft and its attachment systems to the oscillation carrier, one can affect the enhancement coefficient by aiming at $p=1.3\,\omega$ condition. In such case the computed lateral load factors will significantly decrease.

The mathematical expectation $\overline{\beta}$ and dispersion D_β of the enhancement factor, related to the lateral overloads, may be estimated by using the following formulas

$$\bar{\beta} = \frac{1}{\sqrt{(1-(\frac{\omega}{p})^2)^2 + (\frac{\lambda\omega}{\pi p})^2}}; \quad D_\beta = (\frac{\partial\beta}{\partial\omega})^2 D_\omega + (\frac{\partial\beta}{\partial p})^2 D_p + (\frac{d\beta}{d\lambda})^2 D_\lambda,$$

where $\bar{\beta}$ is the mathematical expectation of the enhancement factor, ω, p, λ are the mathematical expectations of the incorporated parameters and D_ω, D_p, and D_λ are the dispersions of incorporated parameters.

The derivatives will have the following form

$$\frac{\partial\beta}{\partial\omega} = \frac{2\omega\beta}{p^2}(\frac{\lambda^2}{\pi^2} + \frac{\omega^2}{p^2} - 1); \quad \frac{\partial\beta}{\partial p} = \frac{2\omega^2\beta}{p^3}(1 - \frac{\omega^2}{p^2} - \frac{\lambda^2}{\pi^2}); \quad \frac{\partial\beta}{\partial\lambda} = \frac{\omega^2\beta}{\pi p^2}.$$

The loads N_a, linearly connected with the enhancement factor, are characterized by the mathematical expectation $N_a = F\bar{\beta}$ and by the dispersion $D_\ell = \beta^2 D_F + F^2 D_\beta$, where F is the static load, β is the enhancement factor, D_ℓ is the dispersion of acting load and D_β is the dispersion of the enhancement factor.

Reliability Estimate of the Separate Block Separation System in the Spacecraft

The random character of separating the spacecraft blocks is due to two groups of perturbation factors.

The first group is the danger of "catch up" phenomenon, of the longitudinal collision, of the mass deviation in the objects of interest and of the delayed action in safe-break bolts. /87

The second group includes the perturbations associated with the lateral displacements and with the rotation (the spatial character of motion): the eccentric action of the propulsion unit thrusts, the time difference when the bolts break, the elastic oscillations of the shell, etc.

The basic method in determining the parametric component of the probability of fail-free block separation, in a general case, is the method of static modeling of the perturbed spatial motion of separable objects.

As starting data to model this phenomenon, one utilizes the probabilistic parameters of perturbations (the laws of mass distribution, effect of the applied thrust, of the forces which ensure separation, etc.) and also the process model (the system of differential equations which describe the separation event) as well as the statistical information associated with field tests of aggregates, systems and components.

The relative velocity V_{rel} of the block and the last stage in terms of time t_k, which terminate the separation process may be used as the parameter of state which defines the probability of fail-free separation. In such case, it is necessary and sufficient that the relative velocity at the moment in time t_k would be positive, in other words, $V_{rel}>0$.

Let us consider the separation process of a block or module by the method of braking during the last stage (considering only the longitudinal relative motion, acted upon by the total force F, generated by the braking propulsion units and the aftereffect thrust P in the propulsion unit which functions at this stage).

The relative motion is described by the following equation

$$\frac{\partial V_{rel}}{\partial t} = \frac{F-P}{M} .$$

By integrating we will find that $V_{k.rel} = \frac{1}{M}[\int_0^{\tau} F(t)\partial t \quad \int_0^{t_k} P(t)\partial t] = \frac{1}{M}(I \quad Q),$

where τ is the operational time of the braking propulsion units ($\tau<t_k$), I is the thrust pulse of the braking propulsion units, Q is the aftereffect of the thrust pulse and $V_{k.rel}$ is the known function of three independent and random quantities M, I and Q.

The distribution of random quantities M, I, Q is subject to the normal law and the mathematical expectations m_M, m_j and m_Q as well as the dispersion σ_M^2, σ_I^2 and σ_Q^2 are known by analogy with the field testing or are obtained by statistical modeling, respectively.

To determine the mathematical expectation m_V and the root-mean-square deviation σ_V of the random relative velocity $V_{k.rel}$, let us utilize the linearization method. In such case, we will obtain

$$m_V = \frac{m_I - m_Q}{m_M}; \quad \sigma_V = \frac{\sqrt{m_V^2\sigma_M^2 + \sigma_I^2 + \sigma_Q^2}}{m_M}.$$

Then the probability of zero coimpact is calculated by using the following relationship

$$P_{sep} = \text{bep}(V_{k.rel} > 0). \qquad (9.1)$$

If the normal distribution of random quantity $V_{k.rel}$ is used, the expression (9.1) will acquire the following form

$$P_{sep} = \Phi\frac{m_V}{\sigma_V}),$$

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where $\Phi(\frac{m_V}{\sigma_V})$ is the table function.

In contrast to the preceding separation of the blocks or modules, we will have

$$\frac{dV_{rel}}{dt} = \frac{F-P}{M} + \frac{F}{m_b},$$

where m_b is the mass of the block or module, F is the summary force of the device which is used for separation and M is the mass of the last carrier stage. The remaining calculation of P_{sep} is analogous to the braking stage.

In calculating the reliability of concrete blocks or modules, one knows the composition and interaction between the incorporated components. Consequently, we can calculate the system reliability on the basis of quantitative parameters of component reliability, obtained by field testing or from reference data.

In such case, the calculation of reliability is based on the structural reliability diagram (SRD), making it possible to estimate the system reliability on the basis of the reliability parameters of components or by selecting, on the basis of comparative analysis, the structural system which would ensure the greatest (or required) reliability.

It is assumed that the system consists of a finite number of components and that the failure of the system may be the result of one component failure or failure of a group of components.

Let us consider the system of the block or module separations consisting of break-free bolts, braking propulsion units and a plunger which functions independently, from the point of view of safety.

In calculating the reliability of blocks or modules, one is to utilize the following relationship

$$P_{SRD} = \prod_{i=1}^{k} P_i, \tag{9.2}$$

where P_i is the reliability of incorporated components,

$$P_i = 1-(1-P_1)(1-P_2) \ldots (1-P_j),$$

in other words, P_i is the reliability of a group of components j, which are joined in parallel and k is the number of components (or of groups).

In such system, the (9.2) relationship will acquire the following form

$$P_{SRD} = P_{bb}^{n_1} P_{pu}^{n_2} P_p,$$

where P_{bb} is the reliability of break-free bolts, P_{pu} is the reliability of braking propulsion units, P_p is the reliability of the plunger and n_1, n_2 is the number of corresponding components.

Since ordinarily one knows only random estimates of component reliability $\hat{P}_i$ and their root-mean-square deviations σ_{P_i}, by using the $\hat{P}_{SRD} = \hat{P}_{bb}^{ni} \hat{P}_{pu}^{n2} \hat{P}_p$ relationships, one can estimate the reliability $\hat{P}_{SRD}$ of the system under consideration and by using the relationship

$$\sigma_{P^*_{SRD}} = \sqrt{\sum_{i=1}^{k} \left(\frac{\partial P^*_{SRD}}{\partial P^*_i}\right)^2 \sigma^2_{P^*_i}}$$ /89

one can obtain the root-mean-square deviation of the random estimate of the system reliability P^*_{SRD}.

The lower confidence level of estimated P^*_{SRD} is computed by using the following expression

$$\hat{P}_{rSRD}=\hat{P}_{SRD}-t_{\gamma}\sigma_{P^*_{SRD}}$$

where t_{γ} is the quantile for the assigned confidence probability γ and $\sigma_{P^*_{SRD}}$ is the root-mean-square deviation of the estimate.

The final expression for the reliability system for blocks or module separation, and its parameters, are calculated as a product of parametric and structural reliability, in other words by using the following formula

$$P_{sep}=P_{sepSRD}P_{par}.$$

One ordinarily must provide the system reliability not lower than the specifically required one, in other words

$$P_{r}\geq P_{req}. \quad (9.3)$$

If it turns out that the condition (9.3) is not provided, it is necessary to carry out the following measures: to increase the reliability of incorporated components, to introduce a set of backup components, to achieve decrease in the root-mean-square deviations of estimates and to facilitate and improve the conditions of the separation system functioning.

Spacecraft Flight Diagrams Toward the Planets of the Solar System

The spacecraft flight diagrams are selected depending on the power expenditures, on the total flight time, on the size of the take-off window, on the sensitivity of nominal trajectory to the possible perturbations, etc. The optimal trajectory and ballistic parameters correspond to the extremal values of one of the criteria, with constraints imposed on the other criteria. However, the utilized analytical relationships, the character of search for the optimal design and ballistic parameters and the obtained results depend significantly on the spacecraft flight schematic selection. The flight selection enables us to determine fundamental ways to solve the concrete task in planning the interplanetary flight.

A great variety of variants in terms of the heliocentric flight segments toward the planets result in a considerable increase in the number of possible flight profiles which are formed by combining the fragments of known types of heliocentric flight orbits [64].

The diagrammatic flights toward the planets may be separated into two types: the first one is the flight toward the planet of interest, without the spacecraft return to the Earth, and the second one is the flight toward the planet of interest with the spacecraft return to the Earth (Figure 9.2). The diagrammatic flights of the first type are characteristic only for the automated spacecraft, the main task of which is to deliver the useful payload into /90
the area of interest and to obtain scientific information about the planets of the Solar System. The flights toward the planets of interest may be of several types.

The journey by using the target flight trajectory (Figure 9.3) commences with the intermediate orbit near the Earth and is terminated either by flying in the vicinity of the planet by using the hyperbolic planetocentric trajectory, or by direct targeting of the spacecraft into the assigned planetary region.

The specifics of flight which includes the step of approaching the planet and landing on it (Figure 9.4) lies in the fact that after departing from the intermediate orbit near the Earth and in approaching the planet, one employs the planetary landing device. Within such framework of flight, either the whole spacecraft or separate modules may be landing on the planet.

In the combined flight pattern (Figure 9.5) one finds such flight diagrams when at the approach segment of the trajectory toward the planet, the interplanetary spacecraft is separated into modules, some of which fly in the vicinity of the planet (the passing trajectory) or are placed into the orbit near the planet (the orbital diagram), and some other modules may land on the /91
planetary surface. Another flight pattern is possible, when one module is placed into the near-planetary orbit and the other one - flies past the planet.

The return to Earth is the requirement imposed upon some automated devices which deliver the useful payload and the major requirement which is imposed on the interplanetary manned spacecraft. The patterns common for all these flights toward the target planet, with return to the Earth, is the Earth's atmospheric entry of the spacecraft with the velocity which exceeds the second cosmic velocity. It is permissible to use the activated pulses at the points of heliocentric parts of the return trajectories. Such pulses make it possible in a number of cases to reduce considerably the entry speed of the spacecraft into the Earth's atmosphere.

The schematic diagrams of "Earth - target planet - Earth" may also be of different types. The spacecraft landing may be accomplished by direct landing in which the hyperbolic arrival trajectory is either directly joined to the descent trajectory or with the near-planetary orbit, into which the spacecraft is first

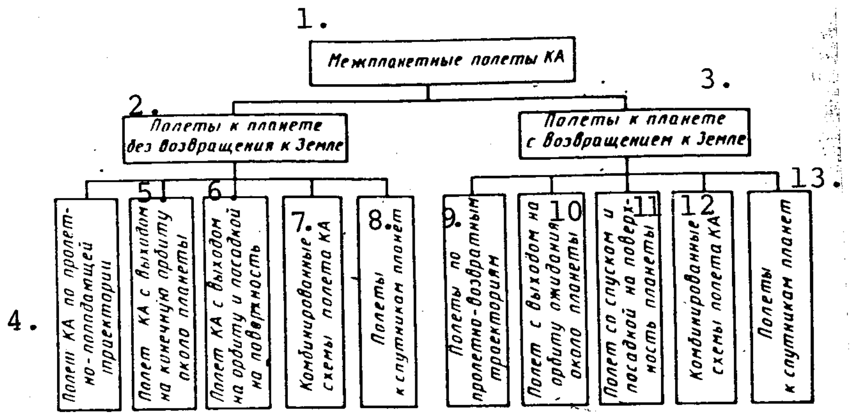

Figure 9.2. Diagrammatic classification of the spacecraft interplanetary flights.

Key: 1. Spacecraft interplanetary flights; 2. Planetary approach without returning to the Earth; 3. Planetary approaches and return to the Earth; 4. Spacecraft flight using the target flight trajectory; 5. Spacecraft flight with placement of it into the end orbit which is near the planet; 6. Spacecraft flight with placement in planetary orbit and landing on the planet; 7. Combined spacecraft flight diagrams; 8. Flights toward the planetary satellites; 9. Space flight using the shuttle type trajectories; 10. Flight with placement of the spacecraft into the waiting orbit, near the planet; 11. Flight with descent and landing on the planetary surface; 12. Combination spacecraft flight diagrams; 13. Flights toward the planetary satellites.

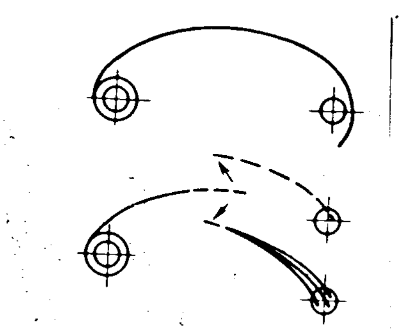

Figure 9.3. Spacecraft flight diagram by using the target flight trajectory.

placed. To return to the Earth, it is necessary to take off from the planet, it involves the active flight trajectory to place the spacecraft into the near-planetary orbit, with the subsequent departure from it.

In the combination flight patterns with return to the Earth (Figures 9.6, 9.7) the task of interplanetary spacecraft flight near the target planet (or placing the spacecraft into the near-planetary orbit) is combined with the landing module which separates from the spacecraft and lands on the planetary surface and the latter step may be accomplished by using the preceding flight pattern. The spacecraft return to the Earth may also be organized by various methods.

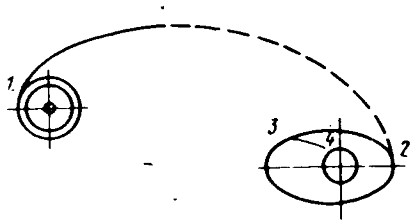

Figure 9.4. Spacecraft flight diagram with placing it into the near-planetary orbit and landing a module on the planetary surface:
1 - departure from the intermediate orbit; 2 - placement of the spacecraft into the near-planetary orbit; 3, 4 - the descent trajectory segment on the planetary surface.

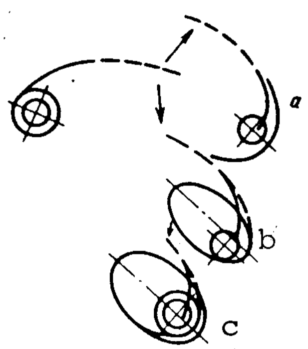

Figure 9.5. Spacecraft combination flight diagram:
a - target-landing diagram;
b - orbital flight with the planetary touchdown and active braking to attain the final orbit; c - orbital-landing diagram with the aerodynamic braking to reach the final orbit.

Spacecraft Landing Trajectories on the Planets with Atmosphere /92

In the course of descent by using the target trajectory, a serious problem is to provide the accuracy of control, making it possible not to employ the skip trajectory, avoiding at the same time excessive overloads and aerodynamic heating. In the course of the planetary atmospheric entry, the errors in control are very dangerous because the steep entry and correlated high load factors may result in the destruction of the spacecraft and very oblique entry may result in the departure of the spacecraft into space, without return.

High power expenditures are required in the course of orbital descent from the planetary satellite. However, the orbital descent on the planet has the advantages that the load factors are lowered and thermal loads on the spacecraft are also lowered because of the atmospheric angle of entry. In addition, the range of landing sites is also expanded.

The spacecraft approaching the atmosphere possesses high stored energy (~100 GJ) which is the sum of kinetic energy of the device and its potential energy with respect to the planetary surface. This energy must be primarily dissipated during the stage of the atmospheric entry. The most economical way of spacecraft braking during the descent on a planet which has atmosphere is the aerodynamic braking.

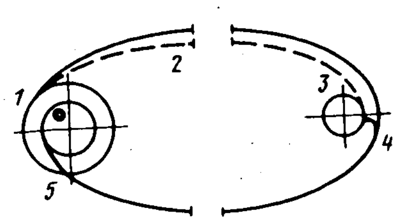

Figure 9.6. Combination flight pattern of the spacecraft returning to the Earth, with the flight around the target planet and landing on its surface: 1 - departure from intermediate orbit; 2 - correction of interplanetary trajectory; 3 - spacecraft landing; 4 - take-off from the planetary surface; 5 - landing on the Earth.

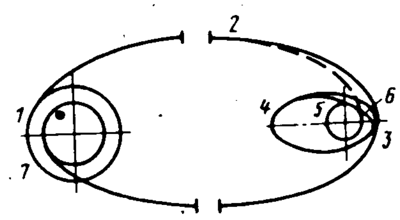

Figure 9.7. Combination flight pattern of the spacecraft which is to return to the Earth with the placement of the spacecraft first into the orbit and then landing on the target planet: 1 - departure from the intermediate orbit; 2, 5 - the variants of direct landing of the spacecraft; 3 - braking step to reach the final orbit; 4 - descent from the orbit to the planet; 6 - take-off from the planetary surface, attaining the orbital flight pattern; 7 - landing on Earth.

The dynamics of spacecraft motion in the course of its atmospheric entry is defined by its inertia and by the gravitational and aerodynamic forces. The aerodynamic forces and accelerations associated with them change directly proportionally to the atmospheric density ρ and to the velocity squared of the device, V^2. The spacecraft first reaches the atmospheric layers of very low density. During further descent through the atmosphere, the density will rapidly increase, and the speed of the spacecraft begins to decrease. As we can see, the load factors which are acting upon the device first increase. However, at a certain point on the trajectory the decreased velocity of the spacecraft begins to prevail over the density increase. The result will be that the overload, after reaching a certain maximum, will begin to decrease.

The nature of change of the load factor as a function of time $n_\Sigma(t)$ is being observed in using any type of descent trajectory. In a general case, one might isolate three segments of the descent trajectories, in accordance with the nature of $n_\Sigma(t)$ change.

In using the aerodynamic braking of the spacecraft, in limiting cases, in other words, as the descent takes place along either the upper or lower entry corridor boundaries, we must base our conclusions on the threshold maximum load factors which operate at the entry corridor boundaries: on the upper entry corridor boundary - the capture load factor and at the lower boundary - the limiting permissible load factor for the construction design of a given spacecraft. The characteristic load factor relationship as a function of time is approximated by the following expression

$$n=at^2e^{-bt}$$

where a and b are the coefficients which are defined on the basis of maximum load factor and speed of the atmospheric entry.

In the case of ballistic descent and when the atmospheric density change is exponential, the maximum overload may be determined by using the following formula /93

$$n_{max}=-\frac{\theta \sin \Theta_{en}}{2eg} V_{en}^2$$

where θ is the density distribution factor as a function of altitude ($\rho=\rho_0 e^{-\nu h}$), Θ_{en} is the spacecraft entry angle into the atmosphere, V_{en} is the spacecraft speed at the point of entry (at the point of the descent trajectory intersection with the upper atmospheric boundary), e is the natural logarithm base and ρ_0 is the atmospheric density near the surface.

The maximum value of the load factor is reached at the velocity of $V=0.605\ V_{en}$.

In the case of automated spacecraft, the levels of load factors, if aerodynamic braking is used, may be several dozen (30-50 in the case of Mars) or several hundred units (in the case of Venus 300-400). The braking process itself lasts from several seconds to two or three minutes.

The flight speed and density of atmosphere during the braking stage is determined with sufficient degree of accuracy by the following expressions [15]:

$$V=V_{en}-ga[e^{bt}(\frac{t^2}{b}-\frac{2t}{b^2}+\frac{2}{b^3})-\frac{2}{b^3}]; \qquad (9.4)$$

$$\rho=\frac{2n}{\sigma_x V^2}; \qquad (9.5)$$

where

$$a=\frac{n_{max}}{t_b^2}e^2,\ b=-\frac{2}{t_b}; \qquad (9.6)$$

and t_b is the time of the load factor maximum. The time of the load factor maximum is a part of the total braking time t_Σ which depends on the velocity pulse V_Σ and on the maximum load factor n_{max}

$$t_\Sigma = \frac{V_\Sigma}{k_\ell n_{max}}, \tag{9.7}$$

where V_Σ is the magnitude of reduced speed from the moment of entry to the moment of flight in an established mode, k_ℓ is the factor which defines the mean load factor along the trajectory for the given atmospheric model, n_{max} is the maximum load factor, permissible for a given spacecraft.

The gradient of acceleration change during the spacecraft braking has the following form

$$C = \frac{n_{max}}{t_{en}} g. \tag{9.8}$$

It is known that during the descent on the planetary surface the initial entry velocity, which must be quenched aerodynamically, is approximately equal to the integral of the load factor

$$V_0 = \int_{t_0}^{t_k} g_0 n_x \, dt. \tag{9.9}$$

It can be easily shown that by assigning the entry velocity of /94
the descending spacecraft with specific parameters, as per expression (9.9), one can determine the two boundary relationships between the load factors and the descent time: the heaviest overload mode will correspond to the minimal descent time through the dense atmospheric layers, and vice versa, as t=max, the overload mode may be quite light.

Along the whole descent trajectory or on its separate segments, the following conditions must be satisfied which correspond to the constraints which are likely during the flight.

1. One of these conditions is the constraint in terms of the excess load

$$|n_y(V, H, c_y)| \leqslant n_{y\,max}, \tag{9.10}$$

$$\text{or } |n_x(V, H, c_x)| \leqslant n_{x\,max}, \tag{9.11}$$

$$\text{or } |n_\Sigma(V, H, c_x, c_y)| \leqslant n_{\Sigma\,max},$$

$$\text{where } n_\Sigma = \sqrt{n_x^2 + n_y^2}. \tag{9.12}$$

In loading the spacecraft, of great importance is the limiting pulse action and the measure of integral action of the overload. The limiting pulse action I_{lim} is the product of the overload n_{Σ} and the limiting time of action Δt_{lim}:

$$I_{lim}=n_{\Sigma}\Delta t_{lim}.$$

The major effect on the strength of the construction design is the magnitude of the total overload factor. The measure of integral action of the overload is characterized by the levels of overload along the whole trajectory:

$$M=\int_{t_1}^{t_2}(n_x^2+n_y^2)^{1/2}dt. \tag{9.13}$$

Consequently, in optimizing the descent trajectories, it is necessary to know the distribution functions of limiting overloads or load factors, the overload gradient, the pulse of its action and to be able to evaluate the excess of limiting overload action.

2. Not less important during the descent are the constraints in terms of the angles of attack, the bank angle and pitch angle:

$$\left.\begin{aligned}\alpha_{min}\leqslant\alpha\leqslant\alpha_{max},\\ \gamma_{min}\leqslant\gamma\leqslant\gamma_{max},\\ \vartheta_{min}\leqslant\vartheta\leqslant\vartheta_{max}.\end{aligned}\right\} \tag{9.14}$$

3. The constraints in terms of the shell temperature and the magnitude of thermal flows is one of the conditions for a successful spacecraft descent:

$$q(H, V, \alpha)\leqslant\bar{q}. \tag{9.15}$$

The load factors to which the spacecraft is exposed during the atmospheric entry is directly proportional to the mean molecular weight of the gaseous environment. As one can see, because of the chemical composition uncertainty, for example of Venus atmosphere, the actual overloads may differ by a factor of one and a half. If one is to add to this the unreliable information about the temperature of Venus stratosphere (the overload is inversely proportional to the temperature), it will turn out that in one case, /95
the spacecraft should employ in calculations the figure of 200 units, and in the other case - 450 units. In the case of such

scattering of initial data, in designing the spacecraft one must aim at the largest load factors. All this indicates how complex and multifaceted is the spacecraft descent process onto the planetary surface which has atmosphere.

The practical realization of spacecraft descent is possible only by solving a comprehensive set of complex problems in different domains of science and technology, namely, the ballistic-dynamic and the aerial gaseous dynamic problems, the problems of navigation and control, the energy-related provisions and provisions for the spacecraft functioning during a long time period, the selection of thermal protection systems and the selection of construction materials, etc. In addition the descent toward any planet within the Solar System is characterized by specific features which require in each case the development of a unique spacecraft for descent. For example, during the spacecraft descent to Venus, it must be able to withstand large load factors, high temperatures and pressures. During the descent toward Mars, the load factor and thermal operational modes will be light because of weak braking action of this planet's atmosphere. However, here we are faced with some other difficulties: the interplanetary station must have an autonomous navigation system to provide the precise spacecraft entry into the atmosphere, the device must have a sufficiently large frontal surface area, so as to quench its kinetic energy aerodynamically, otherwise one has to devise some complex and heavy systems for soft landing of spacecraft on Mars.

The spacecraft descent through the atmosphere of Jupiter is accompanied by huge, so far not precisely known, load factors and high temperatures and pressures as well as intense thermal flows, all of which is connected with the uncertainty of the atmospheric parameters here, and it is possible that the methods will have to be corrected in simulating the spacecraft descent on that planet.

Dynamics of Spacecraft Descent through the Planetary Atmosphere

In conducting the studies to elucidate the basic relationships in the motion of spacecraft, it is reasonable to utilize the system of differential equations which describe the planar motion of the spacecraft center of mass, as it descends through the planetary atmosphere [50]:

$$\left.\begin{aligned}
&\frac{dV}{dt} = -\sigma_x g_{0\,\mathrm{p}\ell}\frac{\rho V^2}{2} - g_{\mathrm{p}\ell}\sin\theta;\\
&\frac{d\theta}{dt} = \sigma_x k g_{0\,\mathrm{p}\ell}\frac{\rho V}{2} + \left(\frac{V}{r} - \frac{g_{\mathrm{p}\ell}}{V}\right)\cos\theta;\\
&\frac{dH}{dt} = V\sin\theta;\ \frac{dL}{dt} = V\frac{r_{\mathrm{p}\ell}}{r}\cos\theta;\\
&g_{\mathrm{p}\ell} = \frac{\mu}{r^2};\ n_x = \frac{c_x S\rho V^2}{2G} = \sigma_x q;\\
&n_y = \frac{c_y S\rho V^2}{2G} = k\sigma_x q;\\
&n_\Sigma = \sqrt{n_x^2 + n_y^2},
\end{aligned}\right\} \qquad (9.16)$$

where r is the current distance from the center of the planet to the spacecraft, $\sigma_X = \frac{c_X S}{G}$ is the ballistic parameter, L is the flight longitudinal range and μ is the product of the mass of the planet and gravitational constant. /96

Let us point out the assumptions which are used in deriving the equations of motion (9.16):

the spacecraft descent occurs when only the aerodynamic and gravitational forces are active;

the planet on the surface of which the descent is to take place is a sphere of $r_{p\ell}$ radius;

the planetary gravitational field is central.

The total trajectory of motion is to be traced out above a comparatively short surface area in a relatively short time and therefore, it is assumed that the planetary surface is "planar" and its rotation is disregarded;

the aerodynamic parameters and gravitational forces are constant;

the spacecraft motion, with respect to its center of mass is disregarded.

The system of differential equations, describing the spacecraft motion with respect to the center of mass, as it descends through the plantary atmosphere, with the projections on the axes of the conjugated coordinate system O_{XYZ} may be written in the following form

$$\left.\begin{aligned} \frac{d\omega_X}{dt} &= \frac{1}{I_X}[M_X - (I_Z - I_Y)\omega_Y\omega_Z]; \\ \frac{d\omega_Y}{dt} &= \frac{1}{I_Y}[M_Y - (I_X - I_Z)\omega_Z\omega_X]; \\ \frac{d\omega_Z}{dt} &= \frac{1}{I_Z}[M_Z - (I_Y - I_X)\omega_X\omega_Y], \end{aligned}\right\} \quad (9.17)$$

where ω_X, ω_Y, ω_Z are the projections of the vector of angular velocity for a given spacecraft on the axes of the conjugated coordinate system O_{XYZ}; M_X, M_Y, M_Z are the projections of the vector of the moment of resulting active forces (the restoring moment) on the axes of conjugated coordinate system O_{XYZ}; I_X, I_Y, I_Z are the major moments of inertia of the spacecraft with respect to the major axes of inertia and t is the current flight time.

The major axial directions of the conjugated coordinate system are: O_X - in the major plane (the velocity vector - radius vector plane), O_Y - in the vertical plane of symmetry of the spacecraft (upward) and O_Z supplements the system, making the right coordinate system.

In actual conditions, however, as the spacecraft descends through the planetary atmosphere, we have oscillatory motion of the spacecraft with respect to its center of mass, which so far has been disregarded.

To decrease the amplitude of such spacecraft oscillations, as it enters the planetary atmosphere, one is to use the damping device. The major components of this device are the damper which moves in the spacecraft plane and a shock absorber, located near the wall of the plane. The increase of the oscillatory amplitude is related to the friction forces, as the damper moves in the plane of the damping device and because of shock absorbing action of the walls. /97

To analyze the motion of the spacecraft it is necessary to compile a set of differential equations of the spacecraft motion, taking into account the movements of the damper in the plane of the damping device. The schematic of the spacecraft motion with a damper in the planetary coordinate system is shown in Figure 9.8.

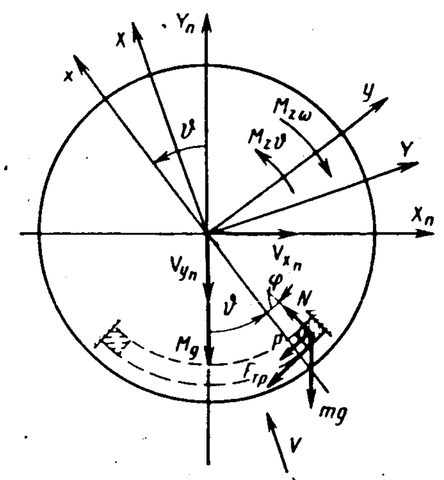

Figure 9.8. Schematic of motion of the spacecraft with damper.

The total kinetic energy of the spacecraft W consists of kinetic energy of the device without damper $W_{w.d}$ and the kinetic energy of the damper itself W_d [32]

The kinetic energy of the spacecraft without damper will have the following form

$$W_{wd} = \frac{MV_{\ell X}^2}{2} + \frac{MV_{\ell Y}^2}{2} + \frac{I\dot{\theta}^2}{2},$$

where M is the mass of the space craft, $V_{X\ell}$, $V_{Y\ell}$ are the axial components of the velocity of motion of the spacecraft centers of masses.

The kinetic energy of the damper will have the following form

$$W_d = \frac{mV^2_{X\ell}}{2} + \frac{mV^2_{Y\ell}}{2} + \frac{m[r(\dot{\vartheta}+\dot{\varphi})]^2}{2} =$$

$$= \frac{m[V_{X\ell} + r(\dot{\vartheta}+\dot{\varphi})\cos(\vartheta+\varphi)]^2 + m[V_{Y\ell} + r(\dot{\vartheta}+\dot{\varphi})\ \sin(\vartheta+\varphi)]^2}{2},$$

where m is the mass of the damper.

The total kinetic energy of the spacecraft, having $V_{X\ell}=\dot{X}$ and $\dot{Y}=V_{Y\ell}$ may be represented in the following form

$$W = \frac{M}{2}(V^2_{X_\ell} + V^2_{Y_\ell}) + \frac{I\dot{\vartheta}^2}{2} + \frac{m}{2}\Big\{[V_{X_\ell} + r(\dot{\vartheta}+\dot{\varphi}) \times$$
$$\times \cos(\vartheta+\varphi)]^2 + [V_{Y_\ell} - r(\dot{\vartheta}+\dot{\varphi})\sin(\vartheta+\varphi)]^2\Big\} =$$
$$= \frac{M}{2}(\dot{X}^2 + \dot{Y}^2) + \frac{I\dot{\vartheta}^2}{2} + \frac{m}{2}\Big\{\dot{X}^2 + 2\dot{X}r(\dot{\vartheta}+\dot{\varphi}) \times$$
$$\times \cos(\vartheta+\varphi) + [r(\dot{\vartheta}+\dot{\varphi})\cos(\vartheta+\varphi)]^2 + \dot{Y}^2 -$$
$$- 2\dot{Y}r(\dot{\vartheta}+\dot{\varphi})\sin(\vartheta+\varphi) + [r(\dot{\vartheta}+\dot{\varphi})\sin(\vartheta+\varphi)]^2\Big\}.$$

The operation of the spacecraft and damper, with respect to possible displacement, may be represented in the following form

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$$\delta A_X = (-X_T\sin\vartheta + Y_n\cos\vartheta)\delta X;$$
$$\delta A_Y = [-X_T\cos\vartheta - Y_n\sin\vartheta + (M+m)g_\ell]\delta Y;$$
$$\delta A_\vartheta = [M_{Z\vartheta} - M_{Z\omega} - mg_\ell\sin(\vartheta+\varphi)]\delta\vartheta;$$
$$\delta A_\varphi = r[-mg_\ell\sin(\vartheta+\varphi) - F_{fr} - P]\delta\varphi.$$

Let us utilize the Lagrange equation of second kind in its general form

$$\frac{d}{dt}\left(\frac{\partial W}{\partial \dot{q}_s}\right) - \frac{\partial W}{\partial q_s} = Q_s,$$

where s is the parameter which may acquire the values of 1,2,3,4:

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$$q_1 = X;\ q_2 = Y;\ q_3 = \vartheta;\ q_4 = \varphi.$$

We will obtain the following system of ordinary differential equations

$$\left.\begin{aligned}
&(M+m)\ddot{X} + m[r(\ddot{\vartheta}+\ddot{\varphi})\cos(\vartheta+\varphi) - r(\dot{\vartheta}+\dot{\varphi})^2 \sin(\vartheta+\varphi)] = Q_X;\\
&(M+m)\ddot{Y} + m[-r(\ddot{\vartheta}+\ddot{\varphi})\sin(\vartheta+\varphi) - r(\dot{\vartheta}+\dot{\varphi})^2\cos(\vartheta+\varphi)] = Q_Y;\\
&I\ddot{\vartheta} + m[r^2(\ddot{\vartheta}+\ddot{\varphi}) + r\ddot{X}\cos(\vartheta+\varphi) - r(\ddot{Y}\sin(\vartheta+\varphi)] = Q_\vartheta;\\
&m[r^2(\ddot{\vartheta}+\ddot{\varphi}) + r\ddot{X}\cos(\vartheta+\varphi) - r\ddot{Y}\sin(\vartheta+\varphi)] = Q_\varphi,
\end{aligned}\right\}$$

where $Q_X = - X_T\sin\vartheta + Y\ \cos\varphi;\ Q_Y = - X_T\cos\vartheta - Y\ \sin\vartheta + (M+m)g;$

$Q_\vartheta = M_{Z\vartheta} - M_{Z\omega} + mg\sin(\vartheta+\varphi);$

$Q_\varphi = - mg\sin(\vartheta+\varphi) - F_{fr} - P.$

The equations may be written in somewhat different form, leaving in the left hand side of these equations only the components containing $\ddot{X}$, $\ddot{Y}$, $\ddot{\theta}$ $\ddot{\phi}$:

$$\left.\begin{aligned}
&\ddot{X} = \frac{1}{M}[-X_T\sin\vartheta + Y_\ell\cos\vartheta + (F_{fr}+P)\cos(\vartheta+\varphi) - N\sin(\vartheta+\varphi)];\\
&\ddot{Y} = \frac{1}{M}[-X_T\cos\vartheta - Y_\ell\sin\vartheta + Mg - (F_{fr}+P)\sin(\vartheta+\varphi) - N\cos(\varphi+\vartheta)];\\
&\ddot{\vartheta} = \frac{1}{I}[M_{Z\vartheta} - M_{Z\omega} + (F_{fr}+P)r];\\
&\ddot{\varphi} = -\frac{1}{r}[(\frac{1}{M} + \frac{1}{m})(F_{fr}+P) + \frac{1}{M}(X_T\sin\varphi + Y_\ell\cos\varphi) - \ddot{\vartheta}],
\end{aligned}\right\} \qquad (9.18)$$

where $N = -\frac{Mm}{M+m}[r(\dot{\vartheta}+\dot{\varphi})^2 +$

$+ \frac{1}{M}(X_T\cos\varphi - Y_\ell\sin\varphi)].$

In accordance with the mathematical model, a number of parameters which are incorporated in the system (9.18) may be obtained only experimentally.

One is to utilize the hemispherical impact damper with interchangeable cores and a hemispherical panel with interior ribs as the experimental prototypes. /99

It should be noted that the shape of the panel surfaces and of the damper, which are being investigated, and the processing of surfaces, result in the scattering of the coefficient of friction in the ranges which are presented below.

Two metals exposed to friction	Range of coefficient of friction
Steel-aluminum	0.15-0.22
Bronze-aluminum	0.3-0.60
Cast iron-aluminum	0.5-0.80
Titanium-aluminum	0.4-0.85

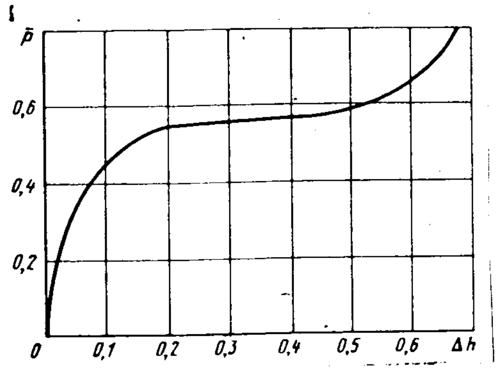

Figure 9.9. Statically applied load as a function of the damper deformation $\overline{P}=f(\Delta\overline{h})$.

Figure 9.9 shows the statically applied load as a function of the impact damper deformation $\overline{P}=f(\Delta\overline{h})$.

Parachute Descent of the Space Model Through Dense Planetary Atmosphere

The schematic diagram of the spacecraft Venera-13 and Venera-14 included the distance which the descending spacecraft traversed by using a parachute and which lasted about 10 min. The parachute system is designed to ensure the required duration of the spacecraft movement through the cloud cover of the Venus atmosphere and for the separation of the thermal shield components of the shell.

To handle these two problems and in order to minimize the mass of the parachute system, the two-stage system was chosen which incorporates a parachute used in the course of detachment and the braking parachute. The spacecraft cyclogram during the parachute descent and some parameters of the parachute systems are presented in the study [3].

The spacecraft motion around its center of mass is assumed to be considered as a summary motion with respect to the spacecraft longitudinal axis OX_{SC} and the latitudinal axes OY_{SC} and OZ_{SC}. The requirements which are imposed on the spacecraft motion around the center of masses during descent, by using the braking parachute system, may be represented in the following form:

$$|\omega_{X_{SC}}(t=t_{\text{act braking parachute}})| \leq 1\ s^{-1}, \qquad (9.19)$$

$$\alpha_{SC}\ (t) \leq 15^{\circ}, \qquad (9.20)$$

where ω_{XSC} is the component of the vector of angular velocity along the spacecraft OX_{SC} axis and $\alpha_{SC}(t)$ is the current spatial angle of attack.

The first constraint defines the angular velocity ω_{XSC} at the end of the parachute descent. It is necessary to exclude a sharp increase of the angle of attack, with the appearance of autorotation at the beginning of the spacecraft descent by using the braking flap. At the moment when the parachute system is activated, the ω_{XSC} may reach 2 s^{-1}. The second constraint is the requirement of a stable communication between the spacecraft and the orbital devices and the contact with some scientific instruments in terms of the "angle of vision." The maximum spatial angle of attack at the moment when the parachute system is activated is equal to 25°. The parachute which is designed for the removal of upper hemisphere of the thermal shielding and for the lowering of the spacecraft speed to the levels which are necessary before the braking parachute is activated, has no noticeable effect on the dynamics of the spacecraft motion around the center of mass. Therefore, the initial conditions of the spacecraft angular motion at the moment when the parachute system is activated may be extended to the moment when the braking parachute system is activated. /100

In order to satisfy the (9.19) constraints, one selects the multipoint attachment of the braking parachute system. The braking parachute system attachment to the spacecraft is accomplished by using four independent links, thus significantly increasing the resistance moment to the twist of the parachute cords, enabling the parachute to retard the spacecraft motion with respect to its longitudinal axis. The complete damping takes place in the case of the aerodynamic symmetry of the parachute canopy, when the moment factor of the bank angle $m_{x_0}=0$. In the actual conditions, the aerodynamic symmetry of the canopy, for a number of reasons, cannot be accomplished. The estimation of m_{x_0} and also of the coefficient of damping $m_x^{\bar{w}_x}$ has been conducted by measuring the angular velocity ω_{XSC} by activating the mockup spacecraft experimentally, under the conditions which reproduced the parachute descent through the atmosphere of Venus.

Of great interest from the point of view of adhering to the constraints (9.20) is to investigate the perturbation motion of the spacecraft during the parachute descent around the lateral axes OY_{SC} and OZ_{SC}. To construct and calibrate the spacecraft model motion around the center of mass during the descent parachute action, in parallel with the experimental data by using the aerodynamic tunnels, the results of prototype system spacecraft-parachute have also been used. It was noticed in actual tests that although during the mutual oscillations of the spacecraft and of the parachute, having the multipoint attachment of the latter to the spacecraft, the shape of the parachute may significantly change while the canopy itself, for all practical purposes, will not deform. A physical model was proposed in which the actual system spacecraft-parachute had been replaced by a system of two solid bodies - the spacecraft and the axially-symmetric canopy, connected by means of four independent and elastic devices, which are not compression-activated and which incorporate the mechanism for internal friction, proportional to the deformation rate (Figure 9.10). The tension in each of four links is determined by using the following formulas

$$T_i = \begin{cases} K_i \Delta l_i + \eta_i \Delta \dot{l}_i \text{ when } \Delta l_i > 0 \\ \qquad\qquad (i = 1, 2, 3, 4), \\ 0 \quad \text{when } \Delta l_i \leqslant 0 \end{cases} \tag{9.21}$$

where $\Delta \ell_j$, $\Delta \dot{\ell}_i$ are the deformation and deformation rate of the links exposed to the load, K_i is the equivalent coefficient of the link elasticity which takes into account the elasticity of the cords and of the link itself and η_i is the coefficient which characterizes the internal friction within the link.

The point O which is at the intersection between the parachute logitudinal axis and the plane of lower rim has been selected as a characteristic point, on the basis of which the canopy speed with respect to the flow has been calculated, since it has been experimentally established that this is precisely the central point of the parachute pressure and its position is practically independent of the parachute angle of attack.

By taking into account the aerodynamic action of the flow on the spacecraft [25] the mathematical model which corresponds to such physical model will have the following form: /101

$$\left.\begin{aligned} & m_{SC}(\dot{\vec{V}}_{SC} + \vec{\omega}_{SC} \vec{V}_{SC}) = -m_{SC}\vec{g} + \vec{R}_{SC} - \vec{T}; \\ & \theta_{SC} \dot{\vec{\omega}}_{SC} + \vec{\omega}_{SC} \theta_{SC} \vec{\omega}_{SC} = \vec{M}_{SC} - \vec{M}^{T}; \\ & m_{K}[\dot{\vec{V}}_{K} + \vec{\omega}_{K} \vec{V}_{K} + \dot{\vec{\omega}} \vec{\rho}_{K} + \vec{\omega}_{K} (\vec{\rho}_{K} \vec{\omega}_{K})] = -m_{K}\vec{g} + \\ & + \vec{R}^{A} + \vec{R}^{I} + \vec{T}; \\ & \theta_{K} \dot{\vec{\omega}}_{K} + \vec{\omega}_{K} \theta_{K} \vec{\omega}_{K} + m_{K} \vec{\rho}_{K} (\dot{\vec{V}}_{K} + \vec{\omega}_{K} \vec{V}_{K}) = \\ & = \vec{M}^{A} + M^{I} - \vec{\rho}_{K} m_{K} \vec{g} + \vec{M}^{T}, \end{aligned}\right\} \tag{9.22}$$

Key to formula (9.22): k is equivalent to c.

where m_{sc}. θ_{sc} are the spacecraft mass and tensor of inertia at its center of mass O_{SC}; m_c, θ_c are the cupola mass and tensor of

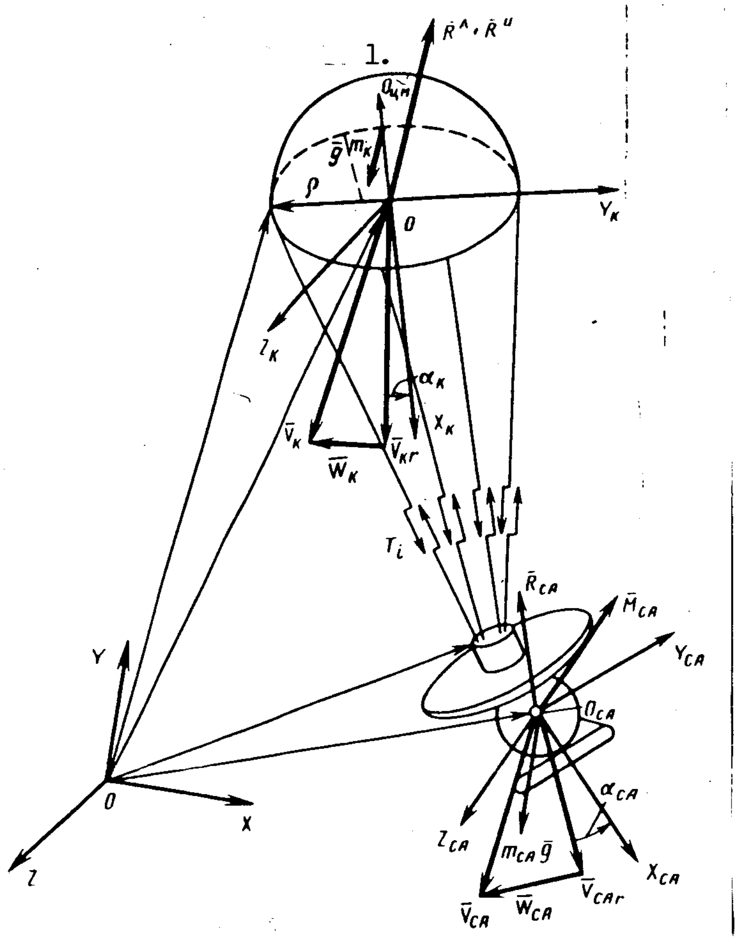

Figure 9.10. Model of the spacecraft (SC) - parachute system in the form of two solid bodies.

Key: 1. O_{cm}. CA is equivalent to SC which stands for spacecraft., k is equivalent to c.

inertia at its center of pressure O; $\vec{\rho}_c$ is the radius vector of the canopy center of mass O_{cm} within the conjugated coordinate system OX_c, Y_c, Z_c, $\vec{V}_{SC}$, $\vec{V}_c$ are the vectors of linear velocities of the spacecraft and of the canopy, $\vec{\omega}_{SC}$, $\vec{\omega}_c$ are the vectors of angular velocities of the spacecraft and cupola, $\vec{R}_{SC}$, $\vec{M}_{SC}$ are the major vector and moment of aerodynamic forces, acting upon the spacecraft, $\vec{R}^A$, $\vec{M}^A$ are the major vector and moment of quasistationary and aerodynamic forces, acting on the parachute canopy, $\vec{R}^I$, $\vec{M}^I$ are the major vector and moment of aerodynamic forces, acting upon /102 the parachute canopy, which is due to the destabilized motion, $\vec{T}$, $\vec{M}^T$ are the force and moment due to the elastic cords and $\vec{g}$ is the free fall acceleration vector.

In addition to the equations (9.22) one must have available the kinematic equations which describe the advance motion of the spacecraft and of the parachute

$$\dot{\vec{r}}_{SC}=\vec{V}_{SC},\ \dot{\vec{r}}_c=\vec{V}_c. \tag{9.23}$$

The kinematic equations of spacecraft and parachute rotational motions are written for the Rodrig- Hamilton parameters λ_i (for i=0, 1, 2, 3), which characterize their angular positions

$$\begin{bmatrix} \dot{\lambda}_0 \\ \dot{\lambda}_1 \\ \dot{\lambda}_2 \\ \dot{\lambda}_3 \end{bmatrix} = \frac{1}{2} \begin{bmatrix} \lambda_0 & -\lambda_1 & -\lambda_2 & -\lambda_3 \\ \lambda_1 & \lambda_0 & -\lambda_3 & \lambda_2 \\ \lambda_2 & \lambda_3 & \lambda_0 & -\lambda_1 \\ \lambda_3 & -\lambda_2 & \lambda_1 & \lambda_0 \end{bmatrix} \begin{bmatrix} 0 \\ \omega_x \\ \omega_y \\ \omega_z \end{bmatrix}, \tag{9.24}$$

where ω_x, ω_y, ω_z are the angular velocities of the spacecraft and of the canopy.

In order to be able to close the system of equations, it is necessary to add to the system of equations (9.22)-(9.24) four equations of joint deformations (9.21). In this case, the length of the i link may be calculated if one knows the coordinates of the attachment points of this link to the spacecraft $\vec{r}_{SCi}=\vec{r}_{SC}+\vec{\rho}_{SCi}$ and to the parachute $\vec{r}_{ci}=\vec{r}_c+\vec{\rho}_{ci}$, where $\vec{\rho}_{SCi}$ and $\vec{\rho}_{ci}$ are the radius are the radius vectors of the attachment points of the i link to the spacecraft and to the canopy.

The mathematical model (9.21)-(9.24) makes it possible to conduct the numerical analysis of the spacecraft-parachute system dynamics, taking into account the inertia of the parachute, the atmospheric turbulence, the methods by which the parachute is attached to the spacecraft and the elasticity and damping effects of the suspended system. The mathematical model, both qualitatively and quantitatively, reflects sufficiently well the actual process. A similar type of oscillations will be found as the spacecraft descends by using the braking flap.

The atmospheric turbulence affects quite considerably the dynamics of the spacecraft and braking parachute oscillations.

By calculations, it was possible to determine that the amplitude of the angular velocity with respect to the lateral axes of the spacecraft within the Venus turbulent atmosphere, increases to 60 degrees/s, as compared to the 25 degrees/s in quiet atmosphere, which corresponds to the increase in the auto-oscillation amplitudes for the spacecraft spatial angle of attack between 5 and 15°.

Although in the quiet atmosphere the computed amplitude of the spacecraft auto-oscillations, if the parachute is used, is less than in the case of the braking flap, in the case of the turbulent atmosphere, the spacecraft-parachute system is more sensitive to the wind action which results in larger spatial angles of attack than if /103
one is to use for descent the braking flaps.

CHAPTER 10.

MATHEMATICAL MODELING OF THE DYNAMICS OF SPACECRAFT SOFT LANDING AND ITS FUNCTIONING ON THE PLANETARY SURFACE

Let us consider now a more important and conclusive stage of spacecraft flight (beginning at the height between 30-10 m above the surface of the planet) which includes the touchdown of it, its skidding action and finally, its complete rest on the surface in the position which is the starting position for the subsequent functioning of the spacecraft on a given celestial body. /103

The spacecraft touchdown on the planet, in addition to the "soft landing" must satisfy the following requirements:

mandatory retention of the stable position of the device during the whole landing process;

exclusion of the possible bouncing of the device, immediately after the first contact with the soil;

position of the spacecraft on the planetary surface, after landing must ensure the normal operation of all its systems (and if this is provided by the program, it must also create the favorable conditions for the subsequent take-off);

reduction to the minimum of the danger of different emergency situations during landing by improving the reliability of all functional systems on the device.

The analysis of landing conditions on the planetary surface shows that the speed of contact between the device and soil must be in the range from several meters per second to a dozen and more meters per second. It follows from this that the spacecraft must be equipped with a reliable landing gear and a special shock-absorbing landing gear. One is familiar with different schematic designs of the landing gear, as for example, the leaf-type, the shock-absorbing rod supports, the truss design or the thin-walled shells.

It is clear that the development of a reliably operating landing gear in the interplanetary space device has required a whole series of comprehensive studies, including the theoretical, experimental and construction design work.

The space landing module (SLM), because of the elastic connections between its separate design components (the payload, scientific equipment, the instrumental and propulsion compartments and various supports) represent a rather complex elastic system of interacting bodies.

The mathematical description of the dynamics of landing on a planet in the presence of such complex system, in a general case of spatial motion and the direct contact between the module supports and soil, result in an extremely bulky system of differential equations including the unknown quantities of the coefficients of elastic connections between the SLM separate components, which may be determined only experimentally, by using a prototype. Therefore, to conduct the calculations on the dynamics of the SLM landing on the planetary surface by using such mathematical model, because of the extensive time required, is possible only by verifying steps, selecting some critical landing cases for the landing device which has already been designed. /104

The comprehensive studies of the SLM soft landing dynamics at this stage of the sketch design may be conducted by using somewhat simpler, and not very time-consuming, methods of engineering studies, making it possible to evaluate different design schematics of the landing gear in the module and to select the optimal parameters.

We shall present below a general case of SLM spatial motion, with the device of rigid construction and incorporating the landing support gear, consisting of a system of rods, forming an inversely suspended tripod (Figure 10.1). The central rod of the tripod is equipped with the energy-absorbing arrangement and may be deformed. The complex motion of the SLM during soft landing on the planetary surface may be represented as two somewhat simpler motions:

the spatial motion of the SLM body;

the motion of bases of the landing supports (which have a specific reduced mass m_i) along the surface of the soil in any direction, but with the limitation which is imposed by the surface of the landing site itself.

The spatial motion of the SLM rigid body takes place as a result of the perturbation forces and moments which are in turn the result of interaction between the landing gear and the soil. In this case, the inclination angle of the landing site and the coefficient of friction in the supports as they touch the surface, may vary within a broad range.

In order to be able to describe such SLM spatial motion during soft landing, it is necessary to compile and solve the system consisting of 6+2N dynamic equations, each of which is a nonlinear differential equation of the second order (N is the number of supports in the landing gear). The complexity of solving such equations requires the use of numerical methods and utilization of a computer.

In order to reduce these equations to the form of ordinary linear differential equations, the following assumptions are introduced: all forces, acting upon the system, are assumed to be constant during a sufficiently small integration step of differential equations.

The dynamics of the SLM soft landing is defined primarily by the following factors:

the design specifics of the device and the specific properties of its energy-absorbing gear;

the inclination of the planetary surface at the landing site;

the SLM orientation during the contact of one of its supports (the moment of the physical contact between the supports and surface may be conviently used as a start of the count-off);

the vertical and horizontal components of the SLM vector of velocity at the moment of touchdown; /105

the components of the spacecraft vector of angular velocity at the moment of touchdown;

the external forces, acting upon the spacecraft during touchdown;

the prameters of the soil.

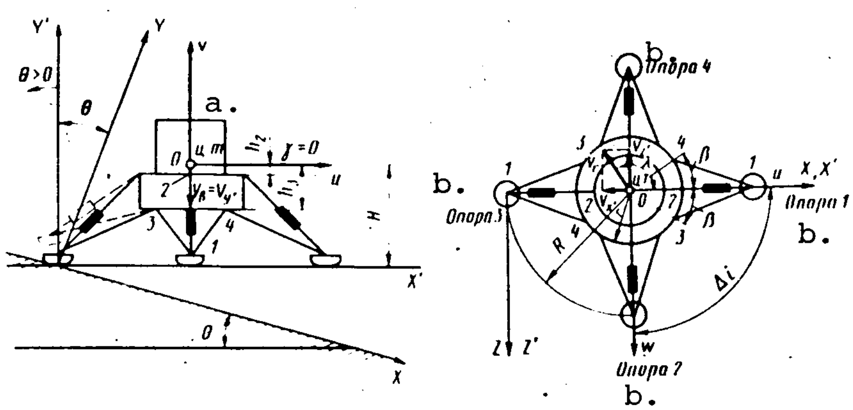

Figure 10.1 Geometric shape of the spacecraft.

Key: a. cg; b. Support

The geometric shape of the SLM computed model for any moment in time during landing (from the moment of the very first contact of any of the SLM supports with the soil, to its complete rest) may be characterized on the whole by the coordinates of several control points for these moments in time.

Among such control points are the SLM center of mass (the point O) and the four points at each of its supports 1i-4i (see Figure 10.1).

During the landing process, the coordinates of control points 2i-4i in the moving coordinate system (u, v, w) remain constant and the coordinates of 1i points (i is the number of supports) will be variable because of the geometry change in the landing gear.

The coordinates of 1i points, after the moment of the touch-down by the bases of support, are determined by solving the equations of support motion.

When the coordinates of all control points are unknown, one can calculate the length of the deforming design components, as per the following formula

$$l_{1,2,i}=[(u_{1i}-u_{2i})^2+(v_{1i}-v_{2i})^2+(w_{1i}-w_{2i})^2]^{1/2}. \tag{10.1}$$

The mass-inertia parameters of SLM (m, i_u, i_v, i_w, m_i et al.) are assumed to be given, since they are determined during the construction design work and in the course of mass and centering calculations.

To absorb the energy during the SLM soft landing, the design components of the landing gear incorporate special energy-absorbing devices. These devices absorb the major part of the available kinetic energy of the SLM. The other, and much smaller part of the energy is absorbed by friction in the hinges, friction against the planetary surface and also by the internal friction between the design components. The energy absorbers of the mathematical SLM model may have the computational parameters of the "force-deformation" of any type but in the future, in order to simplify the mathematical /106
representation, we shall consider only the energy absorbers in which the "force-deformation" parameters are in the form of the step function [15]. During the repeat load exposure of the energy absorbers in the course of landing, it is necessary to know the magnitude of residual deformation from the preceding load exposure.

In studying the dynamics of the spatial motion of the SLM, three coordinate systems are used (see Figure 10.1):

the moving coordinate system (uvw) rigidly connected to the device (the major central axes of the SLM inertia);

the non-moving coordinate system (XYZ) connected to the planetary surface (the X and Z axes are in the plane of the landing surface and the X axis coincides with the direction of the maximum inclination of the surface and the Y axis - with the direction of the local normal);

the non-moveable coordinate system (X'Y'Z') associated with the planetary surface (the X' and Z' axes are in the plane of the local horizon and Z' axis coincides with Z axis and Y' - coincides with the local vertical).

The specefic forces are being developed within the landing gear of the spacecraft, as it lands on the planetary surface. In a general case, the magnitudes of these forces depend on the parameters of deforming components which are used within the landing gear of the device. The external forces are applied at the moment of the touchdown and physical contact between the base of support and the planetary surface and the direction of the force is opposite to the direction of the velocities of the corresponding support bases. The external forces are determined for each integration step (Δt) depending on the conditions in which, at this moment in time, the dynamic system finds itself.

At the base of each SLM support (li) as the module moves along the planetary soil, one is involved with the active force of the SLM body (through the components of the landing gear), the reaction of the soil and the force of friction of the support base along the surface.

The force interaction between the landing gear and planetary surface depends on the relief of the surface, on the structural and mechanical properties of the soil at the landing site. To conduct the theoretical studies of the dynamics of the SLM soft landing one is to select the computational model of the surface which has the parameters not contradicting the present-day information about the planetary soil.

In its first approximation, one may utilize as a model of such landing surface an absolute plane which has a specific inclination angle and **high,** reduced coefficient of friction which enables us to account for the resistance to the movement of the support along the actual planetary surface.

In a general case, the movement of the body of the SLM, which is viewed as a solid body, may be described by Euler equations. Let us write these equations as projections on the coordinate axes of the moving system (uvw) (Figure 10.1).

We will obtain the following system of differential equations:

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$$\left.\begin{aligned} \frac{dV_u}{dt} &= \frac{F_u}{m} - g_{\ell} \sin\vartheta \cos\varphi + V_v\omega_w - V_w\omega_v; \\ \frac{dV_v}{dt} &= \frac{F_v}{m} - g_{\ell} \cos\vartheta + V_w\omega_u - V_u\omega_w; \\ \frac{dV_w}{dt} &= \frac{F_w}{m} - g_{\ell} \sin\vartheta \sin\varphi + V_u\omega_v - V_v\omega_u; \end{aligned}\right\} \qquad (10.2)$$

$$\left.\begin{aligned}\frac{d\omega_u}{dt} &= \frac{M_u}{I_u} + \omega_v\omega_w\left(\frac{I_v - I_w}{I_u}\right);\\ \frac{d\omega_v}{dt} &= \frac{M_v}{I_v} + \omega_u\omega_w\left(\frac{I_w - I_u}{I_v}\right);\\ \frac{d\omega_w}{dt} &= \frac{M_w}{I_w} + \omega_u\omega_v\left(\frac{I_u - I_v}{I_w}\right);\end{aligned}\right\} \quad (10.3)$$

$$\left.\begin{aligned}\frac{d\vartheta}{dt} &= \omega_w \cos\varphi - \omega_u \sin\varphi;\\ \frac{d\varphi}{dt} &= \omega_v - \omega_w \sin\varphi \operatorname{ctg}\vartheta - \omega_u \cos\varphi \operatorname{ctg}\vartheta;\\ \frac{d\psi}{dt} &= \omega_w \sin\varphi \operatorname{cosec}\vartheta + \omega_u \operatorname{cosec}\vartheta \cos\varphi.\end{aligned}\right\} \quad (10.4)$$

The equation (10.2) describes the motion of the SLM center of mass, the equations (10.3) describes the motion around the center of masses. The expressions (10.4) are the kinematic Euler relationships.

Since one cannot assign absolute position and orientation of the module with reference to the moving coordinate system (uvw) the remaining equations may be written in the non-moving coordinate system (X'Y'Z'):

$$\frac{dX'}{dt} = V_{X'};\ \frac{dY'}{dt} = V_{Y'};\ \frac{dZ'}{dt} = V_{Z'}. \quad (10.5)$$

As a result of integration of the equations (10.2) one can determine the projected velocities of the SLM center of mass on the moving coordinate axes (uvw).

The components of velocity of the centers of mass within the (X'Y'Z') system are determined by using the following formulas

$$\left.\begin{aligned}&V_{X'} = A_1 V_u + A_2 V_v + A_3 V_w;\ V_{Y'} = B_1 V_u +\\ &+ B_2 V_v + B_3 V_w;\ V_{Z'} = C_1 V_u + C_2 V_v + C_3 V_w,\end{aligned}\right\} \quad (10.6)$$

where

$$\left.\begin{aligned}&A_1 = \cos\psi\cos\vartheta\cos\varphi - \sin\psi\sin\varphi;\ A_2 = -\cos\psi\sin\vartheta;\\ &A_3 = \sin\psi\cos\varphi + \cos\psi\cos\vartheta\sin\varphi;\ B_1 = \cos\varphi\sin\vartheta;\ B_2 = \cos\vartheta;\\ &B_3 = \sin\vartheta\sin\varphi;\ C_1 = -\cos\psi\sin\varphi - \sin\psi\cos\vartheta\cos\varphi;\\ &C_2 = \sin\psi\sin\vartheta;\ C_3 = \cos\varphi\cos\psi - \sin\varphi\cos\vartheta\sin\psi.\end{aligned}\right\} \quad (10.7)$$

Now, by using the formulas (10.5) we can calculate the coordinates of the SLM center of masses in the non-moving coordinate system (X'Y'Z').

As one can see, by solving a system of equations (10.2)-(10.7) one can determine all major parameters of the SLM body motion: /108

the components of velocity and center of masses within the fixed or non-moving system:

$V_{X'}$, $V_{Y'}$, and $V_{Z'}$;

the coordinates of centers of masses within the non-moving or fixed system:

X', Y' and Z';

the components of instantaneous angular velocity of rotation with respect to the center of masses: ω_u, ω_v and ω_w;

the angular coordinates (Euler angles) which characterize the position of the body with respect to the system which moves in such a way that the beginning of its coordinates always coincides with the center of masses and the axes remain parallel to the corresponding axes of the non-moving coordinate system

(X', Y', Z')-θ, ϕ and ψ.

In order to be able to solve the system of equations (10.2)-(10.7) it is necessary to assign at the moment of time t=0, the values of the following parameters:

$$\begin{aligned}
&X' = X'_0; \quad V_{x'} = V_{x'_0}; \quad \vartheta = \vartheta_0; \quad \omega_u = \omega_{u_0};\\
&Y' = Y'_0; \quad V_{y'} = V_{y'_0}; \quad \varphi = \varphi_0; \quad \omega_v = \omega_{v_0};\\
&Z' = Z'_0; \quad V_{z'} = V_{z'_0}; \quad \psi = \psi_0; \quad \omega_w = \omega_{w_0}.
\end{aligned}$$

If at t=0 $\theta_0=\phi_0=\psi_0=0$, then we will obtain from the equations (10.6) and (10.7) the following $V_{x'0}=V_{u0}$; $V_{y'0}=V_{v0}$; $V_{z'0}=V_{w0}$, in other words, in calculating the first parts in (10.2) one can immediately use the initial data.

If however at t=0 $\theta_0\neq 0$, $\phi_0\neq 0$, $\psi_0\neq 0$ (or if any of the angles is equal to zero) then the values of V_u, V_v and V_w are determined from the equations (10.6) by using the initial values of the Euler angles θ_0, ϕ_0 and ψ_0.

The forces acting upon the dynamic system (body of the device) during landing depend on the character of the planetary surface at the landing site (the inclination angle of the site, the physical and mechanical properties of the soil, the structure of the surface layer, etc.), on the design specifics of SLM (the number of supports, position of the centers of masses, parameters of the energy-absorbing attachments, etc.) on the specifics of the landing support motion during the soft touchdown (the kinematic parameters of elementary motions, comprising the complex motion of the module). The soft landing of the device occurs at the speeds of V=5 ... 30 m/s on the planetary surfaces with the carrying capacity of $\sigma_s=0.2\cdot10^5 \ldots 1.5\cdot10^7$ Pa, and is characterized by high peak impact load factors (for example, in the case of automated space devices, the load factors may reach $n_\Sigma=50 \ldots 200$ units, with the duration of load exposure t=0.01 ... 0.5 s.

In its vector description, the load factor at the module center of mass, as the module makes contact with the surface and using the conjugated coordinate system (uvw) may be written in the following form

$$\bar{n}=\frac{1}{g}M_1(\bar{A}_{p\ell}-g_{p\ell}), \tag{10.8}$$

where M_1 is the transition matrix from the planetary coordinate system /109
to the conjugated one, $\bar{A}$ is the vector of absolute acceleration of the module center of masses in the planetary coordinate system:

$$M_1=\begin{vmatrix} \cos\vartheta\cdot\cos\psi; & \sin\vartheta; & -\cos\vartheta\cdot\sin\psi \\ \sin\varphi\cdot\sin\psi-\cos\varphi\cdot\sin\vartheta\cdot\cos\psi; & \cos\varphi\cdot\cos\vartheta; & \cos\varphi\cdot\sin\vartheta\cdot\sin\psi+\sin\varphi\cdot\cos\psi \\ \sin\varphi\cdot\sin\vartheta\cdot\cos\psi+\cos\varphi\cdot\sin\psi; & -\sin\varphi\cdot\cos\vartheta; & -\sin\varphi\cdot\sin\vartheta\cdot\sin\psi+\cos\varphi\cdot\cos\psi \end{vmatrix},$$

where ϕ, θ and ψ are the yaw angle, pitch angle and bank angle of the module, respectively.

The load factor vector at the LSM point which is being investigated is determined by using the following formula

$$\bar{n}_i=\frac{1}{g}\left\{[M_1\bar{A}_{p\ell}+(\dot{\bar{\omega}}(\overline{O-i}))+\bar{\omega}_u(\bar{\omega}_u+(\overline{O-i}))]-M_1\bar{g}_{p\ell}\right\}, \tag{10.9}$$

where $\bar{\omega}$ is the vector of angular velocity of the LSM rotation, (O-i) is the radius vector of the SLM point which is being investigated in the conjugated coordinate system.

The radius vector $(\overline{O-i})$ is assigned by the following expression

$(\overline{O-i})=u_i\bar{i}+v_i\bar{j}+w_i\bar{k}$, where u_i, v_i, w_i are the vector projections on the axes of the conjugated coordinate system.

Because almost all major factors change during landing, the force action on the module will also change. This complicates the whole problem. However, it can be handled by using a computer and utilizing the numerical integration if one assumes that the force action on the SLM remains constant during a sufficiently small integration step.

In such case, the equations (10.2)-(10.5) must be represented in the finite-difference form:

$$\left.\begin{aligned}
\frac{\Delta V_u}{\Delta t} &= \frac{F_u}{m} - g_\ell \sin\vartheta\cos\varphi + V_v\omega_w - V_w\omega_v;\\
\frac{\Delta V_v}{\Delta t} &= \frac{F_v}{m} - g_\ell \cos\vartheta + V_w\omega_u - V_u\omega_w;\\
\frac{\Delta V_w}{\Delta t} &= \frac{F_w}{m} - g_\ell \sin\vartheta\sin\varphi + V_u\omega_v - V_v\omega_u;\\
\frac{\Delta \omega_v}{\Delta t} &= \frac{M_u}{I_u} + \omega_v\omega_w\left(\frac{I_v-I_w}{I_u}\right);\\
\frac{\Delta \omega_v}{\Delta t} &= \frac{M_v}{I_v} + \omega_u\omega_w\left(\frac{I_w-I_u}{I_v}\right);\\
\frac{\Delta \omega_w}{\Delta t} &= \frac{M_w}{I_w} + \omega_u\omega_v\left(\frac{I_u-I_v}{I_w}\right);
\end{aligned}\right\} \quad (10.10)$$

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$$\left.\begin{aligned}
\frac{\Delta\vartheta}{\Delta t} &= \omega_w\cos\varphi - \omega_u\sin\varphi;\\
\frac{\Delta\varphi}{\Delta t} &= \omega_v - \omega_w\sin\varphi\,\mathrm{ctg}\,\vartheta - \omega_u\cos\varphi\,\mathrm{ctg}\,\vartheta;\\
\frac{\Delta\psi}{\Delta t} &= \omega_w\sin\varphi\,\mathrm{cosec}\,\vartheta + \omega_u\cos\varphi\,\mathrm{cosec}\,\vartheta;\\
X'_n &= X'_{(n-1)} + V_{x'(n-1)}\Delta t + W_{x'(n-1)}\left(\frac{\Delta t^2}{2}\right);\\
Y'_n &= Y'_{(n-1)} + V_{y'(n-1)}\Delta t + W_{y'(n-1)}\left(\frac{\Delta t^2}{2}\right);\\
Z'_n &= Z'_{(n-1)} + V_{z'(n-1)}\Delta t + W_{z'(n-1)}\left(\frac{\Delta t^2}{2}\right).
\end{aligned}\right\} \quad (10.11)$$

The equations of the support base motion are compiled somewhat differently from the equations of motion of the SLM center of mass. In the case when the support bases do not touch the planetary surface, their motion can be completely described by the equations of motion of the SLM body because in such case, the supports are the part of a rigid body. When the support bases are on the planetary surface, they should be viewed as masses which move, being acted upon by the friction forces and by the forces associated with the deforming components of the SLM landing gear (Figure 10.2).

In such case, the SLM motion is a sum of spatial motion of a rigid shell and the motion of the landing support bases along the planetary surface. The forces and moments which appear during the movement of the support bases bring the SLM to rest.

Let us consider the relative motion of the landing supports along the surface. By summating the forces parallel to X and Z axes, we will obtain (see Figure 10.2)

$$\left.\begin{aligned} F_{1ix} - F^{T}_{1ix} - m_i\ddot{X}_{1i} = 0; \\ F_{1iz} - F^{T}_{1iz} - m_i\ddot{Z}_{1i} = 0. \end{aligned}\right\} \qquad (10.12)$$

The total friction force acts along the line of velocity vector. Its components are defined by the following equations

$$\left.\begin{aligned} F^{T}_{1ix} = -fF_{1iy}\dot{X}_{1i}[(\dot{X}_{1i})^2 + (\dot{Z}_{1i})^2]^{-\frac{1}{2}}; \\ F^{T}_{1iz} = -fF_{1iy}\dot{Z}_{1i}[(\dot{X}_{1i})^2 + (\dot{Z}_{1i})^2]^{-\frac{1}{2}}. \end{aligned}\right\} \qquad (10.13)$$

By substituting the expression (10.13) into the equation of motion of the support bases (10.12) we will obtain

$$\left.\begin{aligned} \ddot{X}_{1i} - \frac{fF_{1iy}}{m_i[(\dot{X}_{1i})^2 + (\dot{Z}_{1i})^2]^{\frac{1}{2}}}\dot{X}_{1i} - \frac{F_{1ix}}{m_i} = 0; \\ \ddot{Z}_{1i} - \frac{fF_{1iy}}{m_i[(\dot{X}_{1i})^2 + (\dot{Z}_{1i})^2]^{\frac{1}{2}}}Z_{1i} - \frac{F_{1iz}}{m_i} = 0. \end{aligned}\right\} \qquad (10.14)$$

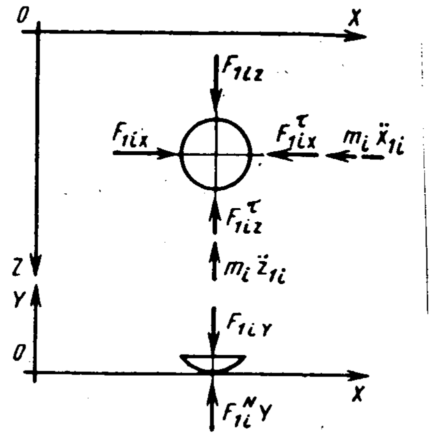


Figure 10.2. Forces, acting upon the spacecraft support base (m_i is the reduced mass of the i support).

If we are to use the assumption that for each sufficiently small interval, in the course of integration, both the forces and the resulting velocities remain constant, then one can reduce the equations of motion of the support bases (10.14) to a system of linear differential equations of the second order: /111

$$\left.\begin{aligned} &\ddot{X}_{1i} - aX_{1i} - b = 0; \\ &\ddot{Z}_{1i} - aZ_{1i} - c = 0, \end{aligned}\right\} \qquad (10.15)$$

where $a = \dfrac{fF_{1iy}}{m_i[(\dot{X}_{1i})^2 + (\dot{Z}_{1i})^2]^{\frac{1}{2}}}$;

$b = \dfrac{F_{1ix}}{m_i}$; $c = \dfrac{F_{1iz}}{m_i}$.

The system of linear differential equations will have the solution of the following kind

$$\left.\begin{aligned} &X_{1i} = X_{1i}^{n-1} - \frac{1}{a}(\dot{X}_{1i}^{n-1} + \frac{b}{a})(1 - e^{a\Delta t}) - \frac{b}{a}\Delta t; \\ &\dot{X}_{1i} = (\dot{X}_{1i}^{n-1} + \frac{b}{a})e^{a\Delta t} - \frac{b}{a}; \\ &Z_{1i} = Z_{1i}^{n-1} - \frac{1}{a}(\dot{Z}_{1i}^{n-1} + \frac{c}{a})(1 - e^{a\Delta t}) - \frac{c}{a}\Delta t; \\ &\dot{Z}_{1i} = (\dot{Z}_{1i}^{n-1} + \frac{c}{a})e^{a\Delta t} - \frac{c}{a}. \end{aligned}\right\} \qquad (10.16)$$

The quantities with the index (n-1) refer to the preceding integration step. Let us note that at the moment of the very first touchdown of the SLM support base (t=0) the 1 point of the support which made the contact is established in such a way that $X_{1,1} = Y_{1,1} = 0$.

During the process of the support base skidding along the planetary surface, we will have the identity Y=0 and $\dot{Y}_{1i} = 0$, since $F_{1iy} = F^N_{1iy}$.

The obtained expressions make it possible to determine in the (XYZ) system the coordinates and velocities of li points in the SLM in the case of contact between the base of the i support and the soil. The recalculation of the above-mentioned parameters in the (uvw) coordinate system is accomplished by using the transformation formulas.

After conducting the above-mentioned calculations for each integration step, one can obtain the complete relationship of all parameters as a function of time, during the SLM landing on the planetary surface. After completing the landing on a planet, the module takes up the position which corresponds to its stable balanced state. However, not every position of the balanced state of the module after landing ensures normal operation of all its systems and the possibility, if it is being planned, for a successful take-off from the planet. /112

Indeed, the flipping of the SLM during landing, on its side, must be classified as a failed landing (and if the crew is present - as an accident).

It is therefore accepted to call the position of the SLM "stable" after the landing when all its supports (or the majority of supports if N>4) make contact with the ground and the longitudinal axis is deflected from the vertical by a small angle, defined by the functional operations of the module (including its take-off).

To evaluate the stability against flipping, and investigating the soft landing process theoretically, one is to introduce a special criterion of the SLM flipping which characterizes the position of the projected center of mass of the module with respect to a broken line, passing through the bases of its supports. The moment of coincidence of the projected center of masses of the module on the plane of supports, with the above-mentioned broken line, is a critical moment and is being viewed as a limiting deflection and if it is exceeded, the module will not return to the stable equilibrated position.

At that moment, the Euler angle θ will exceed its critical value:

$$|\vartheta_{cr}| = \operatorname{arctg}\frac{R\cos\varphi}{|\vartheta_{1i}|}\ (i = 1, 2 \ldots N), \tag{10.17}$$

which is defined by the geometry of the module and by the Euler angle ϕ.

The $|\theta_{cr}|$ is the criterion of the LSM flipping during landing and it is computed for each integration step.

When $|\theta| \geq |\theta_{cr}|$ the space module becomes statically unstable. The integration process stops. If $|\theta| < |\theta_{cr}|$ the module is stable. The integration process in such case may be stopped for example at the moment when the kinetic energy of the SLM will become less than a sufficiently small and positive quantity $E < \eta$.

It should be noted that to generate the solution of the system of the equations of motions (10.2)-(10.7) in their general form is extremely difficult. However in a number of practical cases it would suffice to determine a partial solution of the dynamic equations by numerical integration, utilizing a computer.

The operation of the modules on the planetary surfaces is associated primarily with accomplishment of a set of different dynamic processes (movement, displacement, separation of specific modules and their components, construction operations, etc.). It is for that reason that the determination of dynamic parameters of the SLM design is the most important task. We shall present below briefly the estimates of the effect of design parameters in the mobile modules on the parameters of oscillations, as the module moves along the planetary surface. by knowing the degree of the effect of design parameters on the dynamic parameters of mobile /113 objects, one can reduce significantly in a number of cases the number of special and very time-consuming and expensive experimental studies in determining the true values of the design parameters and one can outline the methods for improving them.

Let us consider the effects of the major design parameters on the oscillations of the mobile wheel-driven transport module (Figure 10.3) which is essentially a planetary mobile vehicle with a hitched semitrailer. Let us compile a set of differential equations of motion which will describe the vertical and lateral-angular oscillations for the calculation diagram as shown in the figure, taking into account the following assumptions: the vehicle moves uniformly and rectilinearly, the suspended part of the vehicle under the springs is considered to be absolutely rigid, the elastic nonlinear parameters of the wheel suspension are linearized within the operational movements.

In using such assumptions, the dynamic system of the vehicle features 21 degrees of freedom.

For example, in the case under investigation, with only vertical oscillations of the vehicle, having in the longitudinal vertical plane 11 degrees of freedom, the system of dynamic equations will be written in the following form

$$a_{i1}\ddot{z}_s + b_{i1}z_s + c_{i1}z_s + a_{i2}\ddot{\phi}_s + b_{i2}\ddot{\phi}_s +$$

$$+ c_{i2}\phi_s + a_{i3}\ddot{\phi}_v + b_{i3}\dot{\phi}_v + c_{i3}\varphi_v +$$

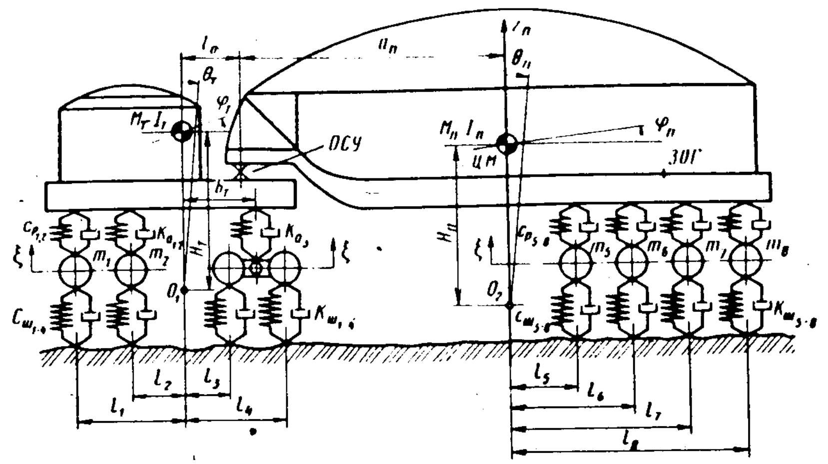


Figure 10.3. Computational diagram of the planetary vehicle.

$$+\sum_{j=1}^{j=t}(b_{ij+3}\dot{\xi}_j+c_{ij+3}\xi_j)=0,$$

when i=1; 2 t=8; when i=3 t=4;

$$b_{i1}z_s+c_{i1}z_s+b_{i2}\dot{\phi}_s+c_{i2}\phi_s+b_{i3}\dot{\phi}_v+$$

$$+c_{i3}\phi_v+a_{ii}\dot{\xi}_j+b_{ii}\dot{\xi}_j+c_{ii}\xi_j=Q_j; \qquad (10.18)$$

when i=4 i=1; when i=5 j=2; /114

$$b_{i1}\dot{z}_s+c_{i1}z_s+b_{i2}\dot{\phi}_s+c_{i2}\phi_s+b_{i3}\dot{\phi}v+$$

$$+c_{i3}\phi_v+a_{ii}\dot{\xi}_j+b_{ii}\dot{\xi}_j+c_{ii}\dot{\xi}_j+$$

$$+b_{i,i\pm1}\dot{\xi}_{j\pm1}+c_{i,i\pm1}\xi_{j\pm1}=Qj;$$

sign "+" when i=6, j=3; sign "-" when i=7, j=4;

$$b_{i1}z_s+c_{i1}z_s+b_{i2}\dot{\phi}_s+c_{i2}\phi_s+a_{ii}\ddot{\xi}_j+$$

$$+b_{ii}\ddot{\xi}j+c_{ii}\xi_j=a_j.$$

Here, each of the values of i=8, 9, 10, 11 corresponds to j=5, 6, 7, 8; the Q_j is the perturbation force, acting from the side of the ground; z_S is the linear vertical displacement of the center of mass of the semitrailer part, suspended under the springs; ϕ_S, ϕ_V are the angular displacements of the suspended part under the springs of the semitrailer and main vehicle, respectively, in the longitudinal vertical plane with respect to the lateral axes of each, passing through their respective center of masses; ξ_j is the linear vertical displacement of the center of masses of the part of the vehicle with the wheel drive j, which is above the spring-loaded suspension (j=1, 2, ..., 8).

The coefficients in front of the generalized coordinates and their derivatives (10.18) indicate the stiffness and damping parameters of the suspension and wheels, respectively and also, the mass-inertia parameters of the parts of the vehicle which are spring-loaded and those which are not.

The comparative evaluation of the effect of change in the major design parameters was based on the maximum root-mean-square acceleration at the specific points and the relative displacements of the semitrailer wheel drive train - the parts which are spring-loaded and those which are not.

The specific support points of the semitrailer part under the springs are the support hitch device, the centers of mass of the semitrailer part under the spring suspension and the rear support of the transportable cargo. Some general relationships have been determined by conducting computerized calculations defining the changes in accelerations at the specific points under the spring-loaded suspension of the semitrailer.

As a result of computerized calculations, some general relationships have been determined in regard to the change in accelerations at the specific points of the spring-loaded part of the semitrailer. By increasing the mass of the moment of inertia in the semitraler part under the spring suspension, of the wheel drive parts above the springs, and also by decreasing the rigidity of the wheel suspension in the semitrailer and of the radial regidity of the wheel rims, one can increase the acceleration in the area of the center of masses of the parts which are under the springs and to decrease the accelerations in the support-hitch device and in the rear supports of the cargo.

The increase in rigidity of the suspension and of the radial rigidity of the wheel rims generates an increased acceleration near the center of masses of the underspring part of the semitrailer and of the rear support of the transportable cargo.

As to the changes in vertical linear displacements of the wheels of parts of the semitrailer which is above the springs with respect to the parts of the semitrailer which is under the springs, as the main vehicle moves, the following has been established. The increase of the mass moments of inertia in the spring-loaded part, in the wheel drives above the springs, the increase of radial rigidity of the wheel rims and lowering of the suspension rigidity will generate a number of linear displacements of the wheel drives of the semitrailer which are above the springs with respect to the part of the semitrailer which is below the springs. /115

The increase in rigidity of the wheel drive suspension in the semitrailer, the coefficients of damping by the energy-absorbing devices, and also the increase in radial rigidity of the wheel rims results in lowered displacements of the semitrailer parts which are above the springs, with respect to those parts which are below.

The analysis of calculated results have shown that the greatest effect on the magnitude of accelerations at the characteristic points of the transport vehicle will be (in the receding order): rigidity of suspension, the mass moment of inertia, radial rigidity of the wheel rim, the coefficients of damping in the energy-absorbing devices, the weight of the wheel drives, the acceleration of which is not compensated by the spring action.

The magnitude of relative displacements during the oscillations of the part of the semitrailer which is spring-compensated, and the wheel drives which are not spring-compensated, will be affected by (in the receding order): rigidity of suspension, the radial rigidity of the wheel rim, the mass moments of inertia in the spring-supported part of the semitrailer, the wheel drive weight in the semitrailer which is not spring-supported and the coefficients of damping in the energy-absorbing devices and in the wheel rims.

The accuracy of obtained results depends significantly on the accuracy of determining the rigidity of the wheel suspension, of the radial rigidity of rims and also of the mass moments of inertia in the spring-loaded part of the semitrailer with respect to the lateral axis, passing through its center of mass. The data on the damping parameters of the suspension and wheel rims and also the levels of mass effect of the spring-loaded wheel drive in the main vehicle play a less important role.

CHAPTER 11.

MATHEMATICAL METHOD OF PLANNING THE SPACECRAFT TESTING

In investigating the complex technological subjects and objects, it is reasonable to utilize the systemic approach, the starting premises of which is to attempt to maximize completely and to take into account all input and output parameters of the object in question. /115

The most well developed and effective methods of practical realization of such systemic approach are the methods of mathematical theory of an experiment (or of experiment planning) which is a concept of idea development, related to the multifaceted analysis.

The efficiency of such a method of studies is higher with the greater complexity of the system which is being studied. Of great significance here is the fact that in solving the extremal problems, the researchers obtain a relatively simple mathematical model which is necessary to automate the test control and to automate the processes and phenomena.

In a general form, the problem of studying the multifaceted processes is being mathematically formulated in the following manner.

It is necessary to obtain a general idea about a function

$$\eta = f(X_1, X_2, \ldots, X_k), \tag{11.1}$$

where η is the parameter of the process, X_1, X_2, ... X_k are the independent variables (factors) which may be manipulated. /116

Ordinarily, the unknown function η is approximated by the polynomial (this is one of the best developed methods of its representation):

$$\eta = \beta_0 + \sum^k \beta_i X_i + \sum^k_{i<j} \beta_{ij} X_i X_j + \sum^k \beta_{ii} X_i^2 + \ldots, \tag{11.2}$$

where β_0, β_i, β_{ij}, β_{ii}, ..., are the coefficients of regression.

The expansion of the function into a power series is equivalent to representation of it by Taylor series, and

$$\beta_1 = \frac{\partial\varphi}{\partial X_1},\ \beta_2 = \frac{\partial\varphi}{\partial X_2}, \ldots, \beta_{12} = \frac{\partial^2\varphi}{\partial X_1 \partial X_2}, \ldots,$$

$$\beta_{11} = \frac{1}{2}\frac{\partial^2\varphi}{\partial X_1^2},\ \beta_{22} = \frac{1}{2}\frac{\partial^2\varphi}{\partial X_2^2}, \ldots$$

The problem is solved by finding the so-called response function $\hat{Y}$ (for the sampled estimate of η) in the following form

$$\hat{Y} = b_0 + \sum^{k} b_i X_i + \sum_{i<j}^{k} b_{ij} X_i X_j + \sum^{k} b_{ii} X_i^2 + \ldots, \tag{11.3}$$

where b_0, b_i, b_{ij}, b_{ii}, ... are the estimates of theoretical coficients of regression β_0, β_i, β_{ij}, β_{ii}, ... Consequently, the solution of this problem is reduced to the determination of b_0, b_i, b_{ij}, b_{ii} ... coefficients which are determined by processing the experimental results.

The most efficient in solving the multifactor problems is the mathematical method of planned experiments which opens up the opportunity to conduct an active (controlled, planned) experiment. By experimental planning, one ordinarily understands the selection of numbers and conditions to carry out the experiment, which are necessary and sufficient to solve the problem at hand.

The methods of planned experiments are most varied and are used to search for optimal conditions and to optimize the parameters in order to obtain the formulas which would reflect the interaction between the factors and explain the interaction mechanisms in the physical phenomena, to select the most important factors and hypotheses, to evaluate and refine the constants of mathematical models. They are applicable to any simple and complex systems which have the properties of being controlled (the values of factors may change as desired by the researcher) and have the necessary degree of reproducibility of results. In such case, the effects and results which are due to numerous strictly determinant, but not controllable, factors are being viewed as random ones. The experiment is subjugated to a specific strategy, based on the set of sequentially and logically thought out operations and all influential factors are being changed simultaneously.

The mathematical theory of planned experiments in a general case does not limit either the number of studied factors or the number of either evaluated or optimized parameters. However, in practical realization of the methods which are used with this theory, and in order to improve their efficiency, one attempts to reduce both the number of investigated factors and the number of optimized parameters (the output indicators). Here one utilizes extensively the a priori information about the behavior of the object in question and one utilizes specially developed methods of sorting out the factors of small significance (the method of ranking, the random search and others), then the transition to the generalized parameter of optimization and the consideration of the part of optimization parameters which are used as the constraints. /117

The object of study in the planned experiments is the "black box" which is used in cybernetics, the inputs of which represent the factors which correspond to the methods which could be used to affect the object in question, and the outputs are the parameters of the process (of the optimization).

The researcher looks for the relationship between the input and output parameters within the system, on the basis of the cybernetic approach, detaching himself from the elementary processes which take place within the system. This releases him from the need to describe the phenomena in terms of differential equations and makes it possible in the case when they cannot be compiled (in other words, as one investigates the complex systems which previously were studied only intuitively) to study and optimize the whole process.

In the course of such approach, the relation of the researcher and the strategy of the experiments changes, namely:

1) one introduces into the experiment the concept of a random case, thus a large number of interfering factors which previously had to be stabilized are transformed by randomization into the random quantities, and their effect on the experimental results may be accounted for and controlled;

2) the researcher knows quite well which experiments must be carried out and how to interpret the observed results, in other words, he is able to conduct the controlled (active) experiment;

3) the active experiment may be organized quite well. It alleviates all assumptions to which the multiple regression analysis is so susceptible;

4) the planning of an experiment makes it possible to increase sharply the efficiency of research, by a considerable decrease in errors, while determining optimization parameters.

It is impossible in practical environment to encounter an ideally strict and rigid functional relationships. Each phenomenon, in addition to the influence of the major factors, is affected by random errors and interferences, resulting in the data scattering (Figure 11.1).

Let us split the whole region of observations, in terms of the argument X and into k equivalent intervals. Let us calculate the number of points θ_j within each interval, and let us calculate the appropriate mean values of $\overline{\theta}_j$. Let us mark off $\overline{\theta}_j$ at the center of each interval. By connecting the $\overline{\theta}_j$ points, using the straight lines, we will obtain a broken line which is the experimental line of regression. If one is to increase the total number N of observations and simultaneously if one is to decrease the size of intervals, then the broken line will become smoother and smoother, and at $N \to \infty$ it will become the theoretical regression line.

But the theoretical regression line (or surface) may turn out to be too complex for analytical recording. The theory of planned experiment suggests to use for practical purposes, not the theoretical regression line (or surface) itself, but its approximation in the form of an expansion in terms of the algebraic Taylor power polynomial, by the use of which it becomes possible to describe adequately the theoretical regression line. The methods /118
of finding the coefficients of such expansion, and their statistical evaluation, is developed within the framework of the mathematical theory of planned experiments.

In approaching and searching for the mathematical model (the response function) the researcher has at his disposal a certain a priori information. The amount of this information may be arbitrarily characterized by three basic levels:

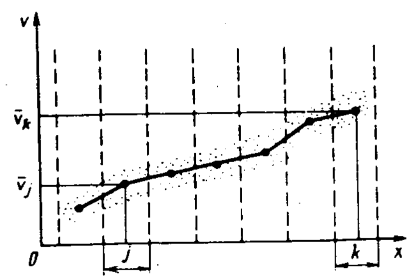


Figure 11.1. Experimental regression line.

a) the response function is known. One must determine or refine its unknown coefficients.

b) it is known that the response function coincides with one of the available relationships, but it is not known which one. It is necessary to determine which of the functions is a true one and to determine its unknown coefficients;

c) the shape of the response function is not known. It is only known that the response function, within the area of interest for the researcher can be sufficiently well approximated by a finite series of the a priori known functions. It is necessary to find the best description of the response function.

The planning of an experiment in having the a priori information, corresponding to b level is being brought about sequentially - in stages.

The most difficult and least developed is the planning of an experiment when the response function is completely unknown (the c level). As it seems, it is impossible to plan an experiment enabling us to solve such a problem completely. However, its solution may be reduced to a certain sequential procedure, which includes the alternation of a series of experiments, planned by using the methods which correspond to both preceding levels.

Optimality Criteria of Planned Experiment

At the present time we have about 20 well-formulated criteria of optimal planning for different situations, utilizing which the researcher has at his disposal the experimental points within the factor-related space and can process the results of observations. For the purposes of apparency, these interpretation criteria are split into two large groups. The first group includes the criteria related to the accuracy of estimated coefficients of regression and the second one - the criteria and properties of planning, associated with the error in estimating the response surface. Let us list the major optimality criteria and properties of planning [16, 71].

The criteria of the first group are essentially reduced to a set of special requirements, imposed on the covariant matrix (and consequently, the informational matrix): /119

D-optimality. The D-optimal planning corresponds to the informational matrix $M=1/N(X^T \cdot X)$ with the greatest determinant (or the covariant matrix $N^{-1}\sigma^2 M^{-1}$ with the least determinant, which is the same) in terms of all possible multiplicity of plans. The D-optimal plan minimizes the generalized dispersion or the volume of an ellipsoid, which defines the scattering in the coefficients of regression (here X^T is the transposed matrix of independent variables X_{ij} which may be both the quantitative and qualitative, and N is the total number of observations).

The E-optimality is the E-optimal plan, corresponding to the least maximum eigenvalue of the covariant matrix. The E-optimal plan minimizes the major axis of the ellipsoid of scattering.

For many models, the construction E-optimal plans is associated with great difficulties and therefore one sometimes utilizes the following criterion of optimality which is close to the former one: the minimum sum of squares of deviations in the eigenvalues of the covariant matrix of estimates, from the mean values.

The A-optimality criterion corresponds to a plan with the minimal mean dispersion of estimated coefficients or with the least value of the covariant matrix trace. The A-optimal plan minimizes the sum of squares of major semiaxes of the ellipsoid of scattering.

Let us also note the following criteria which pertains to the first group:

The minimum of maximal dispersion, in evaluating the coefficients. Geometrically, this means the minimum of maximal of projected axes of the ellipsoid of scattering on the coordinate axes within the space of the parameters.

The minimum sum of relative error estimates.

The orthogonal planning. The orthogonal planning with diagonal covariant (informational) matrix provides the generation of independent estimates of the coefficients of regression. The ellipsoid of scattering is oriented in such a way that the direction of its major axes coincides with the directions of the coordinate axes within the space of the parameters.

Among the criteria of the second group, associated with the error in estimating the response surface, let us note the following:

G-optimality. The G-optimal plan provides for the least, with respect to all plans, maximal magnitude of the dispersion, as predicted within the planning domain.

Minimum of the mean dispersion in estimating the response surface.

Rotatability planning. The rotatability planning which has the covariant matrix, invariant with respect to orthogonal rotation of coordinates, makes it possible to obtain similar dispersion of the predicted values of response function in all experimental points equidistant from the center. The fulfillment of this condition makes any direction from the experimental center equivalent, in the sense of accuracy in evaluating the response surface.

Maximal accuracy in the estimate of coordinates of extrema.

The minimum of general (random or systematic) root-mean-square /120 deviations, in estimating the model of the response surface.

In addition to the above-mentioned criteria which reflect the most significant requirements, one should also note some other practically important criteria:

Composition planning. The composition planning provides for the utilization of the points within a plan, constructed to represent the results polynomial of S power as a submultiple set of points for the optimal plan on the order of s+1. This problem will arise when the polynomial of s power inadequately represents the observed results.

The splitting of the plan into orthogonal blocks to exclude the effect of inhomogeneities while conducting the experiment and to evaluate this effect. This criterion has a direct relationship to the composition plan. For example, the composition rotatability planning of second order may be split into the orthogonal blocks.

Uniformity. This criterion requires that the dispersion in evaluated model within a certain region around the center of an experiment would essentially remain the same. This property is ordinarily associated with the rotatability property.

Let us revert now to such properties of planning which should probably not be viewed as the major optimality criteria, but as practice shows, the fulfillment of which is quite desirable.

The simplicity of calculations and apparency of representation of results. From the point of view of this criterion, it is reasonable to accept such plans for which the whole processing may be conducted by using simple formulas.

Insensitivity to errors in independent variables. The insensitivity to coarse errors in the observed results.

Possiblilty of transforming the independent variables, preserving the planned optimality.

The above-mentioned list may be expanded by including the optimality criteria for the plans involving the discriminating experiments, the planned dispersion analysis, in planning to track the uncontrollable time-related drift of the complex systems under investigation and others.

In a general case, the criteria selection is determined directly by the way the experiment is set up. In practice, the problem of selection may be solved numerically - by selecting from the appropriate data sets related to the problem at hand, such criteria by the use of which the optimality losses pertaining to the other criteria (or pertaining to one of those criteria) are minimized [16].

Full Factorial Experiment and Fracional Replicate

The selection of factors in planning an experiment, in other words the selection of the variables which act upon the object which is being investigated, is one of the important tasks at hand. The factors must be controllable and unambiguous. By the factorial controllability one should understand the ability to establish its required value for a given set of experimental conditions or to change them as per the assigned plan. The sum of factors must be compatible, in other words all combinations of factors should be realizable. /121

One should differentiate the quantitative and qualitative factors. The examples of quantitative factors are self-apparent. Among the qualitative factors are the materials, devices, assemblies, executive devices, technological methods, etc. The qualitative factors may always be made quantitative, by evaluating them for example, using the five point number system.

In proceeding with the investigation, it is necessary to take into account all possible factors (to obtain their total list) and then to discard the insignificant factors.

At the first stage, one may utilize for such purposes the a priori information. The use of a priori information involves the formalization of the experiment by experts who rank the factors in terms of their effect on the parameter which is to be investigated. A further sorting out of factors is conducted experimentally. For experimental factorial selection, one may utilize the full factorial experiment, or the fractional replicates. However, in having a large number of factors it would be more reasonable to utilize the method of random balancing, the purpose of which is to elucidate the true ranking curve.

Another and no less important task in planning an experiment is the selection of optimization parameter, by which we mean the quantitative characteristic of the research target.

The optimization parameter must be effective from the point of view of the goal-attainment, it must be quantitative, unambiguous and statistically effective, in other words, it must have the least dispersion. In addition, the optimization parameter must be a simple one, it must have a clearly-defined physical meaning, it should be easily calculated and must exist for all different states of the system. It is necessary to strive for the conditions when the optimization parameter would be the only one which can be used. This is explained by the simplicity of solving the singular parametric problems. However, one encounters in practice also the multi-parametric problems. In such case, it is necessary first of all to attempt to decrease the number of optimization parameters by generalizing them and considering a number of parameters as the constraining quantities.

In using the traditional method of testing, by using a single parameter experiment, all variables except the one which is being investigated are fixed at constant levels. Such sequential variation of variables requires an extensive amount of labor, time and expenditures.

In employing the factorial planning, which removes this shortcoming, the number of levels for each of the factors cannot be less than two. The experimental planning in which all factors are being varied at two levels are called plans of 2^k type, where k is the number of factors. The experiments in which one obtains all possible combinations of the levels of all factors are being called the full factorial experiments.

The recording of interval boundaries in relative units ±1 results in a standard form of the matrix of planning recording which employs only the signs or symbols.

Let us consider the experiment with three independent variables /122 X_1, X_2, X_3 which are being varied at two levels (of 2^k type). In order to exhaust all combinations of three factors at two levels, one must set up eight experiments ($2^3=8$). The planning matrix for three variables is presented in Table 11.2, and the geometric representation of such a plan is shown in Figure 11.2.

The geometric representation of plan for $k>3$ are the hypercubes in k-dimensional space.

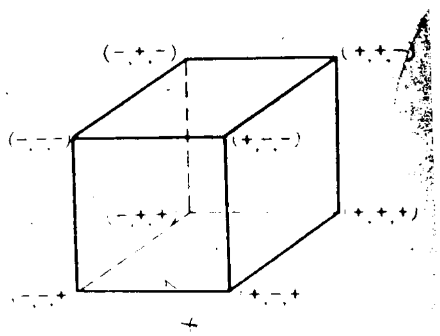


Figure 11.2. Geometric representation of the plan of 2^3 type.

The planning matrices of full factorial experiments are orthogonal and therefore, the coefficients of regression for the model of interest are determined by using a simple formula

$$b_j = \frac{\sum_{i=1}^{n} X_{ij} Y_i}{n}, \quad j = 0, 1, \ldots, k, \qquad (11.4)$$

where Y_i is the optimization parameter in the i experiment (line); X_{ij} is the value of j factor in the i experiment.

The obtained estimates of coefficients in this model are independent of each other. Their numerical values and signs indicate the intensity and the character of the effect of such factors.

The full factorial experiment (FFE) makes it possible if needed to estimate the effect of the interaction between the factors. To do this, one must, by using the rules of column multiplication, obtain the columns of factorial products. One handles such columns in the same way as any other columns of any factors, since while adding the interaction columns, all matrix properties are retained. For the full factorial experiment of 2^3 type, the planning matrix, taking into account the interactions, is shown in Table 11.2. In this matrix, the X_1, X_2 and X_3 columns assign the planned steps (defining the experimental conditions) and the columns X_1X_2, X_1X_3 and others are used only for the calculations.

The total number of all possible effects, including b_0, the linear effects and interactions of all orders, is equal to the number of experiments. From the full factorial experiment it is impossible to extract the information about the quadratic terms of the model, because the corresponding evalutions are mixed up with b_0.

In the full factorial experiment, the number of experiments exceeds the number of coefficients of linear models as a function of the increase in the number of factors. If there is no need here to estimate the effects of interaction, or if part of such interactions may be disregarded, the matrix will have excess experiments and one could get by with a smaller number.

In order to reduce the number of experiments, a matrix column should be attached to each new factor, which is interacting and which can be disregarded. By disregarding one interaction, we /123 will obtain a semireplicate (one half) from the full factorial experiment, by disregarding two interactions - one quarter of the replicate, etc. The regular fractionate replicate may be obtained by dividing the plans of full factorial experiments into a number which is a multiple of two, utilizing the 2^{k-p} designation, where p is the number of linear effects, equated to the interaction effects. In a similar fashion, the semireplicate of 2^3 will be written in the 2^{3-1} form and the quarter replicate of 2^5 - in the 2^{5-2} form.

The fractional replicate may have different mixing system. For example, the 2^{3-1} planning may be represented by two semireplicates (two halves of a full factorial experiment) each of which is assigned by one of the developed relationships: $X_3=X_1X_2$ and $X_3=-X_1X_2$. If both sides of these relationships are multiplied by X_3, then we will obtain $X_3^2=1=X_1X_2X_3$, and $X_3^2=1=X_1X_2X_3$, because $X_i^2=1$ equality always applies. The relationships $1=X_1\ X_2\ X_3$ and $1=-X_1X_2X_3$ are being called the defining contrasts. By using the defining contrasts, it is easy to determine all mixed estimates. To do this, it is necessary to multiply sequentially all independent variables by the defining contrast.

The fractional replicates are used extensively to obtain the linear models. The effectiveness of their use increases with the increase in the number of factors. At the same time, the success of using the fractional replicates depends on the selection of intervals within the factors being varied, and the selection of the mixing system for the linear effects and the interaction effects. The art of a researcher is determined by his skill to select properly the mixing system. It is of great importance here to have the a priori data about the significance of interactions. The so-called block planning is used under the following circumstances, which are encountered quite frequently:

the materials and components which are being used in the experiment change their properties as a function of time (one observes the time-related drift);

the components of a product which is being made, and experimental components are received in batches at different periods of time. Consequently, their parameters are somewhat inhomogeneous;

the experiments which are being conducted at different work shifts or at different test stands may have traces of different environmental conditions, associated with the specifics of their behavior.

The effect of such activities may be weakened to a considerable degree. To do this, one must split a series of experiments into separate blocks in such a way that the effect of the time-related drift (the effect of separate batches of materials, of different work shifts, etc.) would be mixed inside of a block with such interactions which could be disregarded.

In the planning of 2^k type, all experiments may be split into two blocks (either $2^2=4$ or $2^3=8$ blocks, etc.). The block which contains the experiment (1) and which includes all major factors, is being taken at the lowest level and is being called the major block.

To develop the blocks for 2^k plan means to generate a list of experiments, using the code designations which are incorporated within each block. In the course of such process, it is necessary to isolate the variable (or a specific interaction) which characterizes the time-related drift (the batch of materials, the work shift, etc.) which is being called the block variable X_b (the block /124
effect). Let us combine into a single block (into separate blocks) those experiments in which the block variable X_b is incorporated only on either the upper or lowest levels.

Example. It is required that all $2^3=8$ experiments of the FFE plan 2^3 are split into two blocks. As a block variable, let us use the highest interaction X_b=ABC which is being called a priori, as the least significant. The blocks are shown in Table 11.1.

Ordinarily, the block schematic may be recorded in a brief fashion, namely, for the 2^3 plan (ABC is mixed with the interblock effect):

block 1: (1), ac, bc, ab,
block 2: c, a, b, abc;

or, even in a briefer form, by inscribing only the major block and the block variable;

plan 2^3, ABC=X_b; (1), ac, bc, ab.

There are extensive catalogues which can be used to split various plans into blocks. It should be noted that the block-by-block planning is being successfully used also for the fractional factorial experiment.

If one is not certain that the dispersions are homogeneous, one must set up the parallel experiments at all (or at least, at several) points, to calculate the respective dispersions and to verify their homogeneity. Ordinarily, one utilizes for these purposes the Cochran criterion (it is being used for the same number of parallel experiments).

One assumes that the dispersions are homogeneous if the Cochran criterion G does not exceed the table value G_{table}.

The experimental error in the case of homogeneous dispersion is equal to

$$\sigma_y^2 = \frac{\sum_{i=1}^{n} \sigma_i^2}{n}, \tag{11.5}$$

In having at one's disposal the experimental error, one can clarify the question whether the model is applicable, in other words, whether it is adequate. To verify such adequacy, one uses ordinarily the Fisher criterion F:

$$F = \frac{\sigma_{ad}^2}{\sigma_y^2}, \quad \sigma_{ad}^2 = \frac{\sum_{i=1}^{n} (Y_i - \hat{Y}_i)^2}{n - k - 1}, \tag{11.6}$$

where σ_{ad}^2 is the dispersion of adequacy; $\hat{Y}_i$ is the computed value of the optimization parameter, predicted by the equation for the i experiment; k is the number of factors and n is the number of experiments.

TABLE 11.1.

Blocks	Code designations of the experiments	Full factorial experiment (FFE) of 2^3 type, splitting into 2 blocks								Y_u
		I	A	B	C	AB	AC	BC	$ABC=X_b$	
1 (Major)	(1)	+	–	–	–	+	+	+	–	Y_1
	ac	+	+	–	+	–	+	–	–	Y_2
	bc	+	–	+	+	–	–	+	–	Y_3
	ab	+	+	+	–	+	–	–	–	Y_4
2	c	+	–	–	+	+	–	–	+	Y_5
	a	+	+	–	–	–	–	+	+	Y_6
	b	+	–	+	–	–	+	–	+	Y_7
	abc	+	+	+	+	+	+	+	+	Y_8

If for the selected level of significance, the computed value of F criterion does not exceed the F from the tables, the model is adequate. /125

To evaluate the levels of coefficients, one is to construct the confidence intervals:

$$\Delta b_i = \pm \sigma_{b_i} t. \qquad (11.7)$$

where Δb_i is the confidence interval of i coefficinet, t is the Student criterion for the selected level of significance (ordinarily, equal to 5%), σ_{b_i} is the root-mean-square error in the coefficient:

$$\sigma_{b_i} = \pm \sqrt{\frac{\sigma_y^2}{n}}. \qquad (11.8)$$

In the case of linear or incomplete quadratic model, the confidence intervals for coefficients are equivalent.

The coefficients which are smaller in terms of their absolute values than the corresponding confidence intervals are insignificant. Insignificant coefficients will be obtained for the factors which do not affect the optimization parameters.

In constructing the interpolation formulas, once the adequate model is obtained, indicates the end of the problem-solution and during the optimization process - the ability to proceed in moving along the gradient (employing the method of steep ascent).

Example of Mathematical Model Construction

Let us consider an example (of an illustrated nature) for obtaining the interpolation formula to estimate the effect of the load factor at the center of mass of the Martian landing module n_y, the inclination angle of the Martian surface Θ^0, the pitch angle ϕ^0 and the relative height of the center of mass of the module $\bar{H}$, assuming that the tests on a stand are conducted with the remaining conditions being stable (at a fixed level are the landing velocity, the specific parameters of the energy-absorbing devices and of the soil properties, the geometric parameters and the spacecraft orientation). Such a problem will be solved by the FFE method of 2^3 type. On the input we will have the following threee factors (independent variables): $\tilde{X}_1 = \Theta^0$, $\tilde{X}_2 = \Theta^0$ and $X_3 = \bar{H}$, and on the output we will have the optimization parameter $Y = n_y$. Both, the optimization parameter and the other factors satisfy the requirements which are imposed upon them.

Table 11.2 shows the FFE plan for three factors which are being varied at two levels, recorded in the natural $\tilde{X}_j$ and coded X_j figures (j=1, 2, 3). We will also have the averaged-out results of two parallel experiments along each planning matrix column Y_i and the calculated results $\hat{Y}_i$ based on the obtained mathematical model (11.11).

We will be looking for an interpolation formula in the following form

$$\hat{Y} = b_0 + b_1 X_1 + b_2 X_2 + b_3 X_3 + b_{12} X_1 X_2 + b_{13} X_1 X_3 + \\ + b_{23} X_2 X_3 + b_{123} X_1 X_2 X_3 . \qquad (11.9)$$

By computing the coefficients for the model (11.9) using the formula (11.4) we will obtain $\hat{Y} = 10.5 - 0.5X_1 + 1.5X_2 - 0.25X_3 - X_1X_2 - 0.25X_1X_3 + 0.25X_2X_3 - 0.25X_1X_2X_3$ (11.10). /126

The dispersions of the obtained results are homogeneous because

$$G = \frac{\sigma_i^2 \max}{\sum_{i=1}^{i=8} \sigma_i^2} = \frac{0,245}{0,99} = 0,248 < G_{table} = 0.6798 \text{ when } \alpha = 0,05.$$

TABLE 11.2

i	$\tilde{X}_1=\theta^0$, degrees	$\tilde{X}_2=\varphi^0$, degrees	$\tilde{X}_3=\bar{H}$, 0	X_1	X_2	X_3
1	− 20	− 12,5	0,4	− 1	− 1	− 1
2	+ 20	− 12,5	0,4	+ 1	− 1	− 1
3	− 20	+ 11,5	0,4	− 1	+ 1	− 1
4	+ 20	+ 11,5	0,4	+ 1	+ 1	− 1
5	− 20	− 12,5	0,6	− 1	− 1	+ 1
6	+ 20	− 12,5	0,6	+ 1	− 1	+ 1
7	− 20	+ 11,5	0,6	− 1	+ 1	+ 1
8	+ 20	+ 11,5	0,6	+ 1	+ 1	+ 1

i	X_1X_2	X_1X_3	X_2X_3	$X_1X_2X_3$	$Y_i=n_y$, [0]	$\hat{Y}_i$ [0]
1	+ 1	+ 1	+ 1	− 1	9	8,5
2	− 1	− 1	+ 1	+ 1	10	9,5
3	− 1	+ 1	− 1	+ 1	13	13,5
4	+ 1	− 1	− 1	− 1	11	10,5
5	+ 1	− 1	− 1	+ 1	8	8,5
6	− 1	+ 1	− 1	− 1	9	9,5
7	− 1	− 1	+ 1	− 1	14	13,5
8	+ 1	+ 1	+ 1	+ 1	10	10,5

(Note: commas in tabulated material are equivalent to decimal points)

The experimental error σ_y^2 and confidence interval for the coefficients of Δb_i model are determined by using the formulas (11.5), (11.7) and (11.8):

$$\sigma_y^2=\frac{0.99}{8}=0.124;\ \sigma_{bi}=\pm\sqrt{\frac{\sigma_y^2}{n}}=\pm\sqrt{0.0155}=$$
$$=\pm 0.124;\ t=2.365;\ \Delta b_i=\pm\sigma_{bi}t=\pm 0.294.$$

We can see that the insignificant coefficients are b_3, b_{13}, b_{23} and b_{123}.

We will obtain now the mathematical model of interest

$$\hat{Y}=10.5-0.5X_1+1.5X_2-X_1X_2 \quad \text{or} \quad n_y=10.5-0.5\theta+1.5\phi-\theta\phi \tag{11.11}$$

and we can verify its adequacy using the Fisher F criterion (11.6):

$$\sigma^2_{ad} = \frac{\sum_{i=1}^{n} (Y_i - Y_i)^2}{n-k-1} = \frac{2.0}{5} = 0.4;$$

$$F = \frac{\sigma^2_{ad}}{\sigma^2_y} = \frac{0,4}{0.124} = 3.22 < F_{table} = 3.69 \text{ when } \alpha = 0.05.$$

Consequently, the interpolation formula (11.11) is adequate. It shows that the greatest effect which will be exerted on n_y is the pitch angle ϕ and that with the pitch nose-up, the load factor will increase. The n_y will increase as the module lands down-slope $\theta < 0$, particularly when the inclination angle and the pitch angle have different signs. The effect of the altitude change $\bar{H}$ of the module center of mass within the range under consideration on the load factor will be disregarded.

Planning of the Second and the Third Order /127

In parallel with the comparatively simple planning methods which provide the development of linear and incomplete quadratic models, one utilizes also more complex methods, designed to obtain the models of the second and third order. In the curvilinear region of the response surface, the interaction and quadratic effects become important. Such surfaces are described by nonlinear equations [16]. At the present time, the planning is well developed for the polynomials of the second order.

Let us consider briefly the so-called composition (or sequential) planning, as proposed by Bocks and Wilson (Figure 11.3). The nucleus of such planning is the FFE of 2^k type (when $k<5$) or the fractional replicate of it, of 2^k-p type (when $k \geq 5$).

The study commences with constructing the linear plan. If the linear equation of regression turns out to be inadequate, it is necessary to:

add 2k "star" points, located on the coordinate axes of the factorial space $(\pm\alpha, 0, \ldots, 0)$; $(0, \pm\alpha, \ldots, 0)$, ... $(0, 0, \ldots, 0 \pm\alpha)$ where α is the distance from the center of the plan to the "star" point ("star shoulder");

to increase the number of experiments at the center of the plan (n_0). The total number of experiments within the composition plan matrix, having k factors, will be (see Figure 11.3):

$$n = 2^k + 2k + n_0. \qquad (11.12)$$

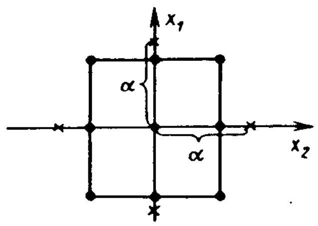
The magnitude of "star shoulder" α and the number of experiments at the center of the plan n_0 depend on the type of plan selected.

The composition plans can be easily reduced to the orthogonal samplings and selection of the "star shoulder" α. In such case, there are no constraints imposed on the number of points at the center of the plan n_0. In this case, ordinarily n_0 is assumed to be equal to 1.

The magnitude of the "star shoulder" α is selected assuming that the nondiagonal term of correlation matrix (X^1, X^{-1}) is equal to zero.

The values of α for different number of factors are presented in Table 11.3 [16]:

/128



$$\alpha = \pm [2^{\frac{k-2}{2}} \sqrt{n} - 2^{k-1}]^{1/2}.$$

Figure 11.3. Composition plan for k=2.

TABLE 11.3

	Number of factors k			
	2	3	4	5
Magnitude of α	1,000	1,215	1,4142	1,547
Nucleus of the plan	2^2	2^3	2^4	2^{5-1}

The coefficients of equations of regression, obtained by using the orthogonal plans of the second order are determined with different degrees of accuracy, while the orthogonal plans of the first order provide similar accuracy of the coefficients.

The adequacy of equation

$$\hat{Y} = b_0 + b_1 X_1 + b_2 X_2 + \ldots + b_k X_k + \sum_{i<j}^{k} b_{ij} X_i X_j + \sum_{i=1}^{k} b_{ii} X_i^2$$

is verified by using the Fisher criterion and compiling the relationship between the dispersions $F_{exp} = S_{ad}^2 / S_0^2$, where S^2_{ad} is the residual dispersion, S^2_0 is the reproducibility dispersion (averaged-out). The equation is adequate if $F_{exp} \leq F_{table}$ which is the tabulated value calculated for the preselected level of significance and for the number of degrees of freedom.

The orthogonal plans of the second order do not display the rotatability and consequently, the errors in determining Y at the experimental points on the response surface may be lower than in the calculations which were obtained by using the equations of regression.

So that the composition plan would display rotatability, the magnitude of the "star shoulder" α is selected by assuming that $\alpha=2^{k/4}$ (for the nucleus containing FFE) and $\alpha=2^{(k-1)/4}$ (for the nucleus containing semireplicates).

The number of points at the center of the plan n_0 is increased so as to make the matrix X^T, X nondegenerative (Table 11.4 shows the values of α and n_0 for different number of independent factors k.

The rotatability plan of the second order for k=2 is shown in Table 11.5. The rotatability planning of the second order is non-orthogonal because

$$\sum_{i=1}^{n} X_{0i}X_{ij}^2 \neq 0, \quad \sum_{i=1}^{n} X_{ik}^2 X_{ji}^2 \neq 0.$$

The b_{ii} coefficient are correlated with each other and correlated with the free term b_0.

TABLE 11.4. /129

Number of factors k

	2	3	4	5	5	6	6	7	7
Nucleus of plan	2^2	2^3	2^4	2^5	2^{5-1}	2^6	2^{6-1}	2^7	2^{7-1}
Magnitude of α	1,414	1,682	2,00	2,378	2,00	2,828	2,378	3,333	2,828
Magnitude of n_0	5	6	7	10	6	15	9	21	14

(Note: Commas in tabulated material are equivalent to decimal points.)

One proceeds with the planning of the third order only after it has been established that the mathematical models of the second order are not applicable. Therefore, the plans of the third order are ordinarily of composition type.

As one begins to develop the plans of third order, the experimental data processing is associated with the calculation of a large number of coefficients of regression in the equations of third degree.

For example, for k=2, such equation will have the following form

$$\hat{Y} = b_0 + b_1 X_1 + b_2 X_2 + b_{11} X_1^2 + b_{22} X_2^2 + b_{12} X_1 X_2 + \\ + b_{112} X_1^2 X_2 + b_{122} X_1 X_2^2 + b_{111} X_1^3 + b_{222} X_2^3. \tag{11.13}$$

In the case of two factors, one obtains the rotatability plan consisting of 16 experimental points - eight points uniformly located on the circumferences of 1.414 and 2.121 radii [16].

If k=3, then the number of points within the rotatability plan of the third order will significantly increase. In such case, one first executes the first 24 experiments, forming the rotatability plan of second order (cube, double octahedron and four zero points). If it turns out that the model of the second order is not adequate, then the plan should be built up to the level of the rotatability plan of the third order by introducing additionally 26 points. There are formulas available [16] which can be used to determine the coefficients of regression for the plan of third order and also to evaluate the significance of these coefficients. The adequacy of mathematical model of the third order is verified as usual by utilizing the Fisher criterion.

It should be noted that on the basis of planned experimental methods, it is possible to solve problems from the scientific-technological and economic prognostication domain by extrapolating /130
the information across the hypersurfaces. For example, by using the rotatability plans, it is possible to prognosticate τ years in advance, if one has the data for the preceding time of 3τ.

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TABLE 11.5.

i	X_0	X_1	X_2	X_1X_2	X_1^2	X_2^2	Y_i
1	+1	+1	+1	+1	+1	+1	92,8
2	+1	+1	-1	-1	+1	+1	88,9
3	+1	-1	1	+1	+1	+1	87,1
4	+1	-1	+1	-1	+1	+1	79,0
5	+1	+1,414	0	0	2	0	94,0
6	+1	-1,414	0	0	2	0	85,6
7	+1	0	+ 1,414	0	0	2	80,0
8	+1	0	1,414	0	0	2	84,5
9	+1	0	0	0	0	0	83,7
10	+1	0	0	0	0	0	86,0
11	+1	0	0	0	0	0	85,8
12 n_0	+1	0	0	0	0	0	83,9
13	+1	0	0	0	0	0	86,3

(Commas in tabulated material are equivalent to decimal points)

Regular Factorial Plans

In the theory of planning an experiment the regular factorial plans have been developed which may be used to solve the problems having either the quantitative or qualitative factors.

The regular factorial planning assumes that the models of interest are linear with respect to the coefficients of regression. For example, in the model of second order the factors may be either linear or quadratic and the coefficients b_0, b_1, b_2, b_{11}, b_{22} etc. may be linear:

$$Y = b_0 + b_1X_1 + b_{11}Z_1 + b_2X_2 + b_{22}Z_2 + \ldots + \Pi,$$

where $X_i=k_i(\tilde{X}_i+A_i)$ defines the linear relationship between the coded (X_i) values and natural values $(\tilde{X}_i)$, $Z_i=k(X_i^2+bX_i+c)$ defines the quadratic relationship with the linear quantities of (X_i), $\Pi=b_{12}X_1X_2+b_{112}Z_1X_2+b_{122}X_1Z_2+b_{1122}Z_1Z_2$ reflects the contribution of interactions.

In practical problems, as one can see from Table 11.6, the models of major effects which have considerably fewer unknown coefficients than the full factorial models, turn out frequently to be quite adequate. Naturally, for a smaller number of unknown coefficients, more compact plans with fewer number of experiments would be required.

TABLE 11.6

Model complexity	Number of factors K	Model of major effects: Model type	Number of coef-ficients sk+1	Full factorial model: Model type	Number of coefficients (s+1)k
Model of the first order s=1	$k=1$	$Y=b_0+b_1X_1$	2	$Y=b_0+b_1X_1$	2
	$k=2$	$Y=b_0+b_1X_1+b_2X_2$	3	$Y=b_0+b_1X_1+b_2X_2+b_{12}X_1X_2$	4
	$k=3$	$Y=b_0+b_1X_1+b_2X_2+b_3X_3$	4	$Y=b_0+b_1X_1+b_2X_2+b_3X_3+$ $+b_{12}X_1X_2+b_{13}X_1X_3+$ $+b_{23}X_2X_3+b_{123}X_1X_2X_3$	8
Model of the second order s=2	$k=1$	$Y=b_0+b_1X_1+b_{11}Z_1$	3	$Y=b_0+b_1X_1+b_{11}Z_1$	3
	$k=2$	$Y=b_0+b_1X_1+b_2X_2+$ $+b_{11}Z_1+b_{22}Z_2$	5	$Y=b_0+b_1X_1+b_{11}Z_1+b_2X_2+$ $+b_{22}Z_2+b_{112}X_2Z_1+b_{221}X_1Z_2+$ $+b_{1122}Z_1Z_2+b_{12}X_1X_2$	9
	$k=3$	$Y=b_0+b_1X_1+b_2X_2+$ $+b_3X_3+b_{11}Z_1+b_{22}Z_2+$ $+b_{33}Z_3$	7	$Y=b_0+b_1X_1+b_{11}Z_1+b_2X_2+$ $+b_{22}Z_2+b_3X_3+b_{33}Z_3+$ II	27
	$k=4$	$Y=b_0+b_1X_1+b_2X_2+$ $+b_3X_3+b_4X_4+b_{11}Z_1+$ $+b_{22}Z_2+b_{33}Z_3+b_{44}Z_4$	9	$Y=b_0+b_1X_1+b_{11}Z_1+b_2X_2+$ $+b_{22}Z_2+b_3X_3+b_{33}Z_3+b_4X_4+$ $+b_{44}Z_4+$ II	

There are symmetric and asymmetric plans. The symmetric plans are such in which all factors have the same number of levels h=const, and the asymmetric ones are such in which one has the factors with different number of levels (h≠const).

For example, the symmetric regular factorial plan, constructed on the basis of nine experiments, makes it possible to investigate not more than four factors and to obtain only the model of major factors of the second order. The coded matrix of such plan is shown in Table 11.7.

In switching over to the coded matrix, the conditions of orthogonality were taken into account:

$$\sum_{i=1}^{n} X_i = 0, \quad \sum_{i=1}^{n} X_i X_j = 0, \quad \sum_{i=1}^{n} Z_i Z_j = 0,$$

$$\sum_{i=1}^{n} X_i Z_j = 0, \quad \sum_{i=1}^{n} Z_i = 0$$ with respect to each factor.

The need to construct the nonsymmetric regular factorial plans arises when the number of levels h with respect to the factors is different. One obtains the asymmetric plans from symmetric ones by transformations.

The "compression" method. Let us assume that we have a regular factorial plan for four factors. It is known from the conditions that two factors may acquire the values not at the three levels, but only at two. Then in proceeding from the base plan to a new one, we are to replace in the columns for these factors the upper level by the lower ones, and thus obtain six times the lowest level and three times the mean level [16]. /132

The "splitting" method is the replacement of a plan with a factor and a large number of levels by the plan with new factors and a smaller number of levels (Table 11.8). Here X_1^H is the four-level factor which is being "split" into three factors X_1^I, X_1^{II} and X_1^{III} at two levels each. The "reconstitution" method is the operation, inverse to "splitting" and it involves the replacement of the plan with the factors having smaller number of levels by the plan in which one of the factors X_1^H has a larger number of levels (Table 11.8).

TABLE 11.7.

i	X_0	X_1	X_2	X_3	X_4	Z_1	Z_2	Z_3	Z_4
1	+1	−1	−1	−1	−1	+1	+1	+1	+1
2	+1	0	−1	0	0	−2	+1	−2	−2
3	+1	+1	−1	+1	+1	+1	+1	+1	+1
4	+1	−1	0	0	+1	+1	−2	−2	+1
5	+1	0	0	+1	−1	−2	−2	+1	+1
6	+1	+1	0	−1	0	+1	−2	+1	−2
7	+1	−1	+1	+1	0	+1	+1	+1	−2
8	+1	0	+1	−1	+1	−2	+1	+1	+1
9	+1	+1	+1	0	−1	+1	+1	−2	+1

TABLE 11.8.

X_1^{I}	X_1^{II}	X_1^{III}		X_1[illegible]
0	0	0	⟷	0
1	0	1	⟷	1
0	1	1	⟷	2
1	1	0	⟷	3
1	1	1	⟷	0
0	1	0	⟷	1
1	0	0	⟷	2
0	0	1	⟷	3

Planning in the Presence of Qualitative Factors

If some or all factors are of qualitative nature (the types of assemblies and instruments, types of materials, various executive pieces of equipment, etc.), in other words, as one conducts the experiment in the presence of inhomogeneous conditions, when it is impossible to test all variants, it is reasonable to use the incomplete block planning.

If during the experiment, one studies within each block all components, then such plan will be called the full block type plan (for example, the full factorial experiment, FFE). The incomplete block plan makes it possible to reduce significantly the number of experiments or tests (because one investigates only some of the components within the block). Therefore, they always save time [16].

One of the types of incomplete plans is the plan of Latin square type and of cube type. /133

Latin square. What is being called the Latin square h X h is a square table, consisting of h elements (letters or numbers) each of which is encountered only once in each line and in each column [16]. Therefore, the planning using Latin squares has the following constraint: each source of inhomogeneity (the qualitative factor) must appear only once within a column and only once within each line. Because of such column and line effect elimination, the residual error is considerably smaller.

In planning an experiment, the lines and columns in the square are used to designate h levels and k factors. One therefore forms a 1/h replicate of the full factorial experiment of h^k type.

In order to be able to use the Latin square in planning an experiment or test, one must subject the standard square to the randomization procedure. In such case, the levels of factors are defined randomly for the columns, lines, rows and Latin letters, respectively.

The experimental results are presented as a linear model

$$Y_{ijk} = \mu + \alpha_i + \beta_j + \gamma_k + \epsilon_{ijk}, \qquad (11.14)$$

where Y_{ijk} is the experimental response obtained for the A factor at i level, of B factors at j level, and for C factor at k level, μ is the common effect, obtained during all tests, the general mean sum from which the sampling was taken, α_i is the effect of the line (of A factor); β_j is the effect of the column (of B factor) and γ_k is the effect of the square element which is the source of inhomogeneity (of C factor); ε_{ijk} is the remainder by the use of which one estimates the random experimental error.

The statistical conclusions will be valid if the square elements do not interact in terms of the lines, columns, etc.

The combinations of two orthogonal squares is being called the Latin square of second order. If the elements of the first square are designated by the Latin letters and of the second one - by Greek letters, such square is called the Greek-Latin square. Each element within the Greek-Latin square is encountered only once per each line and per each column. Under such conditions, in order to conduct each of the nine tests, the following conditions (Table 11.9) must be obeyed.

The experimental model will have the following form:

$$Y_{ijkl} = \mu + \alpha_i + \beta_j + \gamma_k + \delta_l + \epsilon_{ijkl}. \qquad (11.15)$$

Each Greek-Latin square is $1/h^2$ replicate of the full factorial experiment $n_{FFE}=3^4=81$ (for four factors at h=3). In such case, the plan which is based on the Greek-Latin square makes it possible to evaluate the factors by using only nine tests or experiments.

The three orthogonal squares form a Latin square of the third order, or the hyper-Greek-Latin square (corresponding to $1/h^3$ replicate of FFE). The mathematical model of such plan will have the following form:

$$Y_{ijklr} = \mu + \alpha_i + \beta_j + \gamma_k + \delta_l + \xi_r + \epsilon_{ijklr}. \qquad (11.16)$$

A \ B	B_1	B_2	B_3
A_1	$C_1\alpha$	$C_2\beta$	$C_3\gamma$
A_2	$C_2\gamma$	$C_3\alpha$	$C_1\beta$
A_3	$C_3\beta$	$C_1\gamma$	$C_2\alpha$

Greek-Latin square

TABLE 11.9. /134

Number of experiment	Factors (k=4, h=3)				Results
i	A	B	C	D	Y_i
1	A_1	B_1	C_1	α	Y_1
2	A_1	B_2	C_2	β	Y_2
3	A_1	B_3	C_3	γ	Y_3
4	A_2	B_1	C_2	γ	Y_4
5	A_2	B_2	C_3	α	Y_5
6	A_2	B_3	C_1	β	Y_6
7	A_3	B_1	C_3	β	Y_7
8	A_3	B_2	C_1	γ	Y_8
9	A_3	B_3	C_2	α	Y_9

The increase of order of the Latin square results in the increase of the minimal number of levels, making it possible to reduce the sampling of variants (of experiments) by a factor of several dozen. However, the reduction of the number of tests is always related to some information loss about the phenomenon which is being investigated and is justified only in the case when the linear models may be used and in the case of approximate estimate studies.

Latin cubes. On the basis of Latin squares, one can construct some other plans of different power. The Latin cube of $h \times h \times h$ size and of the first order is a cubic table, made of h elements, located at h^3 planes in such a way that each element is incorporated into this table h^2 just once, and is encountered within each of the planes (at any point of the table) a similar number of times, which is equal to h (h is the number of factor levels).

The Latin cube is essentially h number of balanced Latin squares of $h \times h$ size, and each square (or block) is correlated with the level of the fourth factor.

The Latin cubes form 1/h replicate from the full factorial experiment. The plans which are compiled on the basis of Latin cubes are regular and orthogonal. They can be easily processed by satistical methods of dispersion analysis.

The Latin cube of second order is a cubic positional table which is used to study the effect of factors having a different number of h levels. A more detailed presentation of the theory of Latin squares and of rectangles, of cubes and also of parallelepipeds, and the methods of their practical utilization in planning the multifactorial engineering experiments may be found in the study [16] and others.

CHAPTER 12.

PRINCIPLES OF AUTOMATION OF THE EXPERIMENTAL RESEARCH. MATHEMATICAL PROCESSING OF METHODS OF THE SPACECRAFT TEST DATA /134

The system of experimental research automation includes the following components: the experimental stand and means for simulated work, the means of control to conduct the experiments, the means to be able to process the experimental data and the means which reflect the obtained results.

Naturally, one should attempt, if it is possible, to automate /135
in real time the whole test process. The test automation provides the usage of a computer which is connected in a certain way with the object which is being tested. This connection is brought about by using the correlation device with the object, including a set of equipment and program means which provide the data gathering from the primary converting devices and its transfer to the computer. The control signals processed by the computer are transmitted to the executive organs of the experimental stand, with the parallel data transmission about the state of the product to the depiction device (curve plotter, display, etc.).

To transform the analog signal from the primary converting devices into a computer digital code, one is to use the analog-to-digital converter. The signal conversions from computer to the executive organs on the stand are realized by using the digital-to-analog converter. To control the automated test system, one can use a display which has a screen and a keyboard to depict either the letter-digital or graphic data.

In automating the experimental studies, one develops the experimental service program, which ensures the execution of the following operations: reception and recording of experimental data, control of the experimental stand, of the measuring equipment and the procedure which is involved in the experimental studies, the transformation of analog signals from the primary converters and introduction of the coded signals in the computing part of the system, processing of experimental data as per assigned algorithm and its intermediate storage, the primary data processing, the information exchange with the researcher, the processing and analysis of the information, the statistical data processing of a series of experiments, the diagnostics and prognostication of the future course of the tests, the arrangement and development of experimental results in the form which is applicable for the subsequent use.

The automation of experimental studies presupposes the presence of measuring and computing equipment, incorporating a computer and special subassemblies (Figure 12.1).

The computer must satisfy the following requirements: high speed, the availability of high speed communication channels, the availability of external memory units of required size, large operational memory, a sufficient number of processing levels for incoming data, flexible software, time-saving operational steps and functioning, etc.

Figure 12.1 shows the measuring and computing assembly which incorporates the following basic components: standard CAMAC, analog-to-digital converter, digital-to-analog converter and CM-4 computer with the external device. The modular design in the equipment of international standard CAMAC provides the informational logic compatibility of different functional modules. /136
It is based on the major communication line principle, in other words, all connections between the modules are realized via the data channel, which transfers the energy from the power source inside of the enclosure to all modules which are within the enclosure, exchanges the data between the operational models and control units and exchanges information with the computer.

The CAMAC modules, in terms of their functional purpose, are divided into five major groups:

input modules which include the analog-to-digital converter, synchronizers and the devices for reception of digital and analog data;

output modules which include the digital-to-analog converter, control printouts, perforated tapes, etc.;

the interface or connecting modules which allow to connect the data channels with the standard external devices;

module of signal processing which includes the high speed switches of analog signals, the amplifiers with adjustable gain coefficient, etc.;

modules for the conversion of binary codes into binary-decimal code, the devices for multiplication and division, devices to carry out the operations with a floating comma.

The computers which are used in the measuring-computing assemblies in the course of experimental study automation may be divided functionally into the three following types: specialized computers, the computers with time separation and the functional level distribution computers.

The specialized computers are designed to process the signals from one experimental stand. They are designed to execute specific operations.

The computer with time separation within the framework of automated experimental studies is a device of intermediate power and may service several experimental stands.

The functional level distribution computer, depending on its purpose, may operate specific levels, beginning from the lower one and upward (for example, the two-level functional distribution computer).

The computer incorporates the following devices: the device for data input and output, the operational control device for the whole system, the arithmetic-logic device, operational memory, external memory and the buffer memory. /137

In order to automate the tests, one designs the specialized systems which include the following programs: experiments, tests, comprehensive tuning, evaluation of the operational parameters in the system designed, the monitoring programs, the teaching and other programs, and also a number of subsystems. The communication between the subsystem and the test stand is involved in preparing the equipment to be functional, in launching and stopping the tests and ensuring the data exchange, with the transmission to the computer of the control commands and operational parameters.

The monitor control program makes it possible for the researcher not only to assign the experimental modes, but also if needed, to effect it in an active fashion, in other words, the operator is able during the experiment to follow through the functioning of the test devices, to monitor the measuring data and to actively participate in the research.

The design of automated systems to test the technical objects requires the development of a comprehensive and convenient to use software. The greatest complexity in testing such objects lies in the high requirements with respect to languages which are used to describe the tests. The language used must satisfy the following conditions:

the program written in this language must simultaneously be useful as an instruction manual to conduct the tests;

the language must have a jargon which embraces an extensive technical base;

the language must be able to describe all potential situations which may arise within the test system;

the language must be easily learned and not too complicated, so that the program could be written rapidly.

As a minimum requirement imposed on the language is the condition that it would enable the researcher to prepare the test program without calling upon the programmer.

The most acceptable, to generate the objective evaluation of parameters, are the automated test assemblies in which the participation of a human being during the test is almost completely excluded.

In a number of cases, a need arises to have secondary converters, designed to transform the controlled, nonelectric quantities, into electric quantities which are more convenient for use in the automated test assembly.

Such converters may be partially incorporated within the system and may partially be an integral part of the automated test stand. The data from such converters enters the automated test assembly for further processing.

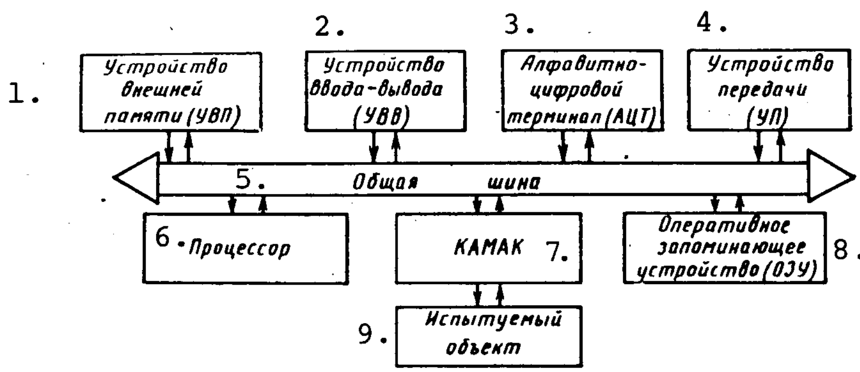

Figure 12.1. Structural diagram of the measuring-computing assembly, based on a minicomputer and CAMAC equipment.

Key: 1. External memory; 2. Device for data input-output; 3. Alphabet-digital terminal; 4. Device for data transmission; 5. Common busbar; 6. Processor; 7. CAMAC; 8. Operational memory device; 9. Object which is being tested.

As has already been mentioned, to simulate the external action on the object which is being tested, it is necessary to have special test stands (vibration test stands, impact stand, centrifugal forces stand, etc.) for which one also provides the remote control operation and adjustment. The control and adjust- /138
ment of the parameters on the stand could also be accomplished by the automated test assembly.

For each type of object one must compile its own test program. However, the structure of the automated test assembly itself may remain unaltered. Consequently, one can visualize a standard automated test stand which can be useful for tests of not just one but a whole series of similar objects.

The switching from tests of one type of products to another will be associated with the test program exchange which may be written on an easily replaceable data carrier, at the interface junction point between the test assembly and the object which is being tested.

The test programs may be written on any data carrier (perforated type, punch cards, magnetic tape, magnetic disks) in the operational memory of the computer. One could utilize either a specialized computer or a general purpose computer as a test control computer. In such case, the test assembly should incorporate the necessary number of converters. A structural diagram of a standard assembly, shown in Figure 12.2, explains the basic principles of its operation.

The data carrier which may be in the form of a perforated tape, magnetic tape, punch card, etc., and employing one of the discrete codes, may record the following information: the distinguishing features of the parameters, the nominal values of the parameters, permissible deviation of the control parameters, the values of mass coefficients and of the coefficients which define the relationship between the economic and esthetic characteristics of the object, compared to the standard values, the specific features of control signals, the levels of control signals, the addresses of communication lines between the object and test stand, the duration of tests or time when the test of a parameter commences and the date of testing.

By using the device of data input programming, the information enters the program control device. Here it is decoded and control commands are issued to all subassemblies within the major tests assembly.

The program which is written on the program carrier which is easily removable, contains information which in principle will change as one proceeds from the test of one type of object to tests of other types of objects. However, the sequence of operations which must be executed by the automated device during the control and monitoring of each parameter, is known a priori, it is a rigid system of operations for a specific set of types of parameters, and therefore, may be programmed within a specific internal, long-term memory device. For these purposes, a synchronizer may be used which in conjunction with the operation of the program control device, will provide the required operation of all subassemblies within the major assembly. The parameter data from the appropriate outputs enters, via a switch, the input of the converters of signal-code type. As a rule, each parameter has its own converter which converts the quantities in question into the form which is convenient for use in a computing device (for example, the d.c. current is converted into binary codes, the impedance is converted into binary code, etc.).

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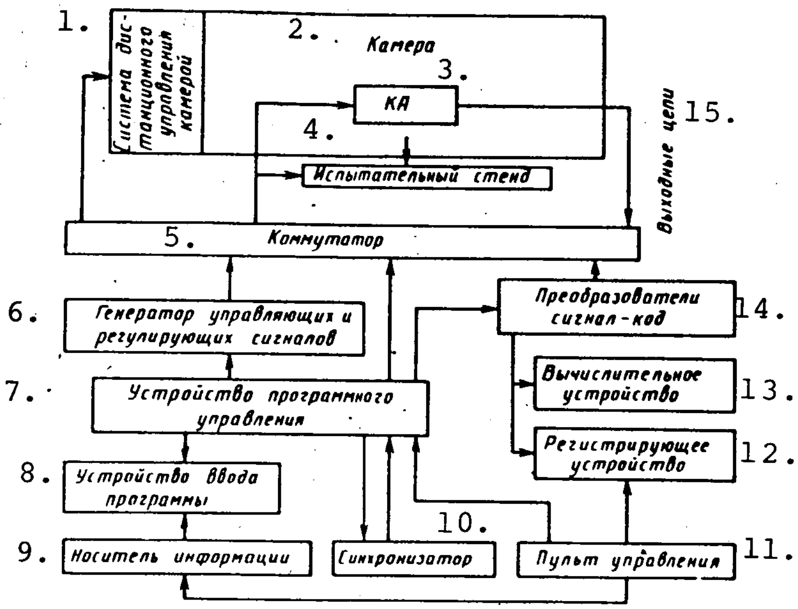

Figure 12.2. Structural diagram of a standard automated test assembly.

Key: 1. Chamber system of remote control; 2. Chamber; 3. Object being tested; 4. Test stand; 5. Switching busbar; 6. Signal control and adjustment oscillator; 7. Program control device; 8. Data input device; 9. Data carrier; 10. Synchronizer; 11. Control panel; 12. Recording device; 13. Computing device; 14. Signal-code converters; 15. Output circuits.

As a result of such tests, one defines all necessary information (the number of objects, the date of testing, etc.) which is fed from the program control unit.

By having the recorded test results, one can always establish which parameters during testing have failed.

The operational quality of the test assembly depends not so much on the equipment, but on the programs available in which one can clearly differentiate three levels: program for system operation, service programs and applied programs.

The operational system program includes a specific common operational system and one or several translators. In addition, it must be able to execute some service functions, facilitating the communication with the computing system.

The service programs are actually a part of the applied software programs which are available. Such programs enable the user to refer quickly and in a simple fashion to the operation of the computer which is being frequently repeated. For example, such program, on a brief recall, may order the transfer of data from one disk file into another.

The third and most widely used part of the software programming are the applied programs, while the first two are used merely to reduce the time expended and the means which are used to develop the applied programs.

It should be noted that the multiple variety of problems in automated testing, associated with the design of the appropriate flight program files, may be basically ensured by having the /140
following algorithms:

the algorithms of the primary processing of measured data (smoothing out, averaging-out, extrapolation and interpolation);

control algorithms (the stochastic approximation and other iteration procedures);

adaptation algorithms which are needed to handle the problems with parameters which change in time;

the computational algorithms of statistical parameters (spectral and correlation analysis, computations of the distribution laws);

algorithms which generate the determinant and random time-dependent sequences with a specific and controllable statistical parameters;

algorithms for identification of dynamic parameters of the test systems and of the objects which are being tested.

The flight programs are written in one of the computer languages - from the computer codes and ASSEMBLER language, all the way to the languages of higher levels FORTRAN, MINSK-22. There are also some special languages for narrow application which facilitate the program development of a specific class.

The effective automation of experimental studies depends to a large degree on the solution of a number of systemic questions, among which are: ease of functional operation of the automated system, the simplicity of readjustment, making it possible to track the experimental base, the development of an interaction system between the researcher and the object which is being tested by means of automation, and the ability to introduce new data into the system, etc.

Processing methods of test results. In processing the data, pertaining to an engineering experiment, one must adhere to the following recommendations: to reduce as much as possible the volume of intermediate calculations; to assess the necessary number of figures after the decimal point in the numerical data which is being used; to ensure that the calculations could be carried out by different operators; to be able not to use accurate and complex analytical methods if one could get by with approximate data; not to use extraneous numbers in the intermediate calculations; attempt to use not the manual but computerized calculations; utilize the rules for approximate calculations, and use abbreviated formulas.

The major sources of errors which affect the experimental accuracy are: limited sensitivity of the measuring instruments; the inability of the read-out part of the measuring instrument to reflect properly the response of its measuring element and the limitations of the researcher in reading the data properly which was obtained by the measuring device.

There are systematic and random errors. By systematic errors one should understand the errors which do not change (constant) during the experimental process, or which change according to a known law, in other words, the errors whose nature is predetermined.

The constant systematic errors retain their sign and magnitude during the whole measuring process and the variables tend to change (either increase or decrease). The mathematical processing of /141 the measured results cannot establish the detection of constant systematic errors. Analysis of such errors is possible only by having some a priori information about the errors or the information which can be generated while verifying the measuring instruments.

The exclusion of each systematic error from the measured results if taken separately requires at least one specially designed experiment. There are different physical quantities and a variety of methods of measuring them, and therefore, the methods for exclusion of systematic errors also are most varied. However, one frequently encounters the so-called traces of these errors which were not excluded. There is a limit below which such residual traces of systematic errors could be disregarded. One of such limits is a specific (for example, 5%) fraction of root-mean-square value in the random error.

By random errors in an experiment, one should understand the errors which by their nature and magitude cannot be defined, and which change without any applicable rules. Random errors are due to random factors and therefore, the totality of such factors which affect the measured results is impossible to assess, and consequently, it is impossible to compile any classification diagram.

The theory of random errors is based on three important premises which are rooted in the experiment: in having a large number of measurements, the random errors, equivalent in magnitude but different in sign, are to be encountered with the same frequency;

small errors are encountered more often that the large errors;

the random errors which are large are not encountered.

As has already been mentioned, the random errors are due to random factors and therefore, theoretically, one could utilize the distribution laws and theorems which the random quantities obey. In the engineering practice, the most widely used is the normal distribution - the normal Gauss law.

For the quantitative evaluation of random errors (of the accuracy of measurements) the most convenient is the root-mean-square error.

During the experimental process, one can encounter gross errors or misjudgements. The misjudgements may be due to the unexpected objective circumstances during the measurements and depend on a number of factors (calculations by using a faulty instrument, improper read-off from the instrumental dial, missing an observation, incomplete observations, improper handling by the researcher of the instrument, etc.).

To elucidate and remove such misjudgements, one could recommend two appraoches: the direct analysis of obtained results with thorough verification of well-fixed conditions, in which the experiment is being carried out, and the statistical methods.

The results of measurements which may incorporate such misjudgements must be discarded. The remaining random errors will define the accuracy of the measurements.

It is desirable that the experimental data processing would be carried out immediately after the experiment's termination. The newly generated data is processed in the same way as the previous set of data. However, one should, in addition, verify the data reliability (by comparing its logic principles), its fulness, the availability of written test information and quality of the execution of these tests, and also the proper correlation between former information and new data within the framework of the working /142 hypothesis (if there is a disagreement, one must develop a new hypothesis).

In the majority of cases, the engineering experimental studies are associated with solution of two types of problems.

1. The determination of connection between the variables which are being varied, while the object is being studied and the changes in one of the parameters (of the properties) of the same object which is being studied.

In essence this problem is reduced to finding the coefficients in the equation (11.2), necessary for the interpolation of the obtained results:

$$Y = f(X_1, X_2, X_3, \dots, X_n). \tag{12.1}$$

In handling such problems, one ordinarily uses the method of least squares, in which the requirement of optimum correlation between the curve (2.1) and experimental points is reduced to the rule that the sum of squares of deviations in the experimental points, from the curve which has been smoothed out, would be minimized:

$$\sum_{i=1}^{n} [y_i - f(x_i)]^2 = \min. \tag{12.2}$$

The method of least squares, when compared to the other methods of smoothing out operations, has the following advantages:

a comparatively simple mathematical process which is used to determine the coefficients in the equations (11.3);

a sufficiently valid theoretical rationale of using this method from the probabilistic point of view (such rationale is based on the normal distribution law in the measured errors and on the requirement of maximum probability for a given totality of errors).

Let us consider the utilization of splines in data processing, when the method of least squares is used. The spline S(t) of ℓ power and the defect d, associated with the multiple points X_1, X_2, ..., X_n ($X_1 < X_2 < \dots < X_n$) is such function which satisfies the following two conditions:

within each of the intervals $t \leq X_1, X_{i-1} \leq t \leq X_i$ (i=2, 3, ..., n), $X_n \leq t$, the S(t) function is the polynomial of ℓ power;

the S(t) function and its derivatives, all the way to (ℓ-d) order are continuous in the $-\infty < t < \infty$ region.

In practice, the most frequently used are the cubic splines of defect 1(ℓ=3, d=1). These splines are essentially the continuous curves, composed of cubic parabolas, and at the junction points $X_1, X_2, \ldots, X_n$ they will have continuous derivatives of the first and second order.

The simplest way of constructing a spline is a linear combination of the so-called B splines (or fundamental splines) $b_j(t)$, in other words, of the following form.

$$S(t) = \sum_{j=1}^{n+4} a_j b_j(t). \quad (12.3)$$ /143

But since one talks here not about the construction of a spline in general, but of a spline which smooths out the experimental points (t_i, y_i), i=1, 2, ..., m, $t_1 < t_2 < \ldots < t_m$, the a_j coefficients in (12.3) can be determined on the basis of the minimum of a sum

$$\sum_{i=1}^{m} [Y_i - S(t_i)]^2 = \min,$$

or, which is the same,

$$\sum_{i=1}^{m} [Y_i - \sum_{i=1}^{n+4} a_j b_j(t_i)]^2 = \min. \quad (12.4)$$

In addition, we shall assume that $X_1 > t_1$ and $X_n < t_m$.

The B spline (more precisely, the cubic B spline of defect 1) $b_j(t)$ is a spline with a multiple set of points $X_1, X_2, \ldots, X_n$ which is equal throughout to zero, except for the $X_{j-4} < t < X_j$ interval.

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Since the cubic polynomial, in a general case, is determined by four coefficients, four such polynomials will be defined by $4 \times 4 = 16$ coefficients. To find these sixteen coefficients, we will have 15 conditions (three conditions of the spline continuity, together with its first and second derivatives, at each of five nodal points X_{j-4}, X_{j-3}, X_{j-2}, X_{j-1} and X_j). It actually means that the B spline (12.3) is defined with the accuracy up to an arbitrary constant factor. It is easy to show that the B spline has the same sign in the $X_{j-4}<t<X_j$ region (ordinarily, it is assumed that it is positive) and it has one extermum. At $t=X_{j-4}$ and $t=X_j$ points, the B spline $b_j(t)$ is equal to zero, together with its first and second derivatives.

In order that the expression (12.4) would have meaning, one must introduce eight additional nodal points X_{-3}, X_{-2}, X_{-1}, X_0, X_{n+1}, X_{n+2}, X_{n+3}, X_{n+4}, which would satisfy the following conditions $X_{-3}<X_{-2}<X_{-1}<X_0 \leq t$; $t_m \leq X_{n+1}<X_{n+2}<X_{n+3}<X_{n+4}$.

One ordinarily assumes that $X_0=t_1$, $X_{n+1}=t_m$.

The conditions of minimum of the (12.4) function result in a system of normal equations

$$\sum_{i=1}^{m} [y_i - \sum_{j=1}^{n+4} a_j b_j(t_i)] b_k(t_i) = 0; \qquad (12.5)$$
$$k = 1, 2, \ldots, n+4.$$

Because of the specificity of B splines, this system has a two-dimensional matrix which makes the solution of the system (12.5) much easier.

In practice, the B splines $b_j(t)$ are determined ordinarily by recurrent formula

$$b_{kj}(t) = \frac{(t - x_{j-k}) b_{k-1\,j-1}(t) + (-t) b_{k-1,\,j}(t)}{X_j - X_{j-k}},$$
$$k = 2, 3, 4, \qquad (12.6)$$

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$$b_{1j}(t) = \begin{cases} \frac{1}{X_j - X_{j-1}}; & X_{j-1} \leqslant t < X_j. \end{cases}$$

where $b_{kj}(t)$ is the B spline of $\ell-1$ power, so that $b_{4j}(t)=b_j(t)$.

As has already been mentioned, the B spline $b_j(t)$ is determined with an accuracy up to an arbitrary constant factor. In the formula (12.6) this factor is selected so that the following condition would be satisfied

$$\int_{-\infty}^{\infty} b_{kj}(t)dt = (1/n)$$

for all j.

The question still remains open as to the number and location of the nodal points X_1, X_2, ..., X_n. In a general case, within the intervals where the quantity which is being investigated changes relatively rapidly, the nodal points which are selected are fairly densely packed, and vice versa.

The considered method of approximation curve construction in the form of a cubic spline may be generalized in the case of construction of the approximating surface in the form of a bicubic spline.

2. Determination of the optimal operational conditions of the object being investigated, in other words, the provision of extremal value of the optimization parameter.

The utilization of multifactorial experimental planning creates favorable conditions in searching for and finding the optimum region by employing the method of "steep ascent" along the plane of the response function obtained.

The essence of this method involves the sequential or, as one ordinarily calls it, "step" movement toward the optimal value of the function, in the direction of the linear approximation gradient. By the latter one should understand only the accounting of the first four summands in the equation (11.9). The search is continued until the moment when the response function will fall within the optimal region. Further, in order to find this region, it is recommended to conduct an additional series of experiments if possible, and to refine the values of the optimal factors if possible.

Movement toward the gradient is the shortest or the fastest way toward the optimal region (the extermum of the optimization parameter). The partial derivatives of the equation (11.2) with respect to the factors which are being varied, correspond in terms of magnitude and sign to the coefficients of regression. Consequently, if one changes the parameters proportionally to the respective coefficients of regression, by increasing or decreasing them in accordance with the sign in front of the coefficients, the movement will be along the gradient [16].

If within the range of the factor variation there is no optimum, it is necessary, depending on the target of investigation, either to expand the variation range and continue the studies, or to limit oneself to the obtained information that within this range there is no optimum. In realizing the optimization process, by using a selected parameter, one should critically estimate the possibility of the deterioration of the other parameters, applicable to the studied subject or object.

The other optimization method is graphical, and although it is less accurate than the anlytical, it is self-apparent and simple.

At the end of an experiment, the researcher obtains data which /145 characterizes the relationship between the factors which are being varied. The end data may be in the form of either tables or graphs. Each of these methods has its own advantages and disadvantages. Let us note here the disadvantages in both methods: the tables and graphs must be stored if they are to be used in future studies;

Let us note here the disadvantages in both methods:

the tables and graphs must be stored if they are to be used in future studies;

if the data is to be used by several individuals, it must be xeroxed;

tables and graphs are size-limited.

The advantages of tables are: precise correlation between the measured results, the ability to record simultaneously the change of several variables. In graphic data processing, such accuracy is practically not attainable.

Among the advantages of the graphic method is its self-apparency, defining the nature of the relationship between the variables, the possibility of being able to determine the intermediate values, etc.

The third method of research data interpretation by using the equations (or formulas) is free of the above-mentioned drawbacks because the equations themsleves represent the relationship between the variables in a convenient and clear form, which is easy to recognize and to record. That is the reason why the scientific and research study may be viewed as a finished one only in the case when the relationship which is found between the factors of interest is described by a scientifically based equation or by an empirical formula.

The transformation of research data into the form of equations and formulas (the development of mathematical models) may proceed along two avenues:

the processes which are being investigated occur in accordance with the known theoretical laws and therefore the formula shape is known to the researcher and all he has to do is to determine the constants which are incorporated into this formula;

in the majority of cases the researcher has to work with unknown relationships and therefore his goal is first of all, to determine the type of relationship on the basis of experimental data and then to determine its constants.

There are still no methods at this time which would make it possible to obtain directly, on the basis of the generated data, the type of empirical formula. Of greater and greater importance in handling this problem are the computers. Under ordinary conditions, this is the method of proper sampling and selection which requires a large time expenditure and numerous cumbersome calculations. The problem of the formula shape determination is significantly simplified if one utilizes the graphic representation.

The essence of graphic methods of experimental data processing is in that while constructing the curves by using such data, it is necessary to select such scales on the coordinate axes (log, semi-log, quadratic, root-mean-square, inverse, etc.) that one would obtain the straight lines or lines which are close to straight ones.

The construction of curves in having two variables ordinarily creates no problems. The situation is more complicated if one has three variables:

$$z = aX + by + c = f(x, y). \qquad (12.7)$$

In such case, one assigns the value to one of the variables, /146
substitutes it into the equation and then one constructs the curve for two variables. By changing the value of the third variable, one obtains a family of curves. Then, one handles analogously the second variable and obtains a new family of curves. The comparison of obtained families of curves makes it possible to draw some conclusions:

if the family of curves describing both variables consists only of straight lines, such shape of the equation is defined quite precisely: both variables are incorporated into the equation of interest only in terms of their first power, and are to be found only in the numerator;

if the family of curves with respect to one variable consists of straight lines and with respect to the other variable - of a number of curves, the type of equation is not always possible to determine;

if the family of curves describing both variables consists of the curves rather than straight lines, it is difficult to determine precisely the type of equation: either both variables are incorporated into the equation of interest in some power which is different from unity or it is of the first power but is to be found in the denominator;

if the relationship between two or three variables turns out to be linear, then the graphic processing of experimental data can be terminated;

if, however, one obtains for these relationships a smooth curve or a family of smooth curves (describing one or both variables) one must continue with the graphic processing of the experimental data, constructing the curves in rectangular coordinates and nonuniform scales. This approach enables us to obtain instead of curvilinear descriptions, the rectilinear description, making it possible to define more accurately the type of equation. If however, it is not possible to "straighten out" the curves, what remains is only the method of sequential sampling and selection of the equation type and verification in terms of its coefficient of correlation.

As it appears, the cases are not excluded when, after a number of selected samplings which failed, the researcher must drop any further tests, trying to find a sufficiently convenient for practical purposes empirical formula for the whole range of the change in variables. Under such circumstances, it is more reasonable to find the equations which would characterize the specific segments on the curves, but in such case, one must indicate the limits within which the variables in each equation may vary.

CHAPTER 13.

PROVISION OF RELIABILITY IN EXPERIMENTAL ASSEMBLIES AND STANDS FOR SPACECRAFT TESTING /146

In spite of the complexity, the problem of determining and controlling the reliability of test stand and assemblies (TSA) may be solved by applying the method of general reliability theory, by using some specific theoretical studies and practical methodologies, involved in the reliability improvement, accumulated in the other areas of machine industry. However, the specificity of TSA appli- /147 cation requires a different approach to handle the problems associated with the TSA reliability.

Major Reliability Indicators in The Test Stand Assemblies

In accordance with the classification features, the test stands and assemblies (TSA) are the singular manufactured objects which are being used with the appropriate servicing and repairs, attaining the limiting operational state of equipment with multiple use. For such operational objects, it is recommended to use the integral reliability factors: K_{tu} which is the coefficient of technical use and T_r which is the designated resource or object, which includes the parameters, characterizing its fault-free operation while in use and the TSA adaptability, so that the failures can be easily detected and properly handled.

The designated resource or object T_r is defined on the basis of safe and economic operation, employing the minimal mean time-to-first failure parameter, in one of the components of the TSA system.

The repair factors are computed by using the following formulas:

a) coefficient of operational readiness

$$K_o = \frac{T_f}{T_f+T_r}, \qquad (13.1)$$

where T_f is the time-to-first failure, T_r is the mean recovery or repair time ($T_r = \frac{1}{m}\sum_{i=1}^{m} t_i$), m is the number of failures and t_i is the time which is to be spent to remove the i failure.

The coefficient of operational readiness (13.1) characterizes simultaneously also the failure-free operation which substantiates the increase in TSA reliability via the increase in the possibility of repair;

b) mean summary expenditures, associated with the technical servicing

$$C_{t.s.} = \sum_{i=1}^{\Pi_u} \sum_{j=1}^{m_0} C_{t.s.\,ij}\,, \tag{13.2}$$

where $C_{t.s.ij}$ are the mean expenditures, associated with the j type of technical servicing of the stand during the i year of use, in thousands of rubles; m_0 is the number of types of technical servicing of the stand; Π_u is the time period of the stand use, in years.

Let us note that the evaluation of the TSA reliability factors in the case when the systems are already in use are being conducted experimentally, on the basis of the initial data. To evaluate the reliability level of the test stand during its design is much more complicated. To solve this problem it is necessary to split a complex technical system (TSA) into simple components (elements) and then to determine analytically the parameters of fail-free operation, ease of repair, etc. For example, in the case of non-repairable components, the major reliability factor is the mean time-to-first failure period T, which is measured in units of time. For the blocks and components which are the standby units or for a set of such blocks and components, the ease of repair factor will be the mean time of component exchange $T_e = \frac{1}{n}\sum_{i=1}^{n} t_e$, /148

where t_{ei} is the time required to remove the faulty component (or a block), to replace and adjust it, in hours, n is the number of such replacements. In addition to t_e, one must determine for such components a specifically designated object or resource which is to be replaced T_d or the time of standard maintenance T_k. The analysis of primary data indicates that the change in the state of the components is due in the majority of cases to the wear phenomenon, making it possible to use the normal time distribution law of their fault-free operation.

The parameter of the set of failures ω in the test stand is determined by using the following expression

$$\omega = \sum_{i=1}^{N} \lambda_i = \sum_{i=1}^{N} \frac{r_i}{T_i + T_{rel\,i}}, \tag{13.3}$$

where T_i is the mean time-to-first failure of the i reliable component, in hours; r_i is the number of components of the same kind, $T_{rel\,i}$ is the time during which the stand remains operational after the i component fails, in hours.

The rate of recovery (of replacement) μ_i of each of the components which failed, may be calculated by using the following formula $\mu_i=(1/T_{ei})$, where T_{ei} is the mean replacement time of the i element, in hours. The mean rate of replacement for a group of reliable components will have the following form

$\mu=\frac{1}{N}\sum_{i=1}^{N}\mu_i$, where N is the number of components within a group.

On the basis of relationships in regard to ω and μ and using the theory of batch servicing, in calculating the probability of fault-free operation of technical systems which utilize the repair or recovery parameter, the following expression may be used

$$P(t_n)=\frac{\omega \exp[-(\omega+\mu)t_n]+\mu}{\omega+\mu}, \tag{13.4}$$

where $P(t_n)$ is the probability of fault-free operation of the repaired systems.

The coefficient of operational readiness of the stand, taking into account the recovery parameter, is determined as follows:

$$\begin{aligned} K_{\ddot{O}}&=\frac{T_p(t_n)}{t_n}=\frac{1}{t_n}\int_0^{t_n} P(t_n)dt=\frac{1}{t_n}\int_0^{t_n}\left[\frac{\mu}{\omega+\mu}+\frac{\omega}{\omega+\mu}e^{-(\omega+\mu)t_n}\right]dt; \\ K_{\ddot{O}}&=\frac{\mu}{\omega+\mu}+\frac{\omega}{(\omega+\mu)^2}\left[1-e^{-(\omega+\mu)t_n}\right]\frac{1}{t_n}, \end{aligned} \tag{13.5}$$

where t_n is the time period during which the calculations are accomplished, of the fault-free operation probability of the stand, in hours, and T_p is the accrued operational time of the stand during t_n period, in hours.

The coefficient of technical use K_{tu} is evaluated for a specific calendar time period, for example, for one year. Let us assume conditionally that the operational use and current repairs are being conducted during 360 calendar days.

Then the total expression for K_{tu} will be /149

$$K_{tu}=\frac{T_{sum}-(T_{pl}-T_{npl})}{T_{sum}},$$

where T_{sum} is the summary time of operation and repair of the stand per year, in hours; $T_{sum} \Sigma 360 \cdot 24$ h; T_{pl} is the summary time intervals provided to complete the planned repairs and technical servicing, in hours, and T_{npl} is the nonplanned idle time, in hours.

The T_{pl} is calculated on the basis of data from the operational repair diagram of the stand (of the assembly):

$$T_{pl}=\sum_{i=1}^{n}(t'_i+t''_i)=nt''+\sum_{i=1}^{n}t'_n, \tag{13.6}$$

where n is the number of planned stoppages during a specific time period, t_i is the time interval of the i planned replacement of a group of standby units and t''_i is the time needed for the stand stoppage and i planned component replacement.

The T_{npl} is calculated by taking into account the probability that a technical system will fail, using the following formula $T_{npl}=\sum_{i=1}^{n} Q_i\tau_m$ where n is the number of planned operational time during a specific time period, Q_i is the probability that the stand will fail at the end of the i period of planned use, τ_m is the mean idle time of the stand or assembly as one conditional standby component, from all components, will be replaced:

$$\tau_m=\frac{1}{k}\sum_{j=1}^{k}(t'_j+t''_j)=1+\frac{1}{k}\sum_{j=1}^{k}t'_j,$$

where k is the number of standby units which are taken into account in calculating the reliability of assembly, t'_j is the time needed to replace the j standby unit, t''_j is the time required for repair and back to the operational mode of the system.

By taking into account the expressions for T_{pl} and T_{npl}, we will obtain

$$k_{t.r.} = \frac{T_{sum} - [nt'' + \sum_{i=1}^{n} (t_i' + Q_i \tau_m)]}{T_{sum}}. \tag{13.7}$$

<u>Requirements Imposed Upon Reliability of Test Stands and Assemblies</u>

The requirements which are imposed upon the newly developed test stands are the organizing initial step in the study of reliability, with the provision of continuity at all stages of TSA work cycle . The most important stage in handling the reliability problems is the stage of the technical project development, because at that stage, one establishes within the design the required properties, in other words one provides for a specific reliability level, with the prognostication in regard to all subsequent stages. The basic principle in compiling a set of requirements in regard to reliability is to formulate them at the preceding stage and to resolve them at the subsequent one.

Technical assignment (TA). The TSA requirements of reliability at this stage are being compiled, so as to fulfill these requirements while the technical project is being developed. This assignment includes the instructions of general nature, as well as of some specifics, but only for a given test assembly, namely: /150

the development, at the stage of technical projects of the stand block design, with the utilization of the appropriate mechanization to provide rapid exchange of blocks;

the appropriate level of using the unified and standardized block components which have high fault-free operation parameters and ease of repair, which limits the variety of standardized equipment and components;

the appropriate level of ease of repair of a given assembly with the provision of safety features for the personnel, in accordance with the existing standards;

the appropriate measures to alleviate the design assembly vibrations and to limit the noise level;

the time periods of normal use which must be ensured while designing the assembly and calculating its operational lifetime;

a number of reliability tests at the stage of the project technical development;

periodic analysis of reliability control level in the assembly which is being designed;

the major requirements on reliability at the stage of design work development and during the operational use;

the major requirements on reliability which may be realized at the manufacturing stage;

the development of the stand technical diagnostic system which can be used during utilization, with the indication of the number of components which are to be diagnosed and the types of generated data.

Technical project (TP). At this stage, one obtains the TA requirements, with respect to the assembly reliability and also one advances a number of new requirements with respect to the reliability which may be satisfied only at the subsequent stages. These requirements are included into the technical operational conditions (TC) while designing and using TSA. Let us mention the major ones:

refinements of requirements on reliability at the stages of the assembly manufacture (AM) and of its use (AU);

the instructions development at the stage of "work drawings" (WD) in regard to the input control and testing of the components during AM;

additional final touches and more refinements of design for AU;

fulfillment of the requirements in regard to the stand ease of repair;

standardized and regular reliability control level for the assembly at the WD and AM stages;

development of technological guarantees in terms of reliability and the appropriate conditions for achieving it;

development of requirements in regard to the accuracy of construction and amount of testing during the tuneup stages in order to provide and verify the projected levels of reliability in the assembly.

Work drawings. At this stage one finalizes all requirements with respect to reliability, developed during the design work and one formulates the instructions for various stages of TSA manufacture and utilization. The reliability requirements which are to be attained at this stage are included into the appropriate documentation file which is being sent to the user. The major documents are: /151

theoretical preparation and practical training of the servicing and repair personnel in the domain of the test assembly reliability;

the secondary control and a number of special tests in regard to the reliability of components at AM stage;

the reliability factory tests of the assemblies, of the blocks and components of the system, its assembly and the results of such tests;

the order in which repair work is to be conducted, with the appropriate corrections, associated with the accumulated experience in service and repair, with the upgraded parameters in regard to specific components, etc.

Table 13.1 below shows the standard procedures which are to be fulfilled to provide reliability testing of the stands and of assemblies.

TABLE 13.1. LIST OF MAJOR WORKS TO PROVIDE FOR THE PROPER RELIABILITY TESTING OF STANDS AND ASSEMBLIES (TSA)

Viable operational cycle	List of work which is involved	Specificity of work at a given stage
Technical assignment (TA)	1. Development of requirements for TSA	1. Analysis of starting data and evaluation of possibility to attain the level of this data at the stand beind designed
	2. Compiling the operational plan to ensure the proper conditions and amount of testing of components and blocks with respect to the reliability at the TP stage	2. Operational plan to provide the lacking technological data by gathering the information about either already existing TSA or by using a preliminary experiment
	3. Compiling the operational plan in regard to the assembly and processing of primary data as to the reliability of blocks and components	3. Selection of an analogue (or a prototype) of the assembly which is being designed

(Table 13.1 is continued on the following page)

TABLE 13.1 (continued)

Viable operational cycle	List of work which is involved	Specificity of work at a given stage
Technical proposal	1. Analysis of requirements with respect to reliability, taking into account the operational levels, attained at the prototype assemblies. Compiling of an operational plan in the refinement work, bringing it to the point of the technical project stage.	1. Development of the test assembly and components, as designed
	2. Development of general requirements of the component parts	2. Preliminary development of technological schematic 3. Evaluation of assigned parameters with respect to the reliability level of the assembly
Technical project	1. Construction of the structural reliability diagram for the assembly which is being designed	1. Attainment of the assigned reliability level
	2. Fulfillment of the requirements with respect to reliability and its evaluation at the TP stage. Development or requirements with respect to WD, AM and AU stages	2. Technical and economic analysis of the reliability level in the system /152
	3. Development of technical conditions at the input control and final refinement of the component parts within the design	3. Refinement of the repair and operational parameters in the assembly on the basis of maximum use of the standardized instructions and decrease in the variety of component parts

(Table 13.1 is continued on the following page)

TABLE 13.1 (continued)

Viable operational cycle	List of work which is involved	Specificity of work at a given stage
	4. Development of the repair chart with the appropriate technical documentation	
	5. Calculations with regard to the standardized backup equipment which is to be used as replacement	
Work drawings (WD), preparatory steps for the production and manufacture (AM)	1. Realization of requirements with respect to reliability, as developed at the TP stage	1. Training of the personnel in the operational specifics to provide the reliability in test assembly
	2. Technolgoical provisions of the reliability requirements with respect to the system	2. Analysis of results of the factory tests in regard to the reliability of components and blocks of the assembly
	3. Development of technological documentation to ensure the assembly reliability during the manufacture and factory testing	3. List of requirements imposed upon the standard operational conditions of the assembly, on the system of data gathering and on the failures and repair and maintenance parameters
	4. To conduct the tests and compile the documentation(filling the appropriate forms)	
	5. Refinement of the repair flow chart	
	6. Development of the technical and operational documentation	
	7. Formulation of the Assigned Functional Instructions	

(Table 13.1 is continued on the following page)

TABLE 13.1 (continued)

Viable operational cycle	List of work which is involved	Specificity of work at a given stage
Assembly, tune-up and utilization (AU)	1. Strict adherence to the requirements of the assembly meticulousness. The proper filling of all forms in accordance with the technical documentation of the manufacturing enterprise	1. Accounting and provision of the standby equipment for replacement in the test assembly
	2. Fulfillment of the requirements on use and repair of the assembly 3. Training of the servicing personnel and of the maintenance personnel 4. Accounting, analysis and reports of the manufacturing enterprise about the failures	2. Correcton of the repair flow chart in the assembly which is done jointly with the manufacturing enterprise, as the operational experience is being accumulated /153

<u>Sources of Primary Information In Regard to the Reliability Parameters</u>

The major sources of primary information in regard to the fault-free and ease of repair parameters in the components (blocks) of the structural reliability flow chart are:

evaluation tests on special stands in the conditions which are similar to the operational conditions (for the new designs);

information in regard to the amount of required work, obtained from the appropriate documentation related to the maintenance work;

information, derived from the repair and operational documentation;

survey of the experts (design developers, operators, maintenance men) with the utilization of the method of expert opinion.

The regular use of information, derived from the operational documentation is sometimes made more complicated by incomplete accounting of data associated with the operational experience and the repair of assembly. The survey of experts with the subsequent expert opinion may supplement to a certain degree the absence of such objective data.

In order to obtain the necessary information, one selects the objects of study which are prototypes (or analogues) of the test assembly which is to be evaluated (or of its block units). This selection is based on the similarity in technical parameters and in the construction design of the blocks and components. The preliminary analysis of the obtained data, as to the block and component reliability is being carried out in order to verify the quality of recorded data and exclude from processing the unreliable information.

A further processing of the initial data includes, first of all, the calculation of the mean time-to-first failure for a component (or a block unit) and the root-mean-square deviation from the data, used from the appropriate documentation.

The calculations are being carried out using the following formulas

$$T_q = \frac{1}{n_q} \sum_{i=1}^{n_q} t_i K'_э; \quad \sigma_q = \left[\frac{\sum_{1}^{n_q} (K'_э t_i - T_q)^2}{n_q - 1}\right]^{1/2}, \tag{13.8}$$

where T_q is the mean time-to-first failure of the block unit (or an element) on the basis of the data, derived from documentation, in 24-hour time periods; t_i is the calendar time-to-first failure for a given block or component (in 24-hour time periods; K'_e is the calendar time utilization factor; n_q is the amount of utilized /154
data and σ_q is the root-mean-square deviation in the time-to-first failure for a block unit or a component, in 24 hours.

The calculation of time-to-first failure of a block (of an element) and its root-mean-square deviation, on the basis of expert opinion, are being carried out by means of the following formulas

$$T_e = \frac{\sum_{i=1}^{n_e} [C_i t_i]}{\sum_{i=1}^{n_e} C_i} K'_э; \quad \sigma_e = \left\{\frac{\sum_{i=1}^{n} [C_i (K'_e t'_i - T_e)^2}{\sum_{i=1}^{n_z} C_i - 1}\right\}^{1/2}, \tag{13.9}$$

where T_e is the mean expert estimate, in 24 hours, σ_e is the root-mean-square deviation of the time-to-first failure in the block unit or an element (in 24 hours); i is the response code of the expert, t_i' is the calendar time-to-first failure of the expert opinion in regard to a block unit or an element, in 24 hours; c_i is the weight coefficient of the expert response.

The generalization of obtained results in regard to all data (in other words, on the basis of documentation and on the basis of expert opinion) is being conducted in the following manner:

$$\left.\begin{aligned} T &= \frac{\sum_{i=1}^{n_q} t_i + \sum_{i=1}^{n_e} [C_i t_i]}{m} K_e'; \quad m = n_q + \sum_{i=1}^{n_e} C_i; \\ \sigma &= \left\{ \frac{\sum_{i=1}^{n_q} [(K_e' t_i - T)^2] + \sum_{i=1}^{n_e} [C_i (K_e' t_i - T)^2]}{m-1} \right\}^{1/2}, \end{aligned}\right\} \quad (13.10)$$

where m is the quantity of iniial data which was taken into account; T is the mean value of time-to-first failure for a block unit or an element, obtained on the basis of all available data, in 24 hours; σ is the root-mean-square deviation of the time-to-first failure T, in 24 hours.

To calculate the mean time of the component (block unit) replacement, one may utilize the following formula

$$T_r = \frac{\sum_{i=1}^{n_e} t_{ri} C_i}{\sum_{i=1}^{n_e} C_i}, \quad (13.11)$$

where T_r is average time (removal, replacement and adjustment) required for the replacement of the block unit or an element (in hours); t_{ri} is the numerical time of the i block unit or an element replacement, in hours; n_e is the number of expert responses.

The processing includes the data about similar block units (components) which in terms of operational conditions and design are similar. On the basis of such expert opinion survey, one can estimate the time, necessary for the component replacement, the limiting values of the parameter which defines the operational viability of the component, the possibility of replacing a component (or a block unit) without stopping tests, and the required time and number of workers who will be involved in the course of such work.

The possibility of replacement of the block units (of a component) during the tests is defined by the accessibility to the block units (to the component) prior to its removal, the ability to attach and adjust it, while adhering to all standards and requirements of safety /155
and the continuity of technological test process during the component or block unit removal and replacement.

Ordinarily, the time required for the block unit, or a component replacement is assumed to be equal to the least time, obtained as a result of the initial data processing.

Repair Cycle Structure of the Stands and Test Assemblies

In developing a new test assembly, one must solve the questions related to the reliability, including the following:

development and incorporation into the whole process of the assembly delivery, the nonstandard means of repair and control;

a high degree of components (block units) repairability;

development and design of the technical diagnostic systems for the block units (for the elements) during the operation of assembly;

development of small scale mechanization means to conduct the repairs and maintenance;

provision that the personnel has at its disposal the technical reference data, related to the components which are to be replaced and compiling of scheduling for the planned maintenance repairs (TPR) taking into account the block units (components) replacement if and when the life term of such components is ended (T_d).

The essence of technical diagnostics consists in obtaining the information about the dynamics of changes in the operational viability of the mechanisms in the test assembly, as a function of the wear, associated with friction, corrosion, erosion, deformation, etc., as a function of the applied loads which were above the permissible ones and as a function of changes in the operational functining of the mechanisms which were functioning beyond the limit, as provided by the design.

The technical diagnostics make it possible to evaluate the general state of the critical components in the assembly on the basis of objective characteristics, obtained by using special measuring devices. The measurement of parameters which characterize the technical state is being conducted either continuously, by using stationary instruments, or periodically, by using the portable measuring means (for example, the attachable thermocouples, vibration meters, etc.).

In a general form, the dynamics of the technical state replacement of any of the blocks (or elements) on the basis of measurements of one or several parameters which objectively characterize its state, may be represented as a curve (Figure 13.1). The intersection point of the curve which describes the technical state of the assembly, with the line of limiting operational level, defines the permissible time for the utilization of a given component.

Under normal operational conditions, one must adhere to the following inequality $T_{lim} \geq T_k$ (in other words, in defining the corrected figure of the component in question, one must take into account the data derived by technical diagnostics).

In order to accomplish the repair and testing of the blocks (of components) at a high technological level, under the conditions in which a given assembly is being used, in putting such an assembly together, one must provide specially designed, for these purposes, /156 testing means and the appropriate control equipment.

The diagnostic equipment and testing of block units and components must be conducted in regard to such components in which 3σ is larger than the minimal value for a given component. It is also recommended to subject to diagnostics the components, the failure of which would represent a danger for the servicing personnel, or which might result either in correlated failure of the other components or to great damage in the case of such component functioning in the mode which is beyond the permissible operational levels.

In having the specific numerical values for specific components and knowing the amount of additional work which must be done to service the test assembly and undertake the work related to the technical diagnostics of the assembly as a whole, one can compile the annual and monthly schedule to conduct such work and a list of failures which occur.

Evaluation of the scope parameter for the component which is to be replaced. Let us evaluate the minimally necessary and sufficient scope and number of the components which are to be replaced which is a sum of operational units in the storehouse (in other words, the units which are capable to reestablish the operational functioning) and also the available reserves, inherent in the assembly.

The replacement by spare components will be differentiated in terms of two types of TSA elements: the first type are elements or components which cannot be repaired and the second type - those which can be repaired. All components inside of each of these two types are systematically compiled in groups. One group combines the components, the functional viability of which differs by not more than 20%.

The determination of the scope of the replaceable components annually, for the i group of components of the first type, are being conducted by using the following formula

$$Q_{ra_i} = \frac{Q'_{ci}}{t'_{MRW}} b, \tag{13.12}$$

where Q_{rai} is the number of replaceable components from storage, necessary per year, number of units; t'_{MRW} is the functional viability of the assembly before the first upgrading or repair term (MRW), per year; b is the coefficient which, for the first year of use, is assumed to be equal to 1.1, for the last term - 0.9, and for the remaining time - 1.0; Q'_{ci} is the number of components from i group which are to be replaced during the time period before the first MRW, number of pieces. /157

The necessary number of components in the i group of the second type, which are to be replaced, are determined on the basis of the following expression:

$$Q''_{ci} = \left(\frac{t_{MRW}}{T_{wo\,i}} K_i + \frac{T_{ri}}{T_{di}}\right) n_i, \tag{13.13}$$

where T_{ri} is the average time to repair the component of the second type in the i group, which can be accomplished outside of the assembly, in hours; T_{di} is the average operational lifetime of the components of i group, in hours; t_{MRW} is the lifetime before the first MRW of the assembly, in hours; n_i is the number of components in i group within the assembly; K_i is the correction factor which depends on the category of the component reliability, and on their number in the i group.

The T_{ri} value includes all time periods from the moment of disassembly to the moment when the item has been repaired and stored. As one can see, the T_{ri} value takes into account the average time to transport the component after repair both ways, to and from the repair shop.

The calculations by using formula (13.13) are conducted for such groups of components which satisfy the following inequalities:

$$T_{wo_i} \leq t_{MRW} \text{ and } T_{ri} \geq 0.7 \cdot T_{di},$$

where T_{wo} is the time before the component of i group is being written off, in hours and the n_i, K_i and T_{di} designations are analogous to those used earlier, but in this case, apply to the components of the second type.

If these conditions are not satisfied, the calculations of available reserves are conducted by using the following formulas:

when $T_{wo_i} > t_{MRW}$ and $T_{ri} \geq 0.7 T_{di}$

$$Q''_{ci} = (K_i + \frac{T_{ri}}{T_{di}}) n_i; \tag{13.14}$$

when $T_{wo_i} > t_{MRW}$ and $T_{ri} < 0.7\, T_{di}$

$$Q''_{ci} = K_i n_i. \tag{13.15}$$

As a rule, the total volume of replaceable components of second type are distributed unevenly in the course of years and the determination of annual figure depends on the mean functional operation time for a given component or block unit within the i group. In addition, a large fraction of replacements by spare parts happens during the first year. In conjunction with this, the calculation of the replaceable volume reserves during the first year is conducted by using the following formula

$$Q''_{ra_i} = \frac{Q''_{ci}}{C_i}, \tag{13.16}$$

where $C_i = T_{di}/L$, and the $C_i < 1$ values are rounded off to whole numbers; L is the average time of the assembly operation during the year, in hours; Q''_{ra_i} is rounded off to a whole number by dropping off the <0.3 values and adding one when the values are ≥ 0.3.

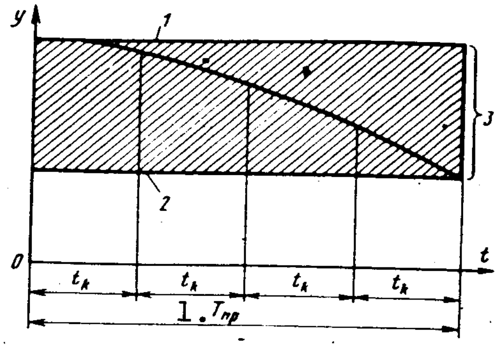

Figure 13.1. Dynamics of the change of technological state in the block unit or an element:

y is the parameter which defines the technological state of the block unit or a component; T_{lim} is the limiting permissible time period of utilization; t_k are the time intervals, between which the technical diagnostics is to be conducted; 1 is the initial level; 2 is the limiting permissible level and 3 is the area of permissible technical state scattering of data.

Key: 1. T_{lim}

Such approach in developing the TSA structural repair cycle (RC) makes it possible to: /158

conduct all repair work at the a priori planned and uniform time intervals which are multiples of the smallest time periods for a given component;

replacements of components (of the block units) the operational life term of which has been exhausted after the last repair;

to conduct the high grade repairs, upgrading the components (the fundamentals of such repair are the component replacement or repair and testing of these components in the specialized workshops or at the manufacturing enterprise);

calculation of the optimal annual volume of the stored items which are to be replaced (of the block units or components);

to gather the information which would indicate the causes of failures, with the feedback information to the manufacturing enterprise;

the RC correction or adaptation as the item is being altered, the increase of technical functional level of the components or block units in the case of sudden failures.

Figure 13.2 shows the structural diagram of all work at all stages of TSA design and operation, providing a high degree of reliability and upgraded efficiency of use.

The problem of providing reliability in the test stands and assemblies is a complex and multipurpose task. In considering the TSA as the assemblies to test the reliability and to analyze the relationships associated with the failures of specific components

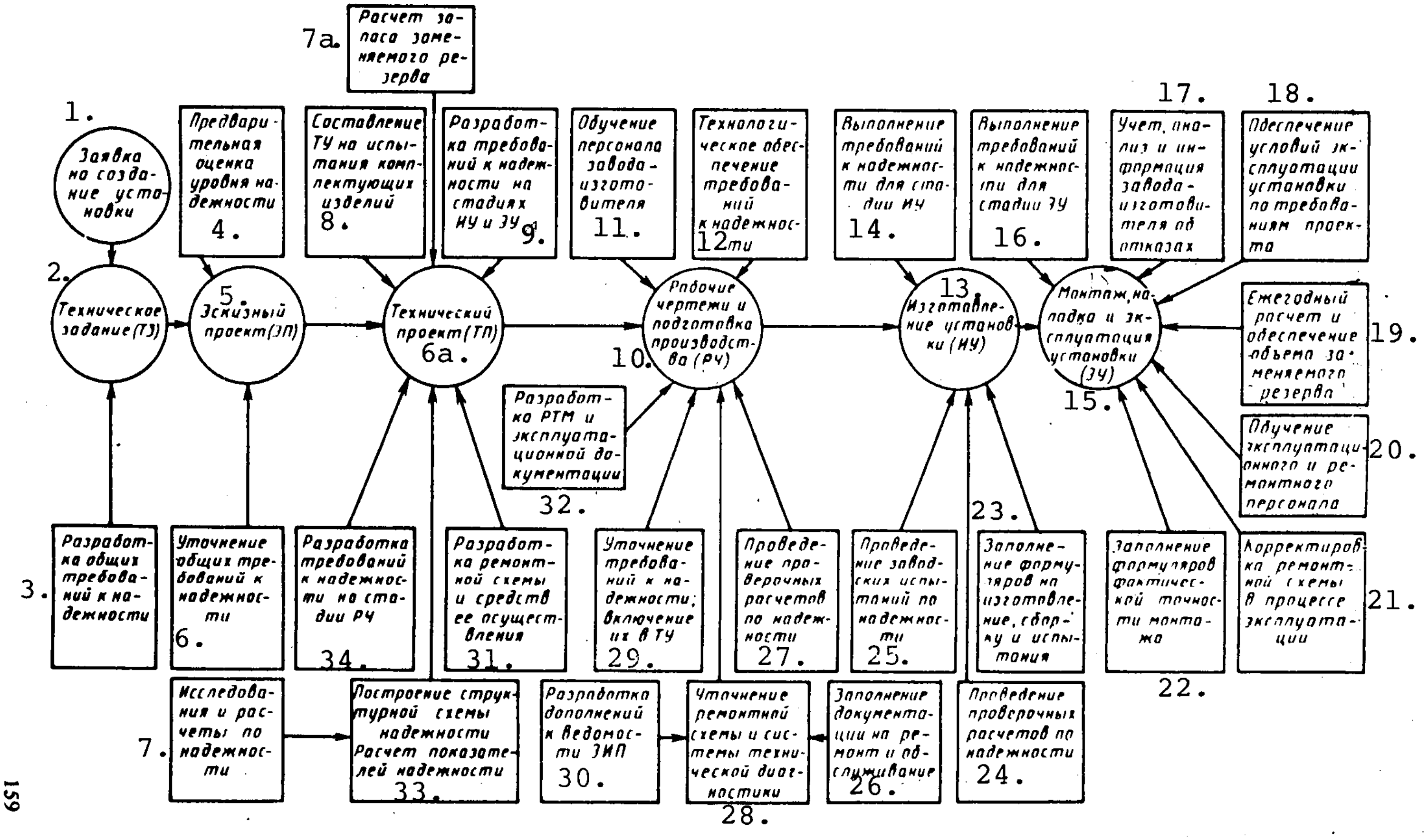

Figure 13.2. Structural diagram of the major work in ensuring the reliability and improving the efficiency of the test stand assembly (TSA).

Key: 1. Request for the design of assembly; 2. Technical assignment (TA); 3. Development of general requirements in regard to reliability; 4. Preliminary evaluation of the reliability level; 5. Sketch drawing; 6. Refinement of general reliability requirements; 7. Study of reliability and correlated calculations; 8. List of tests of the assembly components involved; 6a. Technical project; 7a. Calculation of available reserves in the component which is being replaced; 9. Development of requirements in regard to reliability at the stages of assembly manufacture and final adjustment; 10. Work drawings and preparatory steps before production commences; 11. Training of personnel at the manufacturing enterprise;

(Key to Figure 13.2 is continued on the following page)

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Key to Figure 13.2, continued:

12. Technological provisions to fulfill the reliability requirements; 13. Assembly manufacture; 14. Fulfillment of requirements in regard to reliability at the assembly manufacturing stage; 15. Assembly adjustment and operation of the assembly; 16. Fulfillment of the reliability requirements at the assembly adjustment stage; 17. List analysis and data transmission from the manufacturing enterprise, reflecting the failures; 18. Provision of proper operational conditions for the assembly, as required, according to the project design; 19. Annual calculations, accounting for the scope and volume of backup reserves and items which are to be replaced; 20. Traiing of the operational and maintenance personnel; 21. Correction of the repair and maintenance schedule during operation; 22. Completion of documentation, describing the actual accuracy of the assembling stages; 23. Completion of documentation as to the manufacture, assembly and testing; 24. Verification calculations related to reliability; 25. Reliability testing at the manufacturing enterprise; 26. Completion of documentation as to the repair and maintenance; 27. Reliability verification and related calculations; 28. Refinement of the repair and maintenance scheduling and of the system of technical diagnostics; 29. Refinement of the reliability requirements and their incorporation into the technical manual; 30. Supplemental information, attached to the specifications of the manufacturing enterprise; 31. Development of the maintenance schedule and definition of means to accomplish such maintenance; 32. Flow chart development of the operational documentation; 33. Development of structural reliability diagram. Calculation of the reliability factors; 34. Development of the reliability requirements at the preparatory production stage.

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or block units, it becomes possible to improve and incorporate in depth the engineering control methods associated with reliability. In handling this problem, it is necessary that the designers, chief designers, researchers and other experts who are involved in the TSA utilization and repair would cooperate and function as a unit.

The technical documentation for the TSA manufacturing and utilization cannot be viewed as a complete one if it excludes the technical requirements associated with the reliability and some other data, related to the reliability control.

The economic gain, obtained by improving the reliability, while designing the TSA, will pay in a very short time period for all the additional expenditures.

In the case of operational TSA, the utilization of data, obtained by using the reliability theory, makes it possible to improve significantly the planning of maintenance work, reducing at the same time the expenditure of time and necessary means.

PART III.

FUNDAMENTALS OF EXPERIMENTAL FINE TUNING AND OPTIMIZATION OF SPACECRAFT ON THE GROUND /160

The experimental optimization on the ground with the verification of functioning in thoroughly simulated operational conditions in the case of newly designed spacecraft for the study of near and distant space, for atmospheric studies and planetary surface studies, makes it possible for the scientist and designers to verify the assumptions that the device developed by them is sufficiently reliable for successful fulfillment of the whole flight program (the role of fine tuning and optimization on the ground is particularly important in the design of space robots which are the automated scientific instruments, operating autonomously for a long time, measured in years). The field fine tuning on the ground of the spacecraft and of its experimental prototypes and models is the most expensive and time-consuming stage in the functional cycle of the modern space technology.

We shall consider here the main questions of organizing, structuring and conducting the experimental optimization of the spacecraft on the ground, simulating the multifaceted external action, presenting some technical parameters at the test stand and at the assemblies and the means used to measure and record the physical quantities. A particular attention will be devoted to the physical modeling of the separation of several stages of the spacecraft, the atmospheric entry modes, the parachute activation and soft landing on the planetary surface by using the dynamically similar models and actual mockups of the spacecraft, including the modeling of the properties of the Moon and of the planets and their respective atmospheres. In addition, as much as possible, we shall present the comparison between the fine tuning optimization results obtained on the ground and the theoretical data, as well as the data obtained onboard of the spacecraft during the flight tests.

Some of the above-mentioned problems have been investigated in the studies [15, 35, 43, 63, 80, 81].

CHAPTER 14.

GENERAL INFORMATION ABOUT THE STRUCTURE AND ORGANIZATION OF THE EXPERIMENTAL FINE TUNING AND SPACECRAFT OPTIMIZATION

A new approach has been developing in recent years in the course of spacecraft design. The tremendous tasks which are involved and which are to be handled by the present-day space technology (the development of long-term orbital stations of Salyut and Skylab type, the design of cargo spacecraft, of the automated devices for the flight toward the planets of our Solar System, the close-by grazing flights, the orbital flights, touchdown and take-off) have unavoidably resulted in exceedingly more and more complicated technology and this means - a significant increase in its cost.

It is for that reason that one had to cease experimental testing of new devices primarily during the flight tests. The emphasis in experimental optimization is more and more centered around the comprehensive fine tuning of the spacecraft on the ground, with a complete simulation of the operational conditions. /161

Briefly, such new approach may be characterized in the following manner: it is necessary to do everything possible in order to design the reliable space systems by means of the optimization on the ground. During the flight, one must conduct only such experiments which, at the present level of technology, one cannot conduct on the Earth. Naturally, in order to bring about and realize such approach it is necessary to have the appropriate material base which is a unique comprehensive test stand on the ground, a number of thermal pressure chambers, of tests sites and other experimental assemblies. The incorporation in practice of the comprehensive optimization of the operational devices on the ground, as well as the testing of experimental prototypes makes it possible not only to reduce the costs, but also to "leave on the Earth" the major part of these expenditures in the form of an experimental base on the ground, in the form of new production capabilities, etc.

As one can see, the major reasons, associated with the increase of the experimental fine tuning on the ground during the present-day development stage of space technology, are the following:

accumulation of experience in the spacecraft system design, development of scientific methods, making it possible to ensure the spacecraft reliability by testing it on the ground and also the development of experimental assemblies, of various stands and the measuring means;

a significant increase in the complexity of the space systems, very high cost (the Saturn-5 rocket cost 185 million dollars and the total cost of launching the Saturn-Apollo system was approximately 400 million dollars) which prohibits tests during the flight;

ability to obtain much more information about the spacecraft parameters during the tests on the ground as compared to the flight conditions, because of less complex but more accurate methods which are used in measuring and in repeat testing, and also a possibility of a thorough check out of the device, once the testing is completed.

A complete and thorough experimental optimization of the spacecraft on the ground, the result of which is the spacecraft high reliability, makes it possible to reduce several fold the expenditures associated with flight testing and to reduce time required for such flight testing.

Such high level of experimental studies of the spacecraft may be attained if:

one has a modern scientific and test complex, capable of a continuous experimental fine tuning and optimization of all its systems;

one has the efficient scientific methods for problem-solution, which may occur during tests, including the methods which are used to evaluate the spacecraft reliability on the basis of the ground tests;

one has a sufficient number of well trained test personnel (engineers and technicians) and of the scientific workers who are involved in the development of new methods of spacecraft experimental fine tuning and who are developing methods and means to simulate the operational conditions.

The process of designing a modern spacecraft which commences with the design plans and approval of an appropriate technological assignment (TA) consists of the following stages:

the design of spacecraft and objects within the ground complex /162
of spacecraft servicing (EP);

the autonomous ground testing by the technologists of the separate aggregates and spacecraft systems (ENAI);

factory testing of specific aggregates and subcomponents, incorporated within the airborne device (EZI);

the comprehensive testing on the ground of the spacecraft and of the other objects within the complex (ENKI);

interdepartmental (state) ground tests at the test sites of the whole airborne device complex (EGPI);

the flight design tests of the spacecraft (ELKI);

experimental use of the spacecraft (EOE) during its commercial production.

At the design stage, one develops the technological conditions, drawings and other appropriate design documentation for the manufacture of the mock-up and of the first experimental prototypes of systems and various items within the spacecraft complex, to be optimized on the ground.

In the case of the one-way space travels, by using for example the automated space devices as they fly toward different planets, the two last stages are absent, and therefore the experimental optimization on the ground must provide such high degree of reliability that the purpose of flight would be fulfilled at the very first try.

The subsequent stages of the experimental fine tuning on the ground are the test cycles of the spacecraft and of its systems at the higher and higher hierarchic levels (one first optimizes the component parts of the spacecraft, then one fine tunes its larger and basic systems, after which one tests the spacecraft as a whole, using the ground equipment).

During all these stages of spacecraft development, one improves its quality, trying to achieve the operational reliability levels, as defined by the technological assignments. At various stages of spacecraft design, the reliability factors do not remain constant since the stages under consideration differ, one from another, in terms of approximating the actual operational conditions. Consequently, the reliability factors which are determined at the end of the stage which precedes another stage, and at the beginning of the subsequent stage, in a general case, should not coincide. As one can see, objectively speaking, this is a complex process of the spacecraft reliability change which can be traced out during all stages of the spacecraft design (Figure 14.1).

Because of the fact that during the design stage, one can take into account only the results of the preceding laboratory tests of the materials and various components and the evaluation of the interaction of such components within the spacecraft assemblies, particularly the technology of the spacecraft production and accounting for the actual operational modes during testing of the external activities of the surrounding medium, must be conducted intuitively on the basis of the experience of the designers and also by employing approximation methods, in proceeding from the EP stage to the ENAI stage, a large jump is possible in the spacecraft reliability function. The considerable jumps in the reliability function are also recorded as one proceeds from EZI stage to ENKI stage, and from EGBI stage to ELKI stage. Qualitatively, these jumps characterize the objective reality of the fact that the modeling level at the preceding fine tuning stage reflects the actual operational conditions of the spacecraft and the physical essence of the external activities all of which are reflected in more approximate terms than during the subsequent stage of spacecraft optimization. /163

For example, the change in the reliability function, as one proceeds with the comprehensive testing of the spacecraft on the ground, defines the degree of accounting for the specific relationships (interactions within the spacecraft) which cannot be determined as the designers carry out the autonomous testing of separate assemblies and component systems of the spacecraft. As one commences with the LKI stage, the reliability change reflects the quality of simulation of the space conditions, of the actual blast-off and flight modes, by employing the test assemblies, pressure chambers, blast-off stands, test sites at which, by using the real time scale, one carries out the whole operational program of the launch and flight of the device by using the operational means on the ground.

Statistical data indicates that as a result of a thorough refinement and optimization of all experimental stages on the ground (ENAI, EZI, ENKI, EGPI) it is quite possible to find more than 90% of the defects which are in some hidden form within the spacecraft itself, and which can be simulated by the ground test complex. The remaining hidden defects are found and removed during the spacecraft test flight stage and during tests of its operational viability (ELKI and EOE). As one can see, the experimental fine tuning of the spacecraft on the ground is involved primarily in the elucidation and removal as early as possible of the design shortcomings, of the errors in technological solutions, the errors in the functional schematics and in the control algorithms, all of which is of great importance in reducing to a minimum the probability that a failure may occur during the flights.

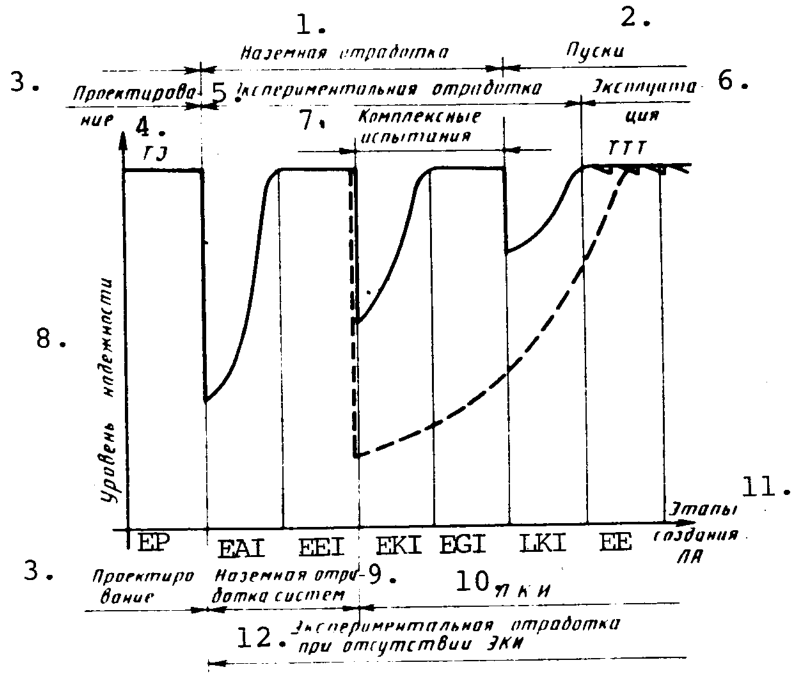

Figure 14.1. Change in the airborne device reliability as a function of design stage.

Key: 1. Fine-tuning and optimization on the ground; 2. Blast-off or launches; 3. Design projects; 4. TA; 5. Experimental optimization; 6. Operation; 7. Comprehensive testing; 8. Reliability level; 9. System fine-tuning and optimization on the ground; 10. LKI; 11. Design stages of the airborne device; 12. Experimental optimization in the absence of EKI.

To express in concrete terms the technology which is being used, let us present some definitions. For example, one must understand by an experiment, a process of active interaction with the object which is being studied, conducted by using the material means within a set of some conditions, in order to elucidate for practical utilization the properties and parameters of the object. In our case, the object which is being investigated is the technological system (the spacecraft, its systems and aggregates) in regard to the specific properties within the area of uncertainty, with the latter being removed by conducting a given experiment. /164

The conditions which are assigned within the framework of an experiment are defined by a set of external actions, in the course of which, the technological system which is being designed is put to use. In a general case, the external actions may be of environmental origin (temperature, pressure, humidity, winds, radiation) and of the man-made origin. The man-made external activities are either the results of interaction between the technological system and environment (vibration, acceleration, aerodynamic heating, ionization) or it can be the result of predetermined activities of man. Let us note that in the process of optimization, using the on-the-ground experiments, the greatest difficulties are associated with the simulated testing by using the test stands which monitor the external actions and which are the result of the technological system functioning, while being used in the environmental conditions.

The experimental optimization on the ground is being viewed as a sequential series of experiments, with the reproduction of all operational modes related to the object which is being tested, the operational conditions, the emergency situations and the ways to alleviate the latter. The experimental optimization on the ground which is a process of sequential removal of uncertainties which occur in real time, is being conducted until all types of discovered defects are removed and until the specifically assigned technological parameters are attained, substantiating the viable operation of the system, plus some other reliability parameters.

In planning the comprehensive testing on the ground, one should be adhering to the following basic assumptions (principles) making it possible to take into account (in the integral form) the most varied questions of the scientific, technolgical, economic and organizational nature, in which the experience related to the experimental spacecraft optimization on the ground is being presented in a general form: the tests to which the spacecraft are exposed are being carried out in accordance with the technological documentation and in the presence of fully equipped spacecraft;

in the beginning, the spacecraft is subjected to tests in regard to the most intense actions, defining the maximum number of defects which are to be removed and this is the priority number one.

Some systems and assemblies are tested for limiting loads, the others - for the operational length of time in the actual operational mode;

the tests should include all types of operational activities and effects, and if it is possible, such interactions are combined. The level of interaction must be able to elucidate the weakest spots in the design and the possible failures, while the spacecraft is in flight;

the tests make it possible for the spacecraft designer to become convinced that this design satisfies the requirements of operational viability in the presence of the loads, the magnitude and duration of which is equal to the flight conditions, multiplied /165
by a specific safety factor;

in the course of testing, the automated data processing is being conducted (at the rate of tests which are being conducted) with a thorough analysis of all detected defects and the recommendations for the removal of such defects;

after the optimization of design, of operational mode or of the production technology, the spacecraft undergoes the repeat comprehensive testing (here one is permitted to reduce the test program for specific testings);

the comprehensive tests of spacecraft on the ground are fully completed prior to the flight testing (for the conduct of which a special release permission is required which ascertains that the device is capable of proper functioning).

A thorough set of spacecraft experimental optimization on the ground, in accordance with the above-mentioned principles, will fully ensure that:

it will be possible to conduct a large number of various tests in regard to one product sample (the volume of the obtained information here is condiserably larger than during the flight tests);

it becomes possible to find definitely the causes of defects in the test samples within the comprehensive set of tests and to remove them without losing the test samples themselves, as well as the other parts of the whole test assembly;

it becomse possible to reduce sharply the time involved in the spacecraft development and to reduce many fold the program costs as a whole.

The organization of experimental spacecraft optimization includes the following types of activities:

planning of the experiments and tests (the projected removal of uncertainties); the substance and volume of experimental testing, the determination of types of tests and sequential use of the test stands, the substantiated optimal combination of the simulating external actions, the development of control algorithms in order to obtain the information with the necessary accounting of limitations in terms of the test times and in terms of the means which are being expended;

planning of the necessary measurements within the test cycle, the volumes of tests, the accuracy and functional viability in order to obtain the information about the operation and general state of the spacecraft which is being designed, the means of simulation which are used to test for external conditions, which describe the parameters of the test stands, the means which are used for simulation work and the list of service systems;

the proper management of the obtained data, including the control of the test product, of the test equipment, of the service systems and equipment used for simulation work, in order to be able to monitor the operational control of the general state of the spacecraft and its assemblies, their physical viability during emergency situations with the provision of safety for the test personnel;

the processing of the obtained data in order to generate the rapid data flow, defining the target-oriented test results, operational control and also to acquire the experience, accumulating the statistical data.

The rate of the test data processing depends on the purpose for which it is generated and on the whole process dynamics during testing (ordinarily, in testing the complex technological systems the analysis of results is conducted automatically, at the rate of the testing itself, with the output of this data in a form convenient for practical use): /166

the depiction and distribution of the obtained data at different management levels, in the interest of operational control and the management of the whole test process. The test personnel who are responsible for the spacecraft and test stand systems, the man in charge of the test and the management issue for each operational level the data in an appropriate form, size, accuracy and operational efficiency, either in its original form (the current numbers, video recordings, oscillograms, movie films) or in the processed form (tables, curves, drawings, diagrams, empirical formulas);

issuance of recommendations on the test results: as to the removal of the defects and failures, determined during testing, with the concrete indication, what and how the optimization is to be achieved, as to the continuation of a given type of test with changed conditions or using different test modes to obtain the additional information which would refine the preceding results, as to the switch-over to the next type of testing, according to the technological flow chart for optimization, as to the general spacecraft control during its use and the state of readiness for such control.

Figure 14.2 shows a typical hierarchic control diagram of the product testing by using a special test assembly. As a rule, the hierarchic control structure is quite characteristic for the complex technological systems, including the system controls of the complex spacecraft testing on the ground.

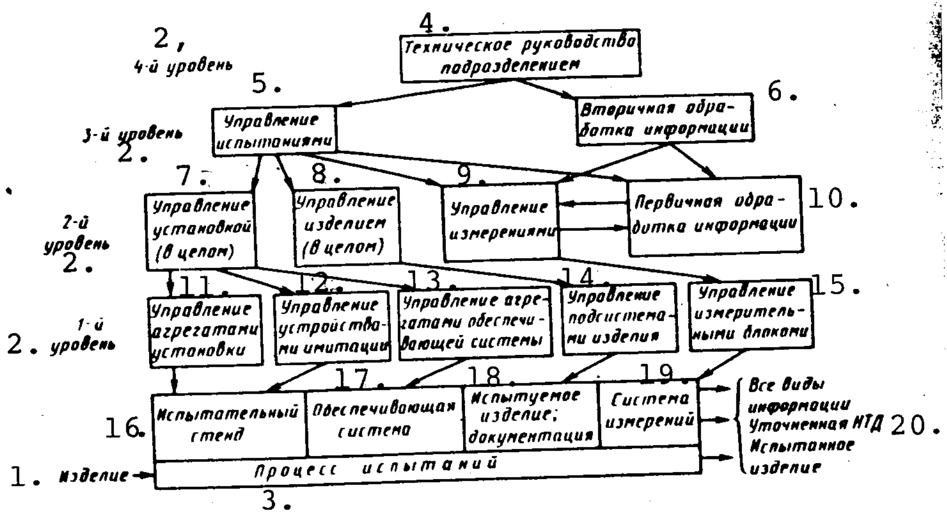

Figure 14.2. Typical flow chart diagram for the testing of a product by using a special test assembly.

Key: 1. Product; 2. Level; 3. Process of testing; 4. Technical management of the department section; 5. Test control; 6. Secondary data processing; 7. Control of the test assembly (as a whole); 8. Product control (as a whole); 9. Control of measurements; 10. Primary data processing; 11. Control of various block units within the test assembly; 12. Control of the simulation assembly; 13. Control of assemblies in the support system; 14. Control of the product subsystems; 15. Control of the measuring block units; 16. Test stand; 17. Support system; 18. Product which is being tested: documenation; 19. Measuring system; 20. All types of information, refined scientific and technical documentation, product as tested

The main specifics of such control structure are as follows:

rational coupling of the global and special criteria of optimality;

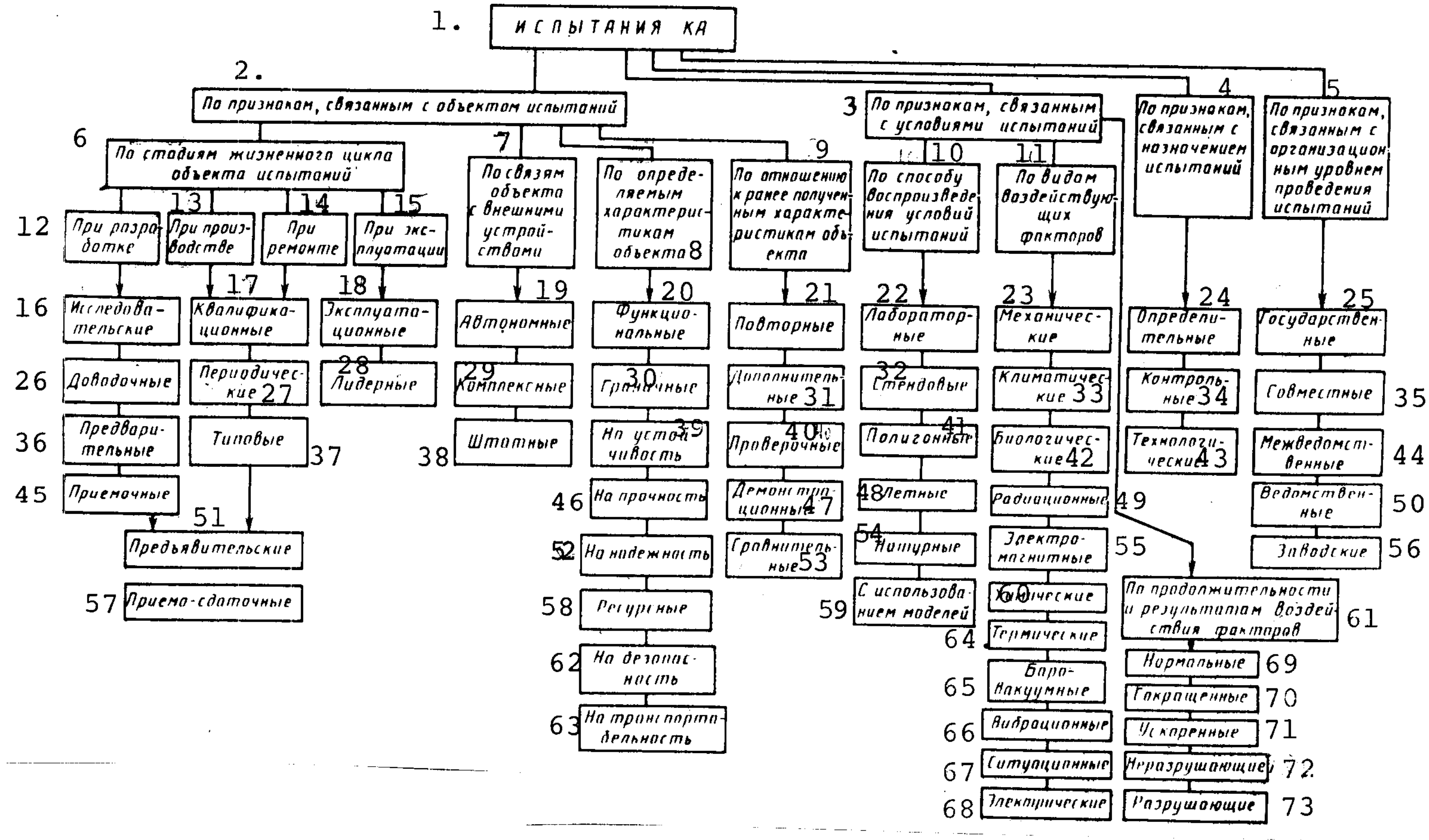

Figure 14.3. Classification of the types of spacecraft tests.
Key: 1. Spacecraft tests; 2. Based on the features, connected to the test object;
3. Based on the features, connected with the conditions of testing;
4. Based on the features, related to the purpose of the tests;
5. Based on the features, related to the organizational level at which the tests are conducted;

(Key to Figure 14.3 is continued on the following page)

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Key to Figure 14.3, continued:

6. In terms of stages of the test object life cycle;
7. Connections between the object and external test devices;
8. Based on the object's parameters which are being determined;
9. Related to the previously obtained parameters of the object;
10. Method of reproducing the test conditions;
11. On the basis of types of the impacting factors;
12. During design; 13. During production; 14. During repair;
15. During use; 16. Research; 17. Qualifying; 18. Operational;
19. Autonomous; 20. Functional; 21. Repeat; 22. Laboratory;
23. Mechanical; 24. Of defining type; 25. On national level;
26. Additional; 27. Periodic; 28. Leading; 29. Comprehensive;
30. Limiting; 31. Supplementary; 32. On test stand; 33. Climatic;
34. Controlled 35. Combined; 36. Preliminary; 37. Based on product type;
38. Standard; 39. Stability; 40. Verifying; 41. On the test site;
42. Biologic; 43. Technical; 44. Interdepartmental;
45. At the reception level; 46. Durability; 47. Demonstration;
48. Flight; 49. Radiation; 50. Interdepartmental; 51. Delivery;
52. Reliability; 53. Comparative; 54. On-site; 55. Electromagnetic;
56. Factory; 57. Delivery and reception; 58. Resource availability;
59. By using the models; 60. Chemical; 61. In terms of the length and results of the impacting factors; 62. Safety;
63. Transportation and mobility; 64. Thermal; 65. Pressure and vacuum;
66. Vibration; 67. Special situations; 68. Electrical; 69. Normal;
70. Abbreviated; 71. Accelerated; 72. Nondestrcutive; 73. Destructive

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freedom of local actions within the established framework;

improved reliability and reduced control time;

reduced data flow as a result of preliminary processing of the primary data, before it is being sent to the higher levels, and flexibility of the control system as the external conditions /168
change and the flexibility in using the available resources.

All levels of the hierarchic structural control function in real time, coordinating the activities of the subjugated components and levels, in accordance with the requirements of the higher levels. At the highest, fourth level (the so-called system of the end results) one generates the corrected control data for the whole test process. At the first level, one brings about a continuous control mode of all subassemblies (block units) in test assembly and in the support systems, the control of assemblies which simulate the external action and the systems within the test objects which are being developed (Figure 14.3).

CHAPTER 15.

MAJOR PARAMETERS OF EXPERIMENTAL ASSEMBLIES AND STANDS FOR THE SPACECRAFT FINE TUNING AND OPTIMIZATION

In conjunction with the more complex construction design and the new space targets, the necessary increase in quality and reliability of the spacecraft create the continuously increasing requirements which are imposed upon the space technology and the scientists and designers are faced with ever-increasing complex scientific and technical problems. The availability of improved experimental means - the special test stands and assemblies, is a mandatory prerequisite which ensures the effective development of space research and the constant improvement of the experimental and scientific base is a mandatory condition for progress in space technology. /168

The design and development of the majority of experimental means avaialble for research and testing of spacecraft which are being developed is associated with the large capital investment and requires considerable time. The bulk of experimental assemblies is, to one degree or another, quite unique in design, with no other design analogues. The research and test studies of the experimental equipment is correlated with the requirements imposed upon the spacecraft which are being developed. The development and use of the test assemblies and stands for fine-tuning and optimization of the spacecraft must take into account the technical and economic criteria of efficiency: the operational efficiency (reliability, the attainment of assigned parameters, extensive informative base), economic (cost of design, efficient use), and the data related to technical realization of the project (whether the project can be accomplished, the possibility of upgrading).

The goals and targest of the experimental studies, associated with the spacecraft research, define the composition and sequence of the actions to which the device is being exposed, with the action which is to simulate on the ground, various operational factors, composition, content and sequence of the control operations which are to be executed, handling the parameters which are to be measured, the type and quantity of the control and measuring equipment which is being used, of various test devices and transition attachments.

The test stands and assemblies which are designed for such work /169 are such technical devices which are to be used to test an object and the test assemblies are to be set up in such a way as to create the action, to be able to obtain the data, and to be able to control the test process and (or) the test target. As a rule, the various purpose test stands consist of two parts, connected by the communication lines: the executive part in which the object which is being tested is placed and the action is provided, which simulates the operational factors and the command part which provides for the generation of the necessary data, transformation of such data, its analysis and its depiction as well as the manual and automated control of the whole experiment.

The test stands and assemblies which are used during the experimental research and during the spacecraft testing, testing of its assemblies and subsystems, may be classified in the following manner. In terms of the objects which are to be tested, in terms of the actions to which the object is to be exposed, in terms of the physical nature of the operational actions which simulate the actual environment, in terms of the rules of change in the reproduced actions, in terms of the nature of the loads to which the test object is exposed, in terms of the types of quality parameters which are used, in terms of the method of parameter control, in terms of the data reception, communication channel of the test means and the object which is being tested, the method of data processing, the method of control, pertaining to the test mode, the degree of automation, degree of standardization and the degree of unification.

The Requirements Which Are Imposed on the Test Objects and on the Test Equipment

The object which is being tested must satisfy a number of the following requirements:

the object which is to be tested is manufactured on the basis of spacecraft work drawings which must be identical, geometrically, mechanically, electrically, etc.;

the mass m, the centering and balancing x_{cm} and the moments of inertia of the object which is being tested (the minimum and maximum) are determined experimentally and correspond to the spacecraft design;

the replacement of specific components in the object which is being tested by the same mass and size mock-ups is permissible only in such case when this will not affect the strength and operational capabilities of design;

the test objects must be air-tight;

the equipment in the test object is verified by using the autonomous and comprehensive functioning, with the measurement of major parameters;

the component parts and work materials which are used in the mechanisms and assemblies must correspond to the exact drawings and the replacements at the testing stage which is not well-founded is not permitted;

the special subassemblies which are mounted on the test object for the purposes of its attachment or load application must not alter the strength and rigidity of the design, and must not hinder the deformations of the objects during its testing;

a number of probes and converters to properly fix the parameters are being mounted on the object which is being tested.

The objects are tested in the presence of high loads and therefore, after testing they cannot be used during the flight. /170

It is possible theoretically to test the whole spacecraft design but in the majority of cases, its separate assemblies are being tested. This is due to three reasons.

The first one is the fact that for various assemblies within the spacecraft, the various cases of load application are the calculated functions. Therefore, in conducting assembly-by-assembly testing one can test the levels of the load applications which were computed for the bulk of the design assemblies, utilizing just one sample. The second reason is the great technical difficulty of testing the spacecraft as a whole. The third reason is the lowered reliability of the whole assembly because of the quantitative increase in the number of pieces of equipment and of different control and recording devices. Frequently, the repeat experiment does not produce the necessary information about the strength and rigidity of the design because of the residual deformations which were created during the first experiment.

The separate assemblies are subjected to testing together with the appropriate transfer and coupling devices which in terms of their parameters must simulate as closely as possible the construction design piece which they are replacing.

The test stand equipment must satisfy a number of requirements:

it must provide the identical or dynamic semblance of the operational processes;

it must ensure that the side effects to which the test assembly is subjected are at a minimum;

the operations must be maximally automated and mechanized with the use of computers, particularly in the area of the program control, general control, data recording, parameter measurements and final data processing;

the tests must be conducted in the shortest possible time with minimal expenditure of means;

the general level of efforts must be sufficient to conduct the tests;

the specific level of efforts and power outputs must be maximized;

the operational lifespan, in unfavorable conditions, must be provided (increased humidity, presence of dust);

the frequency stability parameter must be within the range of established tolerances, as the masses or loads are being changed;

the attainment of the assigned operational mode must occur at the shortest possible time;

the equipment must be simple, uncomplicated to manufacture and not expensive;

the standard sources of power must be used;

it should be possible to generate the load and ensure stable operational mode with the assigned degree of accuracy during a specific time interval, having at the same time the parameters which would satisfy the requirements as per National Technical Documentation Format;

it should be possible to obtain the information about the capacity of the test object to withstand the destructive load application within the assigned accuracy of the qualitative estimates;

the generated data should be in a form, convenient for the operative use;

it should be possible to conduct the tests by checking out the harmonic (sinusoidal and polyharmonic), random (narrow band and wide band) and mixed (harmonic and random) vibrations; /171

it should be possible to reproduce the test results which exclude the dissimilar conclusions;

it should be possible to account for specifics in the parameters of the test object and the methods and means which were used to fasten the object to the stand;

the stand should be simple and convenient for the control of the specific test mode, it should be possible to handle it either manually or automatically;

it should be possible to reproduce and measure the vibration parameters in the frequency range of 0-10 kHz, in terms of the accelerations - up to 1000 m/s^2, in terms of the displacements from

0.1 μm to 1 m, and in terms of spectral acceleration density - up to 1.0 g^2/Hz;

one should provide accuracy of frequency measurements <50 Hz $\pm$ $\pm$(1 ... 2) Hz and >50 Hz $\pm$(2 ... 3)%, the accuracy of displacment $\pm$(10 ... 20)%, of the acceleration $\pm$(10 ... 20)% and of the spectral acceleration density $\pm$(1.5 ... 3.0) dB;

the nonlinear distortion tolerances should not exceed 30%. The levels of cross-sectional component vibration should not be more than 25% and the nonuniform amplitudinal oscillations at the fastening points should be $\pm$25%;

one should provide the impact force up to several hundred thousand N, and the load-carrying (load-lifting) capacity up to several tons.

To monitor the proper operation of the test stand, it is provided with the audio or light signaling system, including the emergency signaling system with simultaneous current cut-off.

On the whole, the test stand must satisfy all the requirements of safety and must, first of all, be equipped with fire extinguishing devices.

The degree of perfection of such test stands is characterized by the availability of technical equipment, by the availability of power, by the availability of the control and measuring equipment, by the time required for preparatory work and for the test as a whole. With the better grade of technical equipment available at the stands, better test results will be produced and the spacecraft fine-tuning and optimization on the ground will be more effective.

The devices for data gathering, measurement and transformation, the means available for the data depiction and other representation, ensuring the assigned accuracy of measurements must be of high speed, stable against noise and interferences and must correspond to the requirements of the technical standard documentation.

Classification of Vibration Stands

In testing a spacecraft and its components, the vibration stands may be classified in terms of their purpose, in terms of the execution, the type and action which is generated by the mechanical oscillations, the number of component parts and oscillation shape, the operational principle of the vibration source, the dynamic schematic and principle of excitation of the variable force in the source of oscillations.

In terms of purpose, we may have:

the vibration test stands to check out the vibration resistance and strength;

the vibration stands to test the fatigue and strength of the test materials, of the components and assemblies;

vibration stands, to graduate, calibrate and verify the vibration measuring equipment. /172

In terms of the executed functions:

portable vibration stands for the normal and specific operational conditions;

stationary vibration stands with one or several excitation oscillations;

In terms of the type of generated mechanical oscillations:

vibration stands for rectilinear oscillations;

vibration stands for rotational oscillations;

In terms of the directional action of the developed mechanical oscillations:

vibration stands which generate oscillations along or around the vertical axis;

vibration stands with variable angle of directional action of the generated oscillations, with respect to the horizontal plane.

In terms of the number of components in the generated mechanical oscillations:

single-component vibration stands to generate the rectilinear and circular oscillations;

multicomponent vibration stands to generate planar and tridimensional oscillations;

In terms of the shape of the mechanical oscillations:

vibration stands to generate the harmonic or quasiharmonic oscillations;

vibration stands to generate polyharmonic oscillations;

vibration stands to generate the broad frequency spectra oscillations.

In terms of the operational principle in the mechanical oscillation exciter:

nonresonance vibration stands;

resonance vibration stands with external self-excited oscillations;

In terms of the dynamic flow chart:

vibration stands with kinematic constraints on the bench displacement;

vibration stands with kinematically unlimited direct excitation of the oscillatory bench displacement;

vibration stands with kinematically unlimited indirect excitation of the oscillatory bench displacement.

By using the principle of variable force excitation in the oscillation exciter (Figure 15.1).

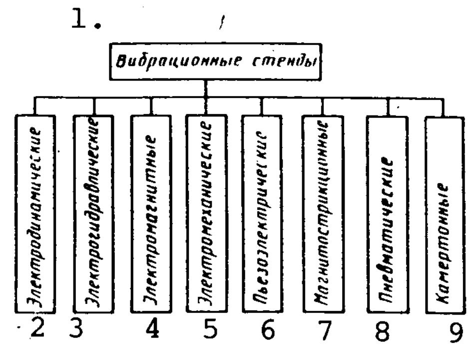
The specifics of the physical properties of the vibration exciters which are used on the test stands are defined to a large degree by their frequency and force parameters. Therefore, regardless of the schematic diagram, of the design and size of the stands, one can determine the areas of its use, employing differing excitation sources for the required type of test (Figure 15.2).

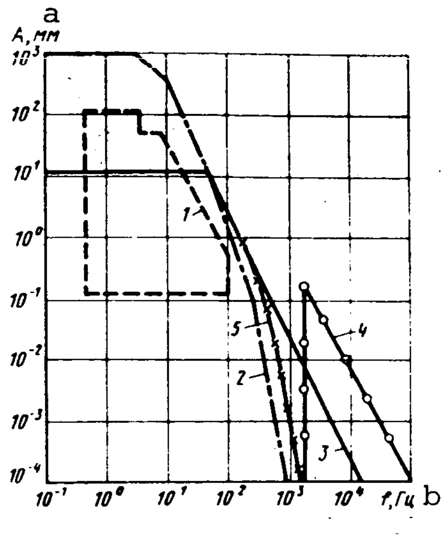
We shall present below the description of various types of stands.

1. The most standard ones are the stands with electrodynamic vibrational excitation (Figure 15.3). Such stands make it possible to smoothly adjust the amplitude and frequency of the assigned vibrations. The vibration sources have no friction parts, their lifetime is sufficiently high and it is defined only by the lifetime of the coil suspension control system.

The vibrators of small power output are simple to manufacture, /173
they are cheap and do not contain any subassemblies which generate increased levels of interferences or noises.

The drawbacks of vibrators is their small specific power output and also the induction electromotive force in the control coil which helps to dampen the oscillations. The powerful vibration sources generate a considerable magnetic field at the level of the vibrator bench and to decrease the intensity of such field it is necessary to use the protective shields, which complicates the vibrator design.





15.1 15.2

Figure 15.1. Types of vibration on stands, depending on method used for vibration excitation.
Key: Vibration stand; 2. Electrodynamic; 3. Electrohydraulic; 4. Electromagnetic; 5. Electromechanical; 6. Piezoelectrical; 7. Magnetostriction; 8. Pneumatic; 9. Tuning for effect.

Figure 15.2. Regions of optimal optional modes on vibration stands: 1. electromechanical; 2. electrohydraulic; 3. electrodynamic; 4. magnetostriction; 5. pneumatic

Key: a. mm; b. Hz

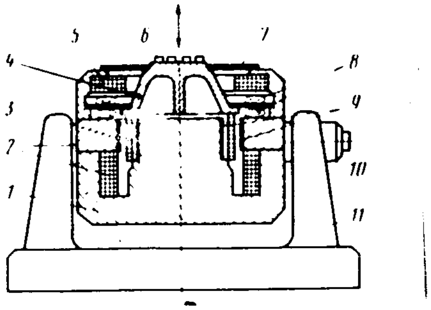


Figure 15.3. Typical schematic diagram of electrodynamic vibration stand:
1 - magnetic conductor; 2 - excitation winding; 3 - moveable coil; 4 - bench; 5 - compensating coil; 6 - threaded bushings of bench; 7. Elastic suspension system of the bench; 8. Pneumatic chambers; 9 - Shields; 10 - Rotating device; 11 - base.

The stands are characterized by a rather broad frequency range which may be generated (5-10^4 Hz). In using some special equipment it is possible to simulate the basic operational modes in the vibration tests. At the high power output stands it is possible to generate harmonic force of up to $5 \cdot 10^2$ kN amplitude.

2. The stands with hydraulic excitation are frequency-constrained in the majority of designs, down to 100 Hz and sometimes 500 Hz. These stands are characterized by high amplitude /174
forces (above 10^3 kN). The remaining parameters depend on the method of primary vibration excitation in the unit (electrodynamic, electromagnetic, mechanical). The vibrations feature high specific power

output, long operational time and the possibility of smooth adjustment of the vibration parameters within a broad range.

3. The stands with electromagnetic excitation function within a limited frequency range (up to 500 Hz). They are used in tests when the reproduction of a specific type of vibration is of no significance. The force involved here is up to 50 kN. The vibrators are simple to manufacture, they last long, they are easy to adjust in terms of the amplitude (by using the current-controlled coil) and the frequency is adjusted stepwise. As a rule, the vibrators are operated by the a.c. standard frequency network.

The small specific power output is the major drawback of such vibrators, making it impossible to use them at high frequencies and with large loads. With the increase of power applied to the vibrator coil, the losses in the iron core increase, as well as losses associated with friction during the deformation of springs, etc. In addition, the electromagnetic vibrator generates during its work the undesirable noises.

4. Stands with mechanical excitation are useful for the long periods of use in the frequency range between 3 and 200 Hz. Such vibrators are easy amplitude-adjustable, the vibration frequency control is somewhat more complicated and is ordinarily accomplished by adjusting the rpm of the electric motor drive. They are not complicated to manufacture, they are cheap and are driven by the /175
standard commercial a.c. network, but they also have the following drawbacks:

the operational life is short because of limited lifetime of the ball bearings;

they produce comparatively small power outputs;

they have extraneous noises and vibrations which is particularly undesirable in the high frequency vibration stands.

They are being used for testing in generating harmonic vibrations. The special vibration stands make it possible to generate harmonic vibrational accelerations with a rather low frequency. The force involved here is up to 10 kN.

5. The stands with piezoelectric and magnetostriction excitation are used to calibrate the low frequency instruments. There are stands for tests at up to 80 kHz frequencies. The force involved here is ordinarily less than 10 N.

6. The pneumatic vibrators are small in size, they are simple to design and are cheap. The frequency and vibration amplitude adjustment in the pneumatic vibrators is quite coarse: the frequency

is adjusted by changing the applied pressure and the amplitude - by setting up special dead stops. The most important drawback of such a vibrator is the impact type of its operation, and as a result of this - the sharp noise. The working frequency of oscillations is up to 800 Hz.

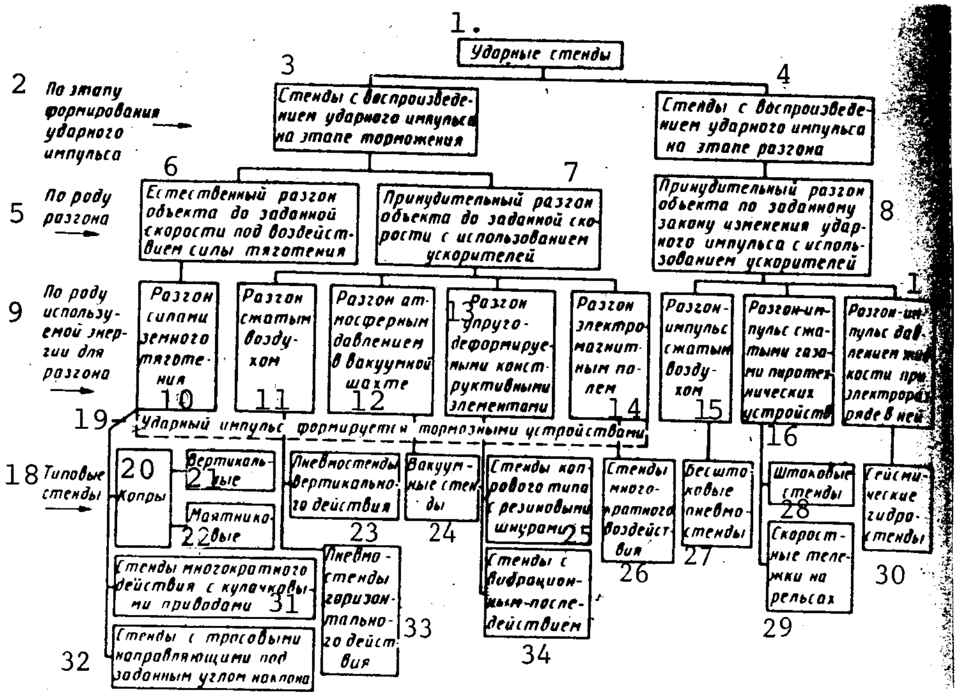


Figure 15.4. Classification of the impact stand.

Key: 1. Impact stands; 2. At the stage of the impact pulse development; 3. Stands for reproduction of impact pulse at the stage of braking; 4. Stands for the reproduction of impact pulse at the stage of acceleration; 5. In terms of the acceleration type; 6. Natural acceleration of the tested object to the assigned velocity level, as a result of gravitational forces; 7. Forced acceleration of the object being tested to an assigned velocity level, utilizing the accelerators; 8. Forced acceleration of the object on the basis of the assigned law of the impact pulse change and utilizing accelerators; 9. On the basis of the type of energy used for acceleration;
10. Acceleration by gravitational forces;
11. Acceleration by compressed air; 12. Acceleration by atmospheric pressure in the vacuum enclosure;
13. Acceleration by the elastic-deforming components within the assembly; 14. Acceleration by electromagnetic field;

(Key to Figure 15.4 is continued on the following page)

Key to Figure 15.4, continued:

15. Pulse acceleration by compressed air; 16. Acceleration by the pulse of compressed gas, using combustion devices;
17. Pulse acceleration by using the pressurized liquid and the electric discharge in the liquid; 18. Test stand types;
19. Impact pulse is formed by braking devices;
20. Copra stands; 21. Vertical stands; 22. Pendulum type;
23. Pneumatic vertical stands; 24. Vacuum-activated stands;
25. Copra stands with rubber belts; 26. Multiple action stands;
27. No-rod pneumatic stands; 28. Rod activated stands;
29. Speed carts on rails; 30. Seismic hydrostands;
31. Multiple use stands with cam drives; 32. Stands with directional cables for a given inclination angle adjustment;
33. Horizontal pneumatic stands;
34. Stands with vibrational after-effect.

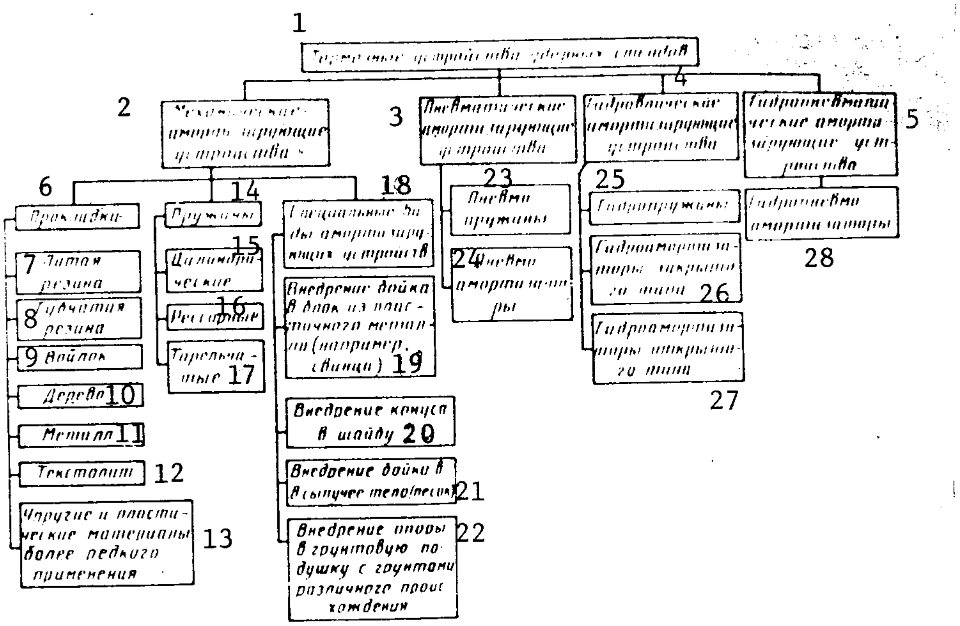


Figure 15.5. Braking devices in the impact stands.

Key: 1. Impact stand braking devices; 2. Mechanical simulating devices;
3. Pneumatic simulating devices; 4. Hydraulic simulating devices;
5. Hydropneumatic simulating devices; 6. Spacers; 7. Solid rubber;
8. Foam rubber; 9. Felt; 10. Wood; 11. Metal; 12. Laminated fabric;
13. Elastic and plastic materials, the use of which is rare;
14. Springs; 15. Cylindrical; 16. Spring-activated; 17. Plates;
18. Special types of simulation devices; 19. Pin penetration into a plate, made of soft metal (for example, lead);

(The key to Figure 15.5 is continued on the following page)

Key to Figure 15.5, continued:

20. Cone penetration into a plate; 21. Pin penetration into free-flowing sandy material; 22. Support penetration into soil of different origins; 23. Pneumatically activated springs; 24. Pneumatic shock absorbers; 25. Hydraulic springs; 26. Hydraulic shock absorbers of open type; 27. Hydraulic shock absorbers of closed type; 28. Hydropneumatic shock absorbers.

The spacecraft tests related to impact accelerations, simulating the touchdown stage by using the parachute system or the soft landing on the planetary surface is accomplished by using various impact stands. The impact stands are classified in terms of the purpose for which they are designed, in terms of the design construction, in terms of the type of impact pulse which is generated and the method of obtaining it, to generate the required impact velocity or acceleration, etc. One of the possible classifications of the impact stands is shown in Figure 15. 4. The types of braking devices used in the impact stand and their basic design components are shown in Figure 15.5. /176

To simulate the linear accelerations which act upon the space-fraft as it is being placed into the orbit and as it enters the planetary atmosphere, it is reasonable to use the centrifugal test stands. The major technical parameters of the centrifuges are the maximum linear accelerations and lifting capacity. The centrifugal stands may be classified by using the following distinguishing features: purpose, type of drive, construction design diagram, lifting capacity and generated acceleration. Depending on the centrifugal acceleration obtained, they may be subdivided into the following groups: up to 100 m/s^2, up to 1000 m/s^2, up to 5000 m/s^2 and above 5000 m/s^2. In terms of the lifting capacity, we have - small ones - up to 10 kg, intermediate ones - up to 100 kg, heavy ones up to 1000 kg, and superheavy ones - above 1000 kg. In terms of the centrifuge drives, we may have electrical, pneumatic, hydraulic, mechanical and combination type. In terms of the construction design - we may have either vertical or horizontal rotational axes and either rigid, or rotating and radially displaced support platform.

Depending on the magnitude and nature of the external action to which the object being tested is exposed, the vibration strength and vibration resistance tests are ordinarily accomplished by using the following means [17, 22, 43]: a rigid harmonic vibration mode, the harmonic vibration of variable and oscillatory frequency, the polyharmonic vibrations, the wide band and narrow band random vibrations and the natural (real) vibrations.

The major dynamic parameters of the spacecraft, associated with the elasticity in design, with the changing quantities of fuel in the tanks, with the mass distribution, etc. - the internal frequencies, the shapes of oscillations, the damping coefficients, the generalized masses - are all of defining importance during all flight stages and operation of the devices.

Among the experimental methods in evaluating the dynamic parameters of the spacecraft and of its models, the most frequently used ones are the following: the method of free oscillations, the resonance method, the Kennedy Penku method, the piezoelectric method, electron method, capacitance method, the method of multipoint excitation and the method of additive masses.

In this study we shall consider only the method of multipoint excitation which is the most effective one in investigating the dynamic parameters of the spacecraft of complex design.

In determining the dynamic properties by the above-mentioned methods, the major difficulties arise because of the response action of the construction design in the presence of the nonresonant oscillations. These difficulties are inherent in all methods in which the excitation of design takes place at one point.

In recent years, one finds more and more extensively used the method of dynamic parameters determination, which is based on the multipoint excitation of oscillations in the design. The utilization of multipoint excitation makes it possible by proper selection of the excitation forces to isolate sequentially the "pure" internal oscillations and to determine on the basis of this the dynamic parameters of the design as a system, having one degree of freedom.

The fundamental possibility of exciting the "pure" self-oscillations in the design, by using the multipoint excitation can be derived on the basis of the general theory of forced oscillations within the linear systems. The problem is to find such force distribution, in the presence of which the "pure" self-oscillations are excited. /177

Let us find from the total levels of forced oscillations the systems with a finite number of degrees of freedom

$$A\ddot{q} + B\dot{q} + Cq = Q. \tag{15.2}$$

If the system oscillates using its own r pattern at specific frequency, which is not necessarily the internal frequency of the system itself, then

$$q = A_r \eta^{(r)} \cos(\omega t + \epsilon r), \tag{15.3}$$

where A_r is a coefficient which characterizes the magnitude of the oscillations.

By substituting (15.3) into the initial equation, we will obtain the following expression for the force distribution

$$Q = A_r[(C - \omega^2 A)\eta^{(r)} \cos(\omega t + \epsilon r) - \omega B\eta^{(r)} \sin(\omega t + \epsilon r)]. \tag{15.4}$$

It consists of two summands. The first summand is a force which is necessary to counteract the elastic and inertia forces. The second summand is a force which is necessary to compensate for the dissipative forces. It has the phase shift with respect to the displacements, equal to $\pi/2$. The distributions in which all forces are either phase shifted or shifted by 180° with respect to each other will be called the multiphase distributions.

By using the multiphase force distributions, one can excite the "pure" internal oscillation patterns within the system under the following circumstances: in the absence within the system of dissipative forces, if the dissipative forces do not involve the major coordinates and if the excitation occurs at the internal frequency.

Of greatest interest is the latter case. By assuming that $Q=F \cos \omega t$ and keeping in mind that in the course of excitation of internal pattern, using the internal frequency, the following condition is satisfied

$$(C - \sigma_r^2 A)\eta^{(r)} = 0, \tag{15.5}$$

we will obtain $\varepsilon_r = (\pi/2)$ and consequently:

$$F = A_r \sigma_r B \eta^{(r)}. \tag{15.6}$$

As one can see, by displacement of all points within the linear system with the phase shift of $\pi/2$ with respect to the multiphase harmonic excitation, the system will be forced to oscillate using a "pure" internal oscillatory pattern, regardless of the fact whether the dissipative forces affect the major coordinates or not. In such case, the frequency of such forced oscillations is equal to the internal oscillatory frequency of the system.

The multiphase force distribution in this case must satisfy the condition (15.5). Since the use of the equation (15.6) to select the forces is extremely difficult, one ordinarily employs in practice the use of the phase resonance criterion, as formulated above. The corresponding force distribution is selected either manually or by semiautomated means, working with the multichannel vibration assemblies. /178

If one is to assume that dissipative forces do not affect the major coordinates, one can obtain a somewhat simpler expression for the multiphase force distribution

$$F = C_r A \eta^{(r)}, \tag{15.7}$$

where C_r is a constant which depends on the excitation frequency. It follows from this that the excitation forces must be proportional to the forces of inertia. And this means that the magnitude of any external force must be selected proportionally to the product of amplitude and internal mass at a given point of the construction design.

It is necessary to point out that the internal oscillation pattern in the case under consideration may be excited at the frequency which is somewhat different from the internal frequency of the system. Therefore, the phase shift will not necessarily be equal to $\pi/2$.

The damping coefficients, when using the multipoint excitation may be determined by various means, by assuming that the system has one degree of freedom. In particular, one frequently utilizes the method which is based on the following formula

$$g_r = \frac{\sum_{j=1}^{n} \eta_j^{(r)} F_j \eta_i^{(r)}}{2 m_{rr} \sigma_r^2 A_i^{(r)}}, \tag{15.8}$$

where $A_i^{(r)}$ is the amplitude of oscillations of a specific point.

The method of multipoint excitation makes it possible to determine dynamic parameters with a high degree of accuracy for the most complex construction designs, but it requires comlex and expensive vibration assemblies.

In determining the dynamic parameters of the spacecraft, one utilizes the multichannel vibration assemblies.

To determine the acoustic strength of the spacecraft, one carries out the following tests: the field testing on the ground, using directly the objects of interest; the open stand with functioning propulsion unit; the closed boxes with natural noise sources, by using the traveling wave channels and in the reverberation chambers.

From the point of view of closest approximation to the operational load application conditions, the on-the-ground field testing is the best method to verify the acoustic strength of the spacecraft design and of its systems. The significant drawback of such approach is extremely high cost, since during the whole course of testing, the propulsion units which generate the acoustic field must operate at the maximum power output. The flight conditions during the acoustic load application on the ground are not being reproduced.

The tests on an open stand with operational propulsion unit is more economical, although it is somewhat different from the field conditions of the load application. In using such method it is possible to test the large sized objects of interest. The duration of tests with the observation of the required conditions in load application in this case depend on the positions of the objects which are to be tested with respect to the noise sources. The test mode is established on the basis of the field measurements of the audio loads and deformations at the control points on the surface of the object in question. /179

The tests in closed boxes make it possible to obtain higher levels of acoustic loads than on the open stands, and as a result of this, the length of testing can be reduced. The drawbacks here are some distortion of the audio field as compared to the natural conditions.

The assemblies with the audio traveling wave are used to test the components of the spacecraft shell (panels) in regard to the acousting strength, applicable to the flight load conditions within the nearest acoustic field and with the direction of the frontal audio wave propagation along the tangential line to the shell surface.

The traveling wave assemblies are more economical to test the panels than the reverberation chambers. However, to test the panels in regard to fatigue, the reverberation chambers may also be used. However, because of very high noise level (160-170 dB and sometimes higher) and because of a broad frequency range (50-1000 Hz) which are used during such tests, it is necessary to build the reverberation chambers of rather large size and the actual magnitude of the acoustic output is practically not acceptable.

The other factor which must be taken into account, while selecting one specific acoustic assembly, is the radiation damping which has been experimentally detected and which affects the load on the panel which is being tested. It depends on the geometric shapes in the test assembly. In a large reverberation chamber, this factor is minimal because the panel is inserted in such a way as if it had been mounted on a infinite sized shield. On the other hand, in the traveling wave assemblies, the nearness of the walls to the panel which is to be tested may influence quite considerably its response. In order to reduce this factor it is necessary to frequency-separate the major resonance panels and the strong standing waves in the operational frequency range of the traveling wave channel.

The reverberation chambers are such enclosures in which, as a result of the resonance enhancement of oscillations in the environment (of either air or gaseous nature) the vibrations are excited by the source and are reflected from the partitioning surfaces, thus creating high intensity acoustic field.

The geometric parameters and operational characteristics of the reverberation chambers are selected in accordance with the size of the objects which are to be tested, with the intensity of the acoustic load and with the range of the working frequencies of the audio pressures.

The reverberation chambers include the test box (a chamber), the preparatory compartment, the audio oscillation system, the coherent assembly (horns), the power system to activate the sound by compressed air, the system for development and control of the acoustic load spectra, the data and measuring system, and the noise quencher for the removal of compressed air, deactivating the sirens.

In selecting the test method, of great importance are the sizes of the object which is to be tested.

The small design components, while being exposed to high intensity audio oscillations are ordinarily being tested by using the traveling audio wave channels and the tests are being conducted on the panels, having the surface area between 1.5 and 6 m^2, which are /180
placed tangentially with respect to the audio wave propagation.

The following advantages are quite characteristic for the reverberation chambers: economic utilization of the acoustic power output of the noise sources, efficient vibration excitation of the design and universality of use.

CHAPTER 16.

FUNDAMENTALS OF PHYSICAL MODELING IN FINE-TUNING AND OPTIMIZATION OF SPACECRAFT

The theoretical studies presuppose the mandatory experimental substantiation that the proper calculation schematic has been chosen, by correlating the assumptions which are being utilized regarding the physical essence of the phenomenon or process which is being studied. /180

However, the experimental substantiation of the operational viability or reliability of functioning, while designing the spacecraft and its system, because of its cost and bulkiness, may be viewed only as a final stage of a large experimental research cycle, conducted on the ground, by using the dynamically and similar in design models and real size mock-ups of the spacecraft [15].

This fact is a basic distinguishing feature of the experimental study, involving the majority of physical phenomena and processes, related to the technological penetration into space (among such phenomena one might refer to the descent through the atmosphere and soft landing, the movements and construction on the surface of the celestial bodies, orbital docking and separation, etc.).

While the utilization of modeling in the study of such phenomena as the aircraft or rocket flutter, the flow of liquid or gas around the streamlined body, the heat transfer within the moving environment, the mechanical oscillations of the equipment on the ground, the oscillations in the flying devices, taking into account the changing amounts of fuel in the tanks - give the researcher certain advantages as compared to the tests of the actual objects in field conditions (in the selection of the system size, in the rate of the process development, in the physical environment, in the operational parameters), in investigating the above-mentioned functional processes associated with the spacecraft and tested on the ground, the method of scientific modeling turns out to be the only means making it possible to reproduce qualitatively and quantitatively the processes and events and to extend the obtained data to the actual test objects.

Such research presupposes the solution of two essentially independent problems:

the reproduction on the ground of the phenomenon (of a process) which is similar to the natural phenomenon or process (modeling);

the accomplishment within such process or event of all necessary observations and measurements.

The modeling methods make it possible to establish the requirements to which a model must correspond, so that the processes /181 which take place within it would be similar to the processes which occur in natural conditions, in other words, the modeling is essentially a replacement of the study in environmental conditions by the study of an analogous phenomenon by using a model of smaller or larger size and utilizing special (laboratory) conditions. The essence of modeling lies in the fact that on the basis of experimental results and using models one can generate the necessary responses as to the character of effects and of various quantities, related to the phenomena, as found in natural environmental conditions.

The modeling may be of two types - physical and mathematical. In the course of physical modeling, the phenomenon which is investigated in the natural conditions is being reproduced by using a model and the physical nature and geometric semblance is being retained. In such case the model differs from the natural conditions in size, in the rates at which the processes of interest are being investigated, etc. Under certain conditions of physical modeling, some departures from the physical nature of the studied phenomenon are permitted. For example, one can utilize for the model the material which differs from the data and material found in nature.

In the case of mathematical modeling, the physical processes are studied by investigating the analogous phenomena of different physical content but described by the same mathematical equations. The mathematical modeling is accomplished by using the computing devices of continuous type (the analog assemblies or modeling machines) making use of the whole complex of numerous physical devices - models, capable to simulate various phenomena. In the course of mathematical modeling one makes use of known analogies between the electrical, mechanical, thermal, hydrodynamic, acoustic, diffusion and other physical phenomena.

As one can see, the mathematical modeling is a technical realization of the analogies in order to obtain the numerical values for the phenomena found in nature, on the basis of the data obtained from a model. The mathematical model itself handles in essence the numerical solution of the equations which describe the natural phenomena, in other words, it is being used as a computing instrument (a mathematical device).

The mathematical modeling is extensively used for various processes (in particular, for the control processes) and is being accomplished by using the digital computers (the machines of discrete type). In the majority of cases, one also takes into account and simulates the random factors. Therefore, such method is being called ordinarily the method of statistical modeling.

All types of mathematical modeling have one very important and characteristic feature: the need for a preliminary compilation of the mathematical description of the process in question or the development of its mathematical model, which is the same. In addition, by mathematical model, one should understand the totality of relationships, formulas, equations, inequalities, logic operators which connect the parameters of the process in space and time, having the parameters which describe the properties of such system and its components, relating all of it to the initial data and the conditions which define the initial state of the system.

The mathematical model is constructed on the basis of objective physical relationships, reflecting the actual structure of the process which is being investigated, in accordance with the principle of structural similarity. It permits the use of new experimental /182
data for verification purposes, for the refinement work and for the develoment of the model itself. The mathematical model has certain boundaries of use which are expressed in the form of clearly formulated conditions.

It is apparent that the mathematical models make it possible to handle the equations which describe the processes and thus obtain the specific answers within the framework of assumptions which are made while compiling the initial set of equations. In this regard, the physical modeling has certain advantages. In such modeling, in parallel with determining the general parameters of the processes which occur at different systems and at different levels of the parameter change, it is possible to verify the basic theoretical postulates and assumptions which were used in compiling the set of equations, describing the processes which are being investigated and also to refine the computational formulas, etc.

In modeling the physical processes which are characterized by a large number of variables, it is necessary to determine the hidden relationships between the quantities since the multiplicity of special interrelationships do not reflect the internal properties of the processes which are associated with their physical nature (the effect of physical quantities is manifesting itself in a complex or comprehensive form rather than separately).

The switching from ordinary physical quantities to generalized arguments (coordinates, complexes) brings about the following advantages of the experimental study: the decreased number of variables (of varied factors), more clearly defined internal relationships, inherent in the physical processes under investigation, the data volume in numerical values of the generalized argument is increased, the time and material expenditures to conduct the experiments is reduced and the probability of finding the optimal solution is increased, etc.

The utilization in the study of compelx variables, instead of initial variables (independent physical ones), increases the range of its application, which is one of the major qualitative criteria in any research.

In developing the complexes, it is necessary to have the process equation

$$f(x, y, z, \theta, \ldots, \xi) = 0, \tag{16.1}$$

where x, y, z, θ, ..., ξ are the initial quantities with dimensions which may be written in the following form

$$f(\pi_1, \pi_2, \pi_3, \ldots, \pi_m) = 0, \tag{16.2}$$

where π_1, π_2, π_3, ..., π_m are the dimensionless generalized arguments (coordinates, complexes) compiled on the basis of the initial, independent variables involved in the process.

Let us note that the number of variables in the expression (16.2) is considerably smaller than in the expression (16.1). To define the final appearance of the (16.2) function, an experiment is required (for some additional functional relationships).

One could formulate the generalized complexes by the similarity method or by the method of dimensionality analysis.

The similarity method is used in the following cases:

in finding the analytical solutions;

in processing the measured data when the process results are /183
in the form of invariant relationships;

in modeling the processes (or phenomena).

Let us assume that we have an equation which characterizes the process which is being investigated:

$$a_1 + a_2 + a_3 + \ldots + a_n = 0, \tag{16.3}$$

where a_1, ..., a_n are the homogeneous summands, each of which characterizes one defininte physical effect.

The switching to dimensionless summands is possible by compiling paired relationships of a_2/a_1, a_3/a_1, ..., a_n/a_1, a_3/a_2, a_n/a_2 etc. type. It is clear that the number of dimensionless summands (which are mutually irreducible) corresponds to n-1 and in terms of their physical essence, each of them is an intermediate measure of relative intensity of two physical effects.

Third theorem: the experimental results must extend only to the similar phenomena, in other words, such phenomena in which the similarity conditions are unambiguous and the similarity criteria are similar.

As one can see, the similarity theory enables us, without integrating the differential equations which describe the physical phenomena, to obtain from such equations the similarity criteria and to establish the criterial relationships which are valid for all processes which are similar. /184

The method of dimensionality analysis of the physical quantities which are significant for the phenomenon (or process) which is being studied and which are used in such cases when it is impossible to lay as a basis of considerations, one or several basic equations and it is necessary to find the dimensionless complexes which will acquire the values of generalized variables. If a quantity acquires a zero dimension value - it becomes dimensionless. Such quantities are invariant during the metric transformations (by using the analogy with similarity transformation). It would be useful to recollect at this time the Buckingham π theorem: the number of dimensionless complexes m is equal to the number of all physically different quantities n, which are significant for the process, minus the number of primary quantities p:

$$m=n-p. \tag{16.6}$$

The formation of dimensionless complexes in this case may be accomplished either by paired combinations, in the form of relationships of physically homogeneous quantities (simplexes) or by grouping physically heterogeneous quantities on the basis of zero dimension [63]. Let us remind the reader that the system of algebraic linear equations may be solved if the number of equations corresponds to the number of unknowns. In practical terms this requirement is satisified when the number of quantities which are introduced into the complex is larger than the number of dimensionalities, when the number of quantities which are introduced into the complex are larger than the number of dimensionalities by one - the dimensionalities with respect to which one first compiles and then solves the system of equations itself.

By using the π theorem, one can establish a system of dimensionless complexes - the similarity critera which characterize a given physical phenomenon. The number of physical quantities n, which are significant for the process, depend on the degree and nature of the physical phenomenon idealization as it is being studied. Therefore, as one speaks of the totality of similarity criteria π_i (i=1, 2, ..., m), one can only refer to the accepted idealized schematic description of the phenomenon. It is clear that depending on the system of complexes selected π_i (i=1, 2, ..., m), one can always switch over to another equivalent system of complexes by combining the quantities which have dimensions in a different fashion. In each concrete case, the selection of one of the possible systems of dimensionless complexes must be conducted on the basis of physical considerations, related to the essence of the phenomenon which is being investigated. Therefore, to give a general recipe for the selection of a specific system of complexes is impossible.

To summarize, by using the dimensionality analysis apparatus, the problem of structure of the generalized variables may be solved by using the following flow chart:

one first determines the problem type and then selects the dimensionality system;

one compiles the list of quantities which are essential for the process (including the constants which have dimensions);

one determines the number of similarity criteria (as a difference between the total number of quantities and the number of primary quantities);

then on transforms into the power complexes the dimensional formulas;

now one excludes the primary quantities which were not incorporated into the list of quantities which are essential for the process. /185

The stage of solving a problem related to the application of dimensionality concepts is thus terminated. All remaining stages (isolation of complexes - arguments and of the complexes-functions, the construction of parametric criteria) are reduced to simple technical operations.

Let us note that of principal importance here are the first two moments within this flow chart. The quality of the solution on the whole depends on how well the first two questions have been handled. It should be emphasized that the applicability of the similarity theory depends on how correctly (in analytical sense) the problem is set up, or at least one must assess and compile a system of fundamental equations and then formulate the conditions of the solution uniqueness.

An important feature of the dimensionality analysis as a form of generalized analysis is the fact that it is supported by the apparatus which does not require the application of the problem equations. To utilize the dimensionality analysis it would suffice to know in which system of generalized relationships are the quantities involved, which are essential for the process.

The problem equations represent actually an accurate quantitative expression of the model used, to investigate a given process. The processing of these equations by using the similarity theory apparatus make it possible to determine the generalized variables in the form of complex quantities, combined into one whole, using precisely such relationships which in the inexplicit form exist in the problem equations. This direct approach is completely closed for the dimensionality analysis, which is independent of the possibility of constructing a detailed process model which would be sufficient to compile the set of equations. The dimensionality analysis apparatus has at its disposal the means making it possible to determine the generalized variables on the basis of the relationships of much more general nature, in the form of dimensionality formulas (the determinant equations).

As once can see, the essence of difference between the similarity theory and the dimensionality analysis lies in the fact that the similarity theory apparatus is applied to the equations of a process, while the dimensionality analysis apparatus is applied to the dimensionality formulas.

Unquestionably, if the utilization of similarity theory is possible, it should be a preferred approach because its apparatus is somewhat simpler. In addition, within the framework of the similarity theory, one can elucidate the physical meaning of all criteria. If the equations within the problem are not known, then the utilization of dimensionality analysis becomes unavoidable. Under such conditions, one is not always certain in the error-free compilation of a list of essential quantities, and the correctness of the dimensionality system which is being used. However, in such cases when the dimensionality analysis is based on an accurately preselected totality of existing quantities and a faultlessly compiled system of determinant equations (the dimensionality formulas), it becomes the research tool which is not inferior to the similarity theory in terms of its completeness and concreteness of the obtained results.

By using the dimensionless variables one can investigate not a singular special case, but an infinite number of different cases, combined by a certain commonness of properties. Therefore, the new variables, in their essence, are the generalized quantities. The ordinarily used names for the methods - the similarity theory and dimensionality analysis, define primarily the technical aspects /189 of the generalized analysis, rather than its fundamental essence. Therefore, the term "method of generalized variables" would be more accurate.

The practical application of the method of generalized variables forfine-tuning and maximizing on the ground of the spacecraft which are to descend toward the planet or to land on it, is presented in

TABLE 16.1

Physical conditions on the planets and basic properties of their soils

Parameters	Characteristis of the atmospheres: Earth			Venus			Jupiter	
	$H=0$ km	$H=11$ km	$H=31$ km	$H=0$ km	$H=30$ km	$H=54$ km	$H=0$ km conditional	$H=20$ km
Force of gravity, m/s^2	9,81	9,77	9,71	8,80	8,71	8,645	26,20	26,2
Temperature, K	288	217	228	750	502	285	171	211
Pressure, Pa	$1 \cdot 10^5$	$0{,}23 \cdot 10^5$	$1 \cdot 10^3$	$99{,}7 \cdot 10^5$	$10{,}6 \cdot 10^5$	$0{,}6 \cdot 10^5$	$1 \cdot 10^5$	$1{,}93 \cdot 10^5$
Density, kg/m^3	1,225	0,36	$1{,}6 \cdot 10^{-2}$	67,3	11,1	1,11	0,16	0,25
Carrying capacity of soil, Pa	–	–	–	–	–	–	–	–
Coefficient of friction	–	–	–	–	–	–	–	–
Chemical composition of the atmosphere and structure of soil		N_2 – 78% O_2 – 21% Ar – 1%			CO_2 – 97% N_2 – 2% O_2 – 0,4% H_2O – 1%		H_2 – 90% He – 10%	
Means used in the study	Direct studies in the atmosphere			Spacecraft of Vega, Venera, Pioneer-Venus and Mariner-10 type		Aerial balloon probes Vega	Potentially promising large aerial balloons, spacecraft Voyager and spacecraft Pioneer-10 and Pioneer-11	

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TABLE 16.1 (Second page)

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Parameters	Characteristis of the atmosphere: Mars, $H=0$ km	Mars, $H=10$ km	Mercury, $H=0$ km	Properties of the surface layer (of soil): Moon	Venus	Mars	Mercury	Earth (for comparison)
Force of gravity, m/s^2	3,88	3,88	3,71	1,62	8,80	3,88	3,71	9,81
Temperature, K	208	202	150–690	123–403	750	173–298	250–673	288
Pressure, Pa	$6{,}55 \cdot 10^2$	$2{,}55 \cdot 10^2$	10^{-6}	$1{,}33 \cdot 10^{-9}$	$99{,}7 \cdot 10^5$	$(6 \div 13) \cdot 10^2$	10^{-6}	$1 \cdot 10^5$
Density, Kg/m^3	$1{,}6 \cdot 10^{-2}$	$0{,}65 \cdot 10^{-2}$	$1 \cdot 10^{-3}$	700–1620	2700–2900	1000–1650	800–1500	1000–3000
Carrying capacity of soil, Pa	–	–	–	$(0{,}2 \div 15) \cdot 10^5$	$2 \cdot 10^5 \div 4 \cdot 10^5$	$\sim 1 \cdot 10^5$	$\sim 1 \cdot 10^5$ (?)	$(0{,}5 \div 5) \cdot 10^5$
Coefficient of friction	–	–	–	$0{,}25 \div 0{,}55$	0,3–0,5 (?)	0,2–0,5	0,2–0,6 (?)	$0{,}2 \div 0{,}65$
Chemical composition of the atmosphere and structure of soil	CO_2 – 95% Ar – 2% N_2 – 3%		CO_2 – 10% N_2 – 90%	Small grain with admixture of gravel & rocks	Compressed grainy material (regolith) resembles basalts	From porous to high density (hard rocks)	Similar to mountain rocks from the Moon	Loam, hard rocks, tufs, basalts
Means used in the study	Spacecraft Mariner 6,7,9 and Mars type		Mariner-10	Spacecraft of Luna type, Lunokhod, Surveyor, Apollo	Spacecraft of Venus, Vega type	Spacecraft of Mars, Viking type	Space craft Mariner-10, radio physical observations	Soil studies in different areas

TABLE 16.2

BASIC PROPERTIES OF THE SOILS-ANALOGUES AND THE MODELS OF PLANETARY SURFACE LAYER

Names of the analogues and the soil models	Density, kg/m^3	Carrying capacity, Pa	Coefficient of friction	Soil structure
Model A	400–900	$(0,2 \ldots 1,0)\,10^5$	0,17–0,28	Aglophorite sand $\theta \leq 10^{\circ}$, $\delta \geq 1$ m Base-either concrete of foam concrete $\delta \geq 0.5$ m
Model B	400–1000, 1500	$(0,2 \ldots 1,0)\,10^5$, $\leq 150 \cdot 10^5$	0,2–0,4	Model A + pieces of crushed rock ⌀ 30-80 mm per each layer
Model C	400–1000, 2800	$\leq 150 \cdot 10^5$, $(1 \ldots 3) \cdot 10^8$	0,2–0,6	Model B + rocks made of foam concrete and granite (150X50X300 mm^3) (see p.)
Model D	700–2000	$(2 \ldots 3) \cdot 10^6$	0,4–0,65	Foam concrete blocks, $\theta = 0^{\circ}$ and 10° $\delta \geq 0.5$ m Concrete base $\delta > 0.5$ m
Gas-silicate calcite	650	$1,25 \cdot 10^5$	0,2–0,3	Man-made analogue
Sand+ mineral oil	1420	$0,29 \cdot 10^5$	0,35	Ibid.
Foam concrete	700	$0,78 \cdot 10^5$	0,4	Ibid.
Andesite-basalt volcanic sand	1290–1620	$0,71 \cdot 10^5$	0,25–0,65	Man-made analogue
Basalt (crushed)	1040–1890	$0,79 \cdot 10^5$	0,2–0,5	Ibid.
Andesite-basalt lava (dense, porous)	2600 / 2150	$2 \cdot 10^5 / 0,5 \cdot 10^5$	0,2–0,5	Ibid.
aglophorite sand	400–900	$(0,2 \ldots 1,0) \cdot 10^5$	0,17–0,28	Man-made analogue
crushed rock (⌀ 30-80 mm)	1000–1500	$\leq 150 \cdot 10^5$	0,3–0,5	Natural analogue
granite (separate rocks)	2800	$(1 \ldots 3) \cdot 10^8$	0,3–0,6	Ibid.

(Note: Commas in tabulated material are equivalent to decimal points.)

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the studies by the following authors [12, 15, 32, 59, 76], and also in the following studies [5, 42, 43, 63]. So as not to repeat ourselves, let us summarize the above-mentioned studies and reduce the obtained results to the table form. For example, the Appendix includes the definitive formulas and figures for several scale derivatives which are used to model the dynamics of the descending and landing devices, to study the Moon and planets of the Solar System.

The latest data about the properties of the planetary surface layer[1] and about the parameters of their respective atmospheres are presented in Table 16.1. On the basis of the objective data about the physical conditions on the planets, the planetary atmospheric models are being developed with the description of the surface layer, to optimize on the ground the spacecraft soft landing and the movement of it across the planetary surface. Table 16.2 includes the natural and man-made materials as the soil analogues, in other words the soils, the mechanical parameters of which are the same as the natural planetary soil.

Table 16.2 also shows the multilayered and single-layered models of the planetary surface layer, within the Earth group, embracing in terms of their parameters the whole range of properties of the soils on these planets (of course, except for such properties as albedo, color, polarization angle, which are of no fundamental importance in the dynamic process analysis). Such models were used successfully to verify the operational capabilities and efficiency of the landing devices of Mars type and especially, in testing the functioning of the Venera-9 - Venera-14 spacecraft.

As practice had shown, the study of the planetary soft landing and moving of the spacecraft across its surface, by using the soil deformations, makes it possible to obtain on the ground a total analogue with the natural event, evaluating with a great degree of reliability the magnitudes of overloads, deformations, sinking of the spacecraft supports into the soil and also the stable areas which have been determined a priori for the solid planetary layer.

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[1]In conjunction with the fact that the Moon, in terms of its physical parameters, is similar to the planets of the Earth type, in describing the planets throughout this text, we also have in mind the Moon.

CHAPTER 17.

PHYSICAL MODELING OF SEPARATION, ATMOSPHERIC ENTRY AND LANDING OF THE SPACECRAFT ON THE PLANETS

The process of spacecraft descent, designed to investigate /189
the atmosphere and planetary surfaces within the Solar System
embraces the following fundamental stages: separation from the /190
space station, entry into the planetary atmosphere, activation of the parachute and the aerodynamic braking of the spacecraft, prior to its contact with the planetary surface.

We shall consider below the specifics of the tests, pertaining to the descending devices and modeling on the ground of their functioning at the above-mentioned stages.

Experimental fine-tuning of the components which are incorporated into the separation system is being conducted within a specific sequence and is subdivided into research, finalized delivery, reception and control tests (see Figure 14.3 and National Bureau of Standards, item 16504-81).

The research tests are the first stage of experimental optimization of separate components in the separation system - explosive charge gates and locks, explosive charge rods and other pyro devices. The purpose of these studies is to determine the design shortcomings in the test component. As a rule, these studies are being conducted at the ambient temperature.

After the determined defects have been corrected, one commences with the final, before the delivery, tests, which are the basic ones. During such tests, one determines the proper relationship between the design component within the framework of the project design and within the range of assigned reliability level.

The final tests before the delivery include, as a rule, the vibration tests, impact strength, the effect of increased and decreased temperature, the strength tests, the long-term vacuum exposure and the transport tests.

The impact tests, vibration tests and transport tests include the total cycle of dynamic testing. In the course of the tests which simulate transporting of the spacecraft, one exposes the design components to cyclic loads which correspond to the transporting of the components and the whole spacecraft, by either using highways or railroads, by using the water transport and aircraft, with the deliveries from the manufacturer to the launch pad. In the course of such testing, one tests the torques of the threaded components.

In the course of impact testing, one simulates the application of abrupt loads and overloads on the specific design components. In the vibration tests, one simulates the in-flight vibration modes. These tests may be of two types: the effect of sinusoidal vibrations and the effect of random vibrations. The tests of sinusoidal vibrations is carried out in accordance with the technical requirements in terms of the frequency and maximum load factor. In studying the effect of low frequency oscillations (10-50 Hz) one could utilize the mechanical devices and hydraulic vibration stands. The high frequency oscillations (50-200 Hz) are generated by using the electromagnetic devices.

Ordinarily the random vibrations are generated by using the noise oscillator, whereby the signal is enhanced and then passed through the filtering system which develops the amplitude-frequency characteristics. The resulting signal is additionally enhanced and then is used to activate the vibration stand. The design component /191
is subjected simultaneously to multifrequency vibrations within an assigned frequency band and at a certain amplitude level.

The tests of increased and decreased temperature simulate the actual conditions of the spacecraft flight. In this case, the design component which is to be optimized is exposed for a certain length of time to a temperature, then the component is removed and activated. The time between the design component removal from the temperature environment and activation must be minimal. The temperature range in which such tests are to be conducted is as a rule between 210-330 K.

Practically all design components which are incorporated at the junction points of the spacecraft undergo testing for the long-term vacuum exposure. The assembly which is to be optimized is placed into the evacuated chamber and kept there for a certain length of time, which should be no less than the operational time in space, and then, the component or assembly is removed from the chamber and is activated. The purpose of such tests is to study the effect of a vacuum on the operational capability of the assembly or element, and also to study the changes of mechanical parameters.

In testing the explosive charge locks, in parallel with such mechanical paramters as, for example, the force of unfolding and limiting strength, an important parameter is the time required for the lock activation which is measured from the moment when the command is given until the time when the lock is activated. A large difference in the explosive charge lock activation may result in the load redistribution at the junction points and as a result, one of the assemblies may be overloaded, with an additional perturbation and change in angular velocity of the part of the spacecraft which is to be separated.

The major mechanical parameter of the explosive charge rods or plungers is the work which is to be performed. It is being determined on the assemblies of either pendulum or track type, and then is recalculated in terms of specific environmental conditions.

The next optimization stage consists of the tests during the assembly reception. These are being conducted to substantiate that the components of the separation system correspond to the assigned requirements. The technical specifications which are clearly defined during the reception tests must correspond to the specific design component which is being optimized, it involves all components which are to be tested, it involves the sequential testing and selection of components from the component batch.

The concluding stage of separate component optimization are the control tests.

The comprehensive or complex tests are the most complicated and expensive stage in the experimental fine-tuning of the separation systems. In the course of such tests, one determines the basic kinematic and dynamic parameters of separation, both the linear relative velocity, the angular velocities of the bodies which are to be separated, being acted upon by all external forces and forces associated with the activation of the components which are located at the junction points. One also determines here the vibration loads which are generated while the explosive charge equipment and mechanical components of the system are activated. On the basis of such test results, one draws a conclusion as to the applicability and acceptance of a given assembly which is to be incorporated into the spacecraft. /192

Modeling of the dynamic processes as the spacecraft enters the planetary atmosphere may be realized by using a centrifuge and special oscillation stands.

The centrifuge makes it possible to reproduce the quasistatic active loads to which the object is to be exposed. By utilizing them, one can simulate the load functions for comparatively large component parts of the spacecraft. Each separate load may be reproduced at a specifically defined orientation of the object which is being investigated, within the centrifuge. This corresponds to only one specific position among the multitude of positions and does not provide a dynamic change in the overload vector.

In the actual use of the spacecraft, the external loads change in time and the actual values of the carrying capacity will change correspondingly. Consequently, it is necessary to combine the quasistatic and dynamic loads.

In simulating on the ground the entry of the descending spacecraft, into the planetary atmosphere, in order to bring about the simultaneous effects of two loads - the changing axial overload and the sign-changing overloads in the perpendicular direction - it is necessary to use special test stands which make it possible to generate the oscillatory overloads of the required frequency and amplitude.

The major distinguishing feature of the quasidynamic load is the introduction of dynamic effects which make it possible to evaluate experimentally the effect of low frequency oscillations on the carrying capacity of the construction design, to refine the calculations, to reduce the load time exposure and to upgrade the synchronization in the test load components, improving the accuracy of carrying capacity determination.

The major operational parameters of the centrifuge are the centrifugal acceleration, the acceleration gradient, the lifting capacity and the factor of conditional power output.

In the centrifuge testing of the spacecraft, the test of the overload change is provided by two methods:

by changing the rotational speed ω of the centrifuge central beam and by moving the assembly which is being tested along the central beam (in other words, by changing the rotational radius R).

The fundamental diagram of such test assembly is shown in Figure 17.1.

The equations of motion and of moments may be represented in the following form

$$\left.\begin{aligned} -mY_c\epsilon - mX_c\omega^2 &= \Sigma N^{a}_{X} + R_{aX} R_{BX}; \\ mX_c\epsilon - mY_c\omega^2 &= \Sigma N^{a}_{Y} + R_{aY} + R_{BY}; \\ I_{ZX}\epsilon + I_{YZ}\omega^2 &= \Sigma M^{a}_{X} - R_{aY}h; \\ I_{YZ}\epsilon - I_{ZX}\omega^2 &= \Sigma M^{a}_{Y} + R_{aX}h. \end{aligned}\right\} \qquad (17.1)$$

So that the responses at the supports of the assembly would not be dependent on ω and ε, it is necessary that the following conditions would be satisfied: $X_c=0$, $Y_c=0$, $Y_{YZ}=0$ and $Y_{ZX}=0$. If the first two conditions are satisfied, then the rotation system is statically equilibrated and when the two last conditions are satisfied - the system is dynamically equilibrated. /193

Keeping in mind the limitations of the working radius in the existing test assemblies, with a high permissible load parameter $(aM)\geq 3000$ kN, the tests of large size objects which are to be investigated must be conducted in several stages, in the following manner:

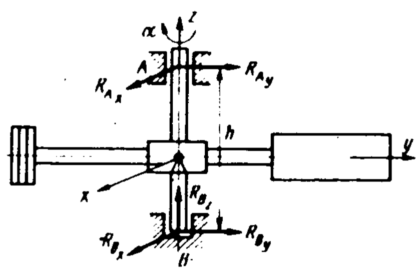

Figure 17.1 Fundamental schematic diagram of the centrifugal test assembly.

if it is possible, the tests object must be split and the tests must be conducted section-by-section, with the subsequent checkout of the junction points in terms of the nominal test load factors;

if it is impossible to separate the object which is being investigated into several parts, one must load it by nominal test overloads, starting with the maximum specific overload with respect to axial rotation and the subsequent checkout of all other axial positions.

The major technical parameters of a typical test assembly are presented below.

Parameter	
Mass of the test assembly with stands, tons	1.5
Maximum overload gradient, s^{-1}	10
Permissible load, MN	3.2
Dimensions of the assembly, m:	
Length	2
Diameter	2
Maximum rotational speed of the shaft, rpm	345
Working length of the arm, m	1.6-3.5
Working zone of the test assembly, m:	
Height of enclosure	2.5
Diameter of enclosure	9.0
Dimensions of the test assembly, m	13X12X6

The centrifuge assembly drive power output W is being expanded to overcome the moment of internal friction between the moving parts M_f, to overcome the moment of inertia M_i and the moment of aerodynamic resistance M_a.

The calculations and experimental results indicate that as the test assembly begins to function with the large overload factors (up to n=300 ... 400), where the possibilities of the centrifuge are limited by its power output, the losses which are due to the internal friction are 1-5%, the losses associated with the inertia force are 15-25% and the losses associated with the aerodynamic resistance are 60-80% of the total power output. From the energy point of view, in the presence of specific overloads, it is more advantageous to conduct the testing by using a smaller radius, but at somewhat higher rotational velocity.

The modeling method by using a quasistatic oscillation stand with free oscillation of the system was used to simulate the Venus atmospheric entry of the interplanetary automated stations Venera-8 - Venera-14.

The basic initial data is shown in Table 17.1 /194

TABLE 17.1

Parameter	Venera-8	Venera-9 - Venera-14
Mass of the module, kg	115-160	200
Maximum oscillation amplitude during entry, rad	0.35	0.30
Maximum operational overload	up to 350	210-224
Oscillation frequency, Hz	5-6	2-4
Lateral operational overload	120	70
Logarithmic decrement of oscillations	0.15	0.20

The mechanical system has one degree of freedom and oscillates freely with the linear attenuation, because of the resistance forces, the nature of which is different: the air resistance, friction at the hinges, internal friction between metal parts, etc. The nature of this relationship may be established only approximately. For the sake of simplicity, let us assume that the resistance force is proportional to the first power of angular velocity $\dot{\phi}$.

Let us determine the natural frequency of the oscillatory system during tests.

The oscillatory system (of M mass) is connected with the carrying frame by means of hinges. The system is deflected from the equilibrated position by ϕ angle. The oscillatory system takes part in two motions - a relative motion around the hinge and the movement together with the frame. The angle between the frame longitudinal axis and the line, passing through the center of mass of the system, and the hinge in the plane of oscillations, will be designated by ϕ_1, while the distance from the axis to the hinge is R, the length of the pendulum is ℓ and the distance from the axis to the center of the system mass will be designated by L. Then

$$L = \frac{R + l\cos\varphi}{\cos\varphi_1},$$

and the force of inertia is

$$F = M\omega^2 L = M\omega^2 \frac{R + l\cos\varphi}{\cos\varphi_1}. \qquad (17.2)$$

The differential equation of relative motion of the system (device-stand frame) has the following form

$$-FR\sin\varphi_1 - k_1 l\dot{\varphi} = (I_1 + Ml^2)\ddot{\varphi},$$

where I_1 is the natural moment of inertia, k_1 is the coefficient of proportionality between the force and velocity. By substituting the expression (17.2), we will obtain

$$M\omega^2 R(R + l\cos\varphi)\operatorname{tg}\varphi_1 + kl\dot{\varphi} + (I_1 + Ml^2)\ddot{\varphi} = 0. \qquad (17.3)$$

But since $\operatorname{tg}\varphi_1 = \frac{l\sin\varphi}{R + l\cos\varphi}$, the equation (.7.3) may be represented in the following form

$$\ddot{\varphi} + \frac{k_1 l}{I_1 + Ml^2}\dot{\varphi} + \frac{M\omega^2 Rl}{I_1 + Ml^2}\sin\varphi = 0. \qquad (17.4)$$

The angular velocity of natural oscillations in a real system /195

$$\omega_0 = \sqrt{\frac{ngMl}{I_1 + Ml^2}} = \omega\sqrt{\frac{MRl}{I_1 + Ml^2}}.$$

By designating that $\frac{k_1 \ell}{I_1 + M\ell^2} = 2k$ and taking into account that the angles of oscillation are small, the equation (17.4) may be written in the following form

$$\ddot{\varphi} + 2k\dot{\varphi} + \omega_0^2\varphi = 0. \qquad (17.5)$$

Three following cases are possible here:

$k<\omega_0$ is the case of small resistance - the system produces receding oscillations;

$k>\omega_0$ is the case of great resistance - the system produces aperiodic motion;

$k=\omega_0$ which is case if critical resistance and this is the limiting case of aperiodic motion.

The actual stand system represents the "a" case.

The equation (17.5) will have in such case the following form

$$\varphi = A e^{-kt} \sin(\sqrt{\omega_0^2 - k^2}\, t + \varphi_0). \tag{17.6}$$

The system of equations which define the motion of the device on a stand will be written in the following form:

$$n_X = \frac{\omega^2 (R+X)}{g} + \frac{Y}{g}\frac{d^2\varphi}{dt^2} + \frac{X}{g}\left(\frac{d\varphi}{dt}\right)^2; \quad \varphi = A e^{-kt}\sin(\sqrt{\omega_0^2 - k^2}\, t + \varphi_0);$$

$$n_Y = \frac{\omega^2 (R+X)}{g}\varphi + \frac{X}{g}\frac{d^2\varphi}{dt^2} - \frac{Y}{g}\left(\frac{d\varphi}{dt}\right)^2; \quad n_\Sigma = \sqrt{n_X^2 + n_Y^2}.$$

The moving system produces exponentially receding oscillations (Figure 17.2). The periodicity here, in a strict sense, is absent, since there is no complete simulation of motion. On the other hand, the passages of ϕ coordinate through zero occur at similar time intervals T*. The T* magnitude is a conditional period.

The times when the ϕ displacements are either maximal or minimal do not coincide with the central time periods between the subsequent passages of ϕ through zero but are shifted from these values toward the beginning of coordinates by

$$\Delta t = \frac{1}{\omega_0} \operatorname{arctg} \frac{k}{\omega_0}.$$

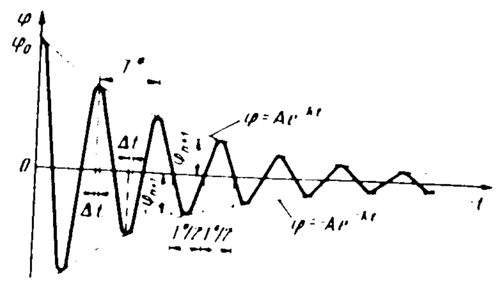

Figure 17.2. Exponentially receding oscillations.

It follows from the expression (17.6) that

$$k = \frac{\omega_0}{2\pi} \ln \frac{\varphi_n}{\varphi_{n+1}},$$

where ϕ_n and ϕ_{n+1} are the lagging displacements in the direction of the time increase during a complete conditional time period. /196

The period or receding oscillations may be determined by using the following formula

$$T^* = \frac{2\pi}{\sqrt{\omega_0^2 + k^2}}.$$

The resistance increases the period of free oscillations but only slightly. It affects much more the reduction of amplitude. For example, when $k=0.05\omega_0$, the resistance increases the period by 0.125% and the amplitude during one full oscillation decreases by more than 25%.

The attenuation coefficient is

$$\psi = \frac{Ae^{-kt}}{Ae^{-k(t+T^*/2)}} = e^{\frac{kT^*}{2}}.$$

To estimate the rate of the amplitudinal decrease it is convenient to use the logarithmic decrement of oscillations

$$\lambda = \ln \psi = \frac{kT^*}{2} = \ln \frac{\varphi_n}{\varphi_{n+1}}.$$

After conducting a number of transformations, one can obtain the relationship between the period and the logarithmic decrement of oscillations

$$\frac{T^*}{T} = \sqrt{1 + \left(\frac{\ln \psi}{\pi}\right)^2}.$$

The second case is the aperiodic motion (strong attenation).

If $k>\omega_0$, the general solution of the equation (17.5) will have the following form

$$\phi=Ae^{-kT}\mathrm{sh}(\sqrt{k^2-\omega_0^2}t+\phi_0).$$

The depiciton of aperiodic motion is shown in Figure 17.3.

The third case is the limiting aperiodic motion (critical attenuation).

When $k=\omega_0$, the general solution of the equation (17.5) will have the following form

$$\varphi=e^{-kt}(C_1t+C_2). \qquad (17.7)$$

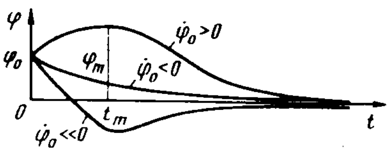

Figure 17.3. Typical relationship during aperiodic motion.

After determining C_1 and C_2 on the basis of the initial conditions of motion, the equation (17.7) may be represented in the following form:

$$\varphi=e^{-kt}[\varphi_0+(\dot\varphi\ +k\varphi_0)t].$$

The general character of the aperiodic motion of the system is defined by the fact that in the presence of unlimited increase of t, the ϕ coordinate approaches zero.

The test results of the parachute and instrument compartments in the spacecraft of Venera type, by using the quasistatic oscillation stand, with free oscillations of the system, are shown in Table 17.2.

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Technical Characteristic of the Quasistatic Oscillation Stand With Free Oscillation System

Maximum load factor n_Σ	360
Gradient of the change in the load factor, s^{-1}	10
Frequency of oscillations of the frame f, Hz	2.0-8
Maximal amplitude of oscillations ϕ, rad	±0.35
Duration of tests in the oscillatory mode, 5, s . . .	2-3
Maximal lateral load factor n_Y, n_Z	130

Logarithmic decrement of oscillations λ	0.15-0.30
Maximal mass of the objects which are being tested M,kg	200
Dimensions of the objects being tested, m:	
diameter	0.8
length L	0.5

The simulation method, by using the quasistatic oscillation stand, with the forced oscillation of the system has been used to simulate and optimize the Martian atmospheric entry of the modules or devices of Mars-6 type.

Basic starting data:

Mass of the compartment M, kg	280
Dimensions, m:	
diameter	2.0
length	0.4
Maximum amplitude of oscillations during entry ϕ, rad	0.35
Maximum operational load factor $n^{o}_{X\,max}$	35
Gradient of change in the load factor s^{-1}	1-1.5
Frequency of oscillations f, Hz	1.5
Lateral operational load factor $n^{o}_{Y} \approx n^{o}_{Z}$	9.5
Duration of oscillatory process t, s	30

Let us assume that our actual system has one degree of freedom and must oscillate with the frequency of natural oscillations.

A perturbation force, changing harmonically, is applied to the moveable system

$$P(t) = P_{max} \cos \omega_1 t,$$

where P_{max} is the maximum force, ω_1 is the angular frequency of the force change.

The differential equation of the system's motion is

$$\ddot{\varphi} + 2k\dot{\varphi} + \omega_0^2 \varphi = \frac{P_{max}}{M} \cos \omega_1 t. \quad (17.8)$$

TABLE 17.2

Parameter	Venera-8		Venera-9 - Venera-14
	Parachute compartment	Instrument compartment	Instrument compartmnet
Mass of compartment, kg	158	115	216
Maximal load factor at center of mass	330–335	347–358	223–226
Rotational radius of center of mass, m	2,72	2,66	2,70
Frequency of the oscillations of the compartment, Hz	5,9–6,0	5,0–5,3	4
Lateral load factor at center of mass	118,5–121,0	124,2–128	73
Logarithmic decrement of oscillations	0,26	0,33–0,27	0,27
Number of computed oscillations	10	8	7

The general solution of the equation omitting the right hand part, when $\omega > k$ will have the following form /198

$$\varphi_1 = Ae^{-kt}\sin\left(\sqrt{\omega_0^2 - k^2}\, t + \varphi_0\right). \qquad (17.9)$$

The partial solution of the nonhomogeneous equation will be in the following form

$$\varphi^* = C_1 \cos\omega_1 t + C_2 \sin\omega_1 t.$$

By substituting this function into the equation (17.8) and equating the coefficients for cos ω_1t and sin ω_1t in the right and left hand sides, for the determination of C_1 and C_2 constants, we can obtain a system of equations

$$\left.\begin{aligned} &(\omega_0^2 - \omega_1^2)C_1 + 2k\omega_1 C_2 = 0,\\ &-2k\omega_1 C_1 + (\omega_0^2 - \omega_1^2)C_2 = \frac{P_{max}}{M}. \end{aligned}\right\}$$

From which

$$C_1 = -\frac{P_{max}}{M}\,\frac{2k\omega_1}{(\omega_0^2 - \omega_1^2)^2 + 4k^2\omega_1^2};$$

$$C_2 = \frac{P_{max}}{M}\,\frac{\omega_0^2 - \omega_1^2}{(\omega_0^2 - \omega_1^2)^2 + 4k^2\omega_1^2}.$$

After substituting C_1 and C_2, the expression for ϕ^* function will be rewritten in the following form:

$$\varphi^* = \frac{(\omega_0^2 - \omega_1^2)P_{max}}{M[(\omega_0^2 - \omega_1^2)^2 + 4k^2\omega_1^2]} (\sin\omega_1 t - \frac{2k\omega_1}{\omega_0^2 - \omega_1^2}\cos\omega_1 t).$$

By introducing the following designation

$$\frac{2k\omega_1}{\omega_0^2 - \omega_1^2} = \mathrm{tg}\beta$$

and taking into account that

$$\frac{1}{\cos\beta} = \sqrt{1 + \mathrm{tg}^2\beta} = \frac{\sqrt{(\omega_0^2 - \omega_1^2)^2 + 4k^2\omega_1^2}}{\omega_0^2 - \omega_1^2},$$

the partial solution will be represented in the following form

$$\varphi^* = \frac{P_{max}}{M\sqrt{(\omega_0^2 - \omega_1^2)^2 + 4k^2\omega_1^2}} \cos(\omega_1 t \quad \beta). \tag{17.10}$$

As one can see, the complete solution of the equation will have the following form

$$\varphi = Ae^{-kt}\sin(\sqrt{\omega_0^2 - k^2}\, t + \varphi_0) + \frac{P_{max}}{M\sqrt{(\omega_0^2 - \omega_1^2)^2 + 4k^2\omega_1^2}} \times$$
$$\times \cos(\omega_1 t \quad \beta). \tag{17.11}$$

The system of equations which define the movement of the module or device will have the following form

$$\left.\begin{aligned}
n_x &= \frac{\omega^2 (R+x)}{g} + \frac{y}{g}\frac{d^2\varphi}{dt^2} + \frac{x}{g}\left(\frac{d\varphi}{dt}\right)^2;\\
n_y &= \frac{\omega^2 (R+x)}{g}\varphi + \frac{x}{g}\frac{d^2\varphi}{dt^2} - \frac{y}{g}\left(\frac{d\varphi}{dt}\right)^2;\\
\varphi &= A e^{-kt} \sin\left(\sqrt{\omega_0^2 - k^2}\, t + \varphi_0\right) + \frac{P_{max}}{M\sqrt{(\omega_0^2-\omega_1^2)^2 + 4k^2\omega_1^2}} \times\\
&\times \cos(\omega_1 t - \beta); \quad n_\Sigma = \sqrt{n_x^2 + n_y^2}.
\end{aligned}\right\} \qquad (17.12)$$ /199

In the case under consideration, the moving system participates simultaneously in two oscillatory motions. The first one is the natural oscillatory motion, the amplitude and phase of which is defined by the initial conditions. These oscillations are of receding type and after a specific time interval, will practically disappear. The second oscillatory motion occurs with the frequency of the perturbation force and the phase shift. It will not decrease but will continue as long as the perturbation force is applied. These are the forced oscillations. The amplitude of forced oscillations may be written in the following form

$$A_f = \frac{P_{max}}{M\sqrt{(\omega_0^2 - \omega_1^2)^2 + 4k^2\omega_1^2}}.$$

The dynamic state factor will be

$$\lambda = \frac{A_f M\omega_0^2}{P_{max}} = \frac{\omega_0^2}{\sqrt{(\omega_0^2 - \omega_1^2)^2 + 4k^2\omega_1^2}}.$$

The dynamic state factor depends on two quantities: on the ω_1/ω_0 ratio which is the ratio of frequencies of the perturbation force and of the system's natural oscillations and of the attenuation factor k.

As the frequency of perturbation force begins to approach the frequency of natural oscillations, the dynamic state factor λ will rapidly increase. In actual conditions, because of the presence of the resistance forces, the λ factor will not become infinity, although it may attain high values. This means that in the presence of the resonance, just like within the whole resonance zone from $(\omega_1/\omega_0)=0.7$ to $(\omega_1/\omega_0)=1.3$, even comparatively small perturbation forces may cause considerable displacements within the oscillating system.

For a more complete simulation of the actual load factor conditions as the spacecraft enters the atmosphere, the simultaneous effects of the receding linear accelerations to which the module or device is exposed, the angular oscillations with respect to the centers of mass and the aerodynamic loads, the module or device, the shell of which has the burn-off thermal shield, should be exposed to the aerodynamic load. In such case, the quality of tests on the ground will improve, compared to the traditional static tests and the reliability of the module or device will improve, while in use. Because of simultaneous application of loads, the duration of tests on the ground is reduced (by a factor of 2-2.5) and the number of module prototypes which are to be tested on the ground will also be reduced.

Technical Characteristics of the Quasistatic, Oscillation Stand with Forced Oscillation of the System /200

Maximum load factor n_x	45
Gradient of the load factor increase, s^{-1}	3-5
Rotational speed of the frame, rad/s	6-10
Frequency of the frame oscillation with the object which is being tested, Hz	1-20
Maximum amplitude of oscillations, rad	±0.4
Duration of tests in the oscillatory mode, s . .	60
Lateral load factor n_y, n_z	10
Maximum mass of the objects being tested, kg . . .	300
Size of the objects being tested, m:	
diameter	2.0
length	0.4
Mass of the stand, kg	300
Size of the stand, m:	
diameter	1.5
length	1.5

The complexity of tests is due to the fact that the spacecraft is imersed into the reservoir, filled with liquid and rotating in the centrifuge.

The reproduction of angular oscillations of the device with respect to its center of mass is brought about by a spring. The required oscillation frequency of the device f, is achieved by changing the spring rigidity EI and its length L (Figure 17.4).

The device and spring are deflected from its neutral position by the ϕ angle. The device participates in three motions - the transfer motion together with the rotating reservoir, the relative oscillatory motion around the center of rocking C which is caused by the spring and the relative motion with respect to the reservoir slots.

The transfer force of inertia is

$$P = M_a \left(\frac{d\vartheta}{dt}\right)^2 R = M_a n_a g.$$

The law of the load factor change is assigned by the following function

$n_a = f(t)$.

The Coriolis force of inertia has the following form

$$F_c = 2M_a V_r (\dot{\vartheta} + \dot{\varphi}) \sin[(\dot{\vartheta} + \dot{\varphi})\widehat{\ } V_r].$$

In this case, the cofactor is $[(\dot{\theta}+\dot{\phi})\hat{\ } V_r]=1$, therefore $F_c = 2M_a \times V_r(\dot{\theta}+\dot{\phi})$.

The friction force will have the following form

$$F_{fr} = 2\mu_1 M_a V_r (\dot{\vartheta} + \dot{\varphi}) + \mu_2 P.$$

where $\mu_1 = f(V_r)$ is the coefficient of friction in the end journals, $\mu_2 = f(\dot{\phi})$ is the coefficient of friction of the device against the liquid.

The following force is acting on it by the liquid

$N = P(\phi)S$.

The force which activates the oscillatory motion of the device has the following form

$$F = cr \sin\varphi. \quad (17.13)$$ /201

where $c = \frac{3EI}{L^3}$ is the stiffness coefficinet and $EI = EI_0(1-\bar{x})^{\beta}$.

The kinetic and potential energies are determined by the following expressions

$$T = \frac{1}{2} I_a \dot{\varphi}^2; \quad \Pi = \frac{1}{2} cq^2 = \frac{1}{2} cr^2 \sin^2\varphi.$$

Let us utilize the Lagrange equation

$$\frac{d}{dt}\left(\frac{\partial T}{\partial \dot{q}}\right)\ \frac{\partial T}{\partial q} = Q.$$

Then

$$\frac{\partial T}{\partial \dot{\varphi}} = I\dot{\varphi};\ \frac{d}{dt}\left(\frac{\partial T}{\partial \dot{\varphi}}\right) = I\frac{d^2\varphi}{dt^2};\ \frac{\partial T}{\partial \varphi} = 0;$$

$$\frac{\partial \Pi}{\partial \varphi} = cr^2 \sin\varphi \cos\varphi;\ Q = -\frac{\partial \Pi}{\partial \varphi} = -cr^2 \sin\varphi \cos\varphi.$$

By taking into account the expression (17.13) we will obtain

$$I\frac{d^2\varphi}{dt^2} = -cr^2 \sin\varphi \cos\varphi = -Fr\cos\varphi.$$

The system of differential equations which defines the motion of the device will acquire the following form

$$\left.\begin{aligned}
&M_a\frac{dV_r}{dt} = M_a\omega^2 R \quad p(\varphi)S \quad F\sin\varphi \quad F_1;\\
&I_a\frac{d^2\varphi}{dt^2} = -Fr\cos\varphi = -\frac{1}{2}cr^2\sin 2\varphi;\\
&n_X = \frac{\omega^2(R+X)}{g} + \frac{Y}{g}\frac{d^2\varphi}{dt^2} + \frac{X}{g}\left(\frac{d\varphi}{dt}\right)^2;\\
&n_Y = \frac{\omega^2(R+X)}{g}\varphi + \frac{X}{g}\frac{d^2\varphi}{dt^2} - \frac{Y}{g}\left(\frac{d\varphi}{dt}\right)^2;\\
&n_\Sigma = \sqrt{n_X^2 + n_Y^2}.
\end{aligned}\right\} \qquad (17.14)$$

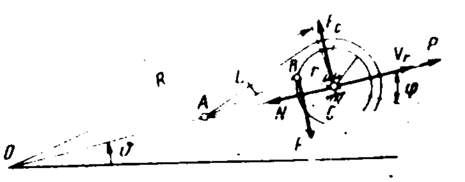


Figure 17.4. Computed schematic diagram of test stand.

The integration of these equations is conducted by using the Runge-Kutta method. While the main shaft of the centrifuge is rotating, the device will be exposed to the centrifugal force $P=M_a n_a g$, and the distributed force N will act upon the frontal surface of the thermal shield, from the side of the liquid.

The condition of the "attached device-liquid equilibrium" is /202

$$M_a n_a g = V_{a\ell} \rho_\ell (n_a + \Delta_{n\ell}),$$

where $V_{a\ell}$ is the part of the device immersed in the liquid, ρ_ℓ is the density of the liquid, $\Delta_{n\ell}$ is the difference in load factors for the immersed part of the device and for the center of its mass.

As the desired test mode is attained $n_u = f(t)$, the lock releases the spring which then forces the device to execute the angular oscillations around the journal axis.

To control the position of the device which is being tested and to read off the appropriate relationships, the reservoir and end journals mount the accelerometers

$$S = f(n, t); \ \varphi = f_1(n, t).$$

In testing the spacecraft device by using the quasistatic-oscillatory operational stands, the following sequence should be adhered to:

first one determines the mass of the device M, the coordinates of the center of masses X_{CM}, and the moments of inertia I_X, I_Y, I_Z;

now one checks out the vacuum in the compartments;

now one carries out the comprehensive testing of equipment;

now some auxiliary and additional assemblies are attached to the device;

now one adjusts the primary converters of the angle ϕ, of the load factors n, of the displacements L, of stresses σ, temperature T, etc;

the device which is being tested is mounted now on either a moveable frame or into the basket;

now one ensures the static and dynamic balancing of the oscillatory system;

the moveable system with the device which is being tested is now placed into the centrifuge;

now one selects the proper oscillation frequencies f^u and angles ϕ^u in the moveable system;

now one begins to generate (maximize) amplitude of the oscillations ϕ_0;

now one provides the static and dynamic balancing of the centrifugal assembly, while the oscillatory system is in the neutral position;

now one adjusts the control and measuring equipment;

now one selects the test mode;

now one generates the test operations $n_X = f(t)$, $n_Y(t)$, $P = f_1(n, \phi)$ and determines the measured parameters (ω, ϕ, f, n, L, σ, T);

now one disassembles the system, determines the defects and processes the data.

In testing the device, of particular importance is the overload distribution across its tridimensional outline.

The total overload, acting on the tested device at its center of mass may be determined in the following manner:

$$n_\Sigma^u = \frac{4\pi^2 \nu^2 R}{g_3} \approx 4\nu^2 R. \tag{17.15}$$

The magnitude of the transverse overload n_o, acting on the plane which is perpendicular to the axis of the moveable frame oscillations depends on the deflection angle of the moveable frame ϕ, on the frequency of oscillations f, and on the amplitude of oscillations A_i.

The transverse overload n_o (taking into account the components of axial overload $n_0 \sin\phi$) may be found by using the following formula

$$n_{O_i} = \pm \left(\frac{4\pi^2 f^2 A_i}{g_3} + n_0\right)\sin\varphi \approx \pm\, (4f^2 A_i + n_0)\sin\varphi. \tag{17.16}$$

The logarithmic decrement of oscillations is /203

$$\lambda = \ln \psi = \ln \frac{A_i}{A_{i+1}}.$$

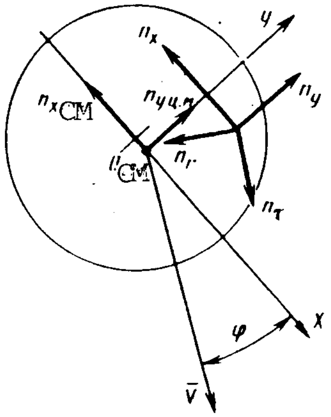
As the overloads act upon the device at its center of mass n_{XCM} and n_{YCM} at the point of interest, having the coordinates (X, Y) the total overload consists of $n_X = n_{XCM}$ and $n_Y = n_{YCM}$ components, which correspond to the movement of masses and n_r, n_τ which correspond to the movement of the device which is being tested relative to the center of masses. The n_X component is directed parallel to the OX axis, the component n_Y is parallel to OY axis and the n_r and n_τ components are caused, by the centripetal and tangential accelerations of the points in the device, as it moves with respect to the center of masses and the n_r and n_τ components are directed along the vector-radius of the point which is being investigated and normal to the vector-radius, respectively. The schematic diagram of overloads which act upon the device is shown in Figure 17.5. The arrows show the positive directions, selected in such a way that by having the positive angle ϕ n_{XCM} and n_{YCM}, n_X, n_Y will become positive.

The formulas to calculate the overloads n_X, n_Y, n_r and n_τ may be represented in the following form:

$$\left.\begin{aligned} n_X &= \frac{\omega^2 R}{g}\cos\varphi; \quad n_r = -\frac{\sqrt{X^2+Y^2}}{g}\ddot{\varphi}; \\ n_Y &= \frac{\omega^2 R}{g}\sin\varphi; \quad n_\tau = \frac{\sqrt{X^2+Y^2}}{g}\dot{\varphi}^2. \end{aligned}\right\} \qquad (17.17)$$

By having the experimental functions $\phi(t)$ and $\omega(t)$, the total overloads $N_{\Sigma X}$ and $N_{\Sigma Y}$ as projected on the OX and OY axes, may be determined by using the following formulas

$$\left.\begin{aligned} N_{\Sigma X} &= \frac{1}{g}\left(\omega^2 R\cos\varphi + Y\ddot{\varphi} + X\dot{\varphi}^2\right); \\ N_{\Sigma Y} &= \frac{1}{g}\left(\omega^2 R\sin\varphi + X\ddot{\varphi} - Y\dot{\varphi}^2\right). \end{aligned}\right\} \qquad (17.18)$$ /204



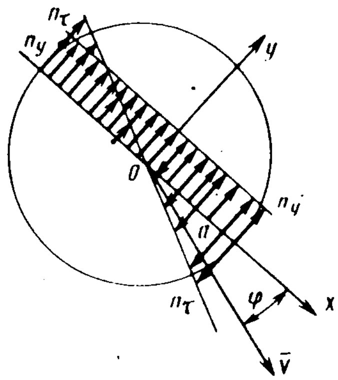


Figure 17.5. Schematic diagram of the overloads, acting upon the spacecraft or test device.

Figure 17.6. Distribution of the overloads n_y, n_τ along the spacecraft periphery.

By analyzing the effect of total normal overloads N_b along the longitudinal axis of the device, we will obtain the overload distribution, as shown in Figure 17.6. The highest total normal overloads will be at the tail end of the spacecraft or device.

The gradient of overload change during tests will have the following form

$$C = (n^u_\Sigma / t_b).$$

The limiting pulse action is $I_{lim} = n^u_\Sigma t_{lim}$.

The measure of integral action of the overload will be

$$M_O = \int_{t_1}^{t_2} \sqrt{(n^u_X)^2 + (n^u_Y)^2}\, dt.$$

After these tests, the device or module is subjected to the functional testing and a thorough inspection of all its components in the assembled and disassembled form.

Because of design simplicity, high resistance and relatively small size and weight, parachutes are used in the systems of spacecraft aerodynamic braking, during the descent to Venus and Mars and during the return to Earth.

During the stage of the spacecraft parachute system (PS) testing, the experiments are being conducted by using the aerodynamic wind tunnels, rockets, aircraft and helicopters.

A specific operational feature of the PS activation is its ejection by using the explosive charge catapulting assembly (CA) [31], which ejects from the container the encased parachute, with initial straightening of the cords, while the canopy still remains encased. The force acting upon the module or device (the load at this moment is being called the "stretching impact") will sharply increase. As a result of the aerodynamic forces, the enclosure is stripped off the canopy. At the moment of the strip-off and complete stretching of the parachute, the canopy is only partially filled. Immediately after that, the canopy fills, acquiring a stable shape. The process of filling is characterized by a considerable increase in the loads ("the filling impact"). As the next stage of operation, the parachute-module system begins to descend at a uniform rate.

Figure 17.7 shows the load change in the spacecraft module or device as a function of PS activation, during its unfolding and filling.

Because of absence of reliable and sufficiently clear similarity criteria which would make it possible to evaluate the scale effects during tests, the difficulties arise in the attempt to properly scale the whole process - the fabric of canopy, its strength, geometry of seams and other junction points. The small q_∞ in Martian conditions require large size parachutes of very small weight. The practical tests have shown that the effective gas penetrability of the canopy made of fabric decreases practically to zero at low densities. Under such circumstances, it is necessary /205
to utilize the actual PS of large size, so as to exclude the scale effects and the investigation of the basic parameters of the parachutes for example, the spacecraft parachutes for the Martian work are possible only in the upper atmospheric layers of the Earth (approximately at the height of 40-46 km).

The delivery of PS which are to be tested at such height may be accomplished either by employing a rocket or the air balloons.

Figure 17.8 shows the schematic diagram of such testing which combines the rockets and air balloon methods. The characteristic feature of these methods is the fact that the parameters which are modeled during the parachute activations are attained during the ascending trajectory. The tests of the proceding from the first stage - when the parachute is being filled and the device or module undergoes intense braking, to the second stage which is an established descent - is accomplished during the tests in the conditions which are different than the natural ones.

Since the M_∞ during parachute activation is one of the basic reproducible parameters, and taking into account that the speed of sound in the Martian atmosphere is lower than in the air at the experimental altitude, it is not difficult to establish that the PS speed must be higher during the tests by a factor of 1.48 than within the presumable Martian atmosphere. In accordance with this, on the basis of equality in the q_∞ speeds in the models and actual conditions, it follows that the Earth's atmospheric density at the height of the PS activation during testing must be lower by a factor of 2.2 than the density of the Martial atmosphere.

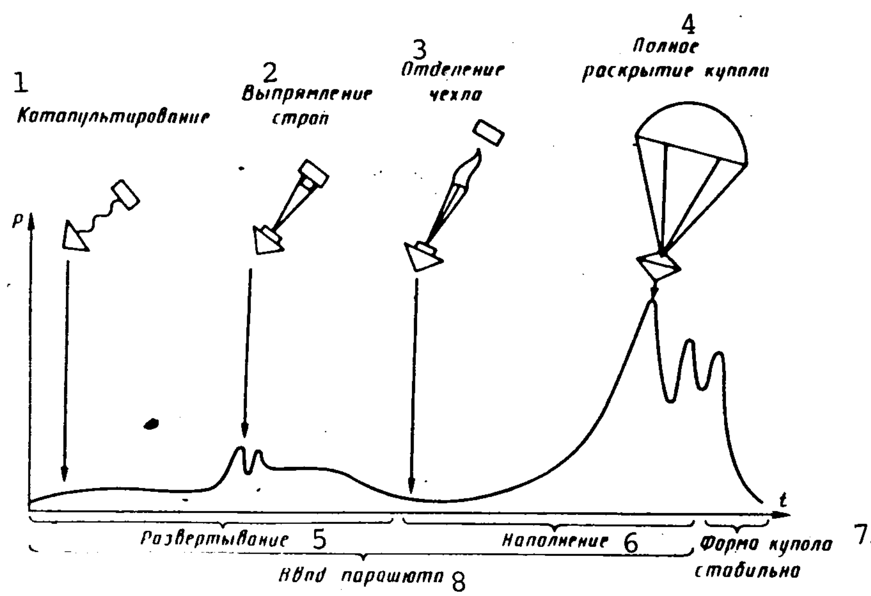


Figure 17.7. Change of the load P in the spacecraft during the activation of the parachute.
1. Catapulting; 2. Straightening of cords; 3. Stripping-off of the enclosure; 4. Canopy is fully open; 5. Unfolding; 6. Filling; 7. Stable shape of canopy; 8. Activation of the parachute.

The following conditions must be satisfied during the parachute system (PS) testing:

enclosure stripping-off and filling of the parachute must take place fairly uniformly, without tilting of the canopy; /206

the canopy must be sufficiently resistant-stable, after the initial phase of filling;

the canopy must develop sufficient resistance which would satisfy the target oriented tests of the spacecraft and its module in terms of the velocity at the end of the parachute descent;

the parachute must have sufficient excess design criteria, so as to withstand the maximum overloads, as it unfolds in the planetary atmosphere, ensuring safe conditions during braking;

the amplitude and angular velocity of the module oscillations should not exceed the permissible values ($\pm 25^{\circ}$, 30°/s) during the stationary descent, as recalculated, to the planetary atmosphere;

the q_∞ is selected in such a way as to compensate (by lowering the loads on the parachute) for the increased heating because of descent and high M_∞ and at the same time, one must take into account (by increasing the loads somewhat) that the PS would not be subjected to the early effect of the other factors which are involved in the interplanetary flight.

One of the important parameters in studying the spacecraft parachute systems is its filling time, which in the Martian conditions may be higher by 40-50%, since the velocity during the parachute activation in the Martian atmosphere is lower than during the tests in the Earth's atmosphere for the same M_∞ and q_∞.

As one can see, the conditions of modeling of the transition and established operational modes are incompatible since the transition motion is associated with the mass and the equilibrated motion depends on the weight. Therefore, in the course of the experimental studies, both types of motion must be investigated separately, simulating the most important factors.

Parachute testing of the spacecraft and modules for work in Martian conditions which are being conducted in the terrestrial atmosphere at low altitudes (3-4 km) are to verify the strength factors and functional parameters. Here one must develop the loads to which the PS will be exposed, associated with the design stresses which should be higher than the calculated values by approximately a factor of 1.5. /207

In spite of high velocity at the initial moment of the PS activation at low altitude, the maximum loads will be lower than if one uses rocket activations because of sharper decrease in the velocity as the parachute is unfolding in the environment of higher atmospheric density.

It must be pointed out that the PS testing at low altitudes is quite effective for the development of computational methods, enabling us to account for the high altitude testing and estimate the loads and functional efficiency of the Martian PS.

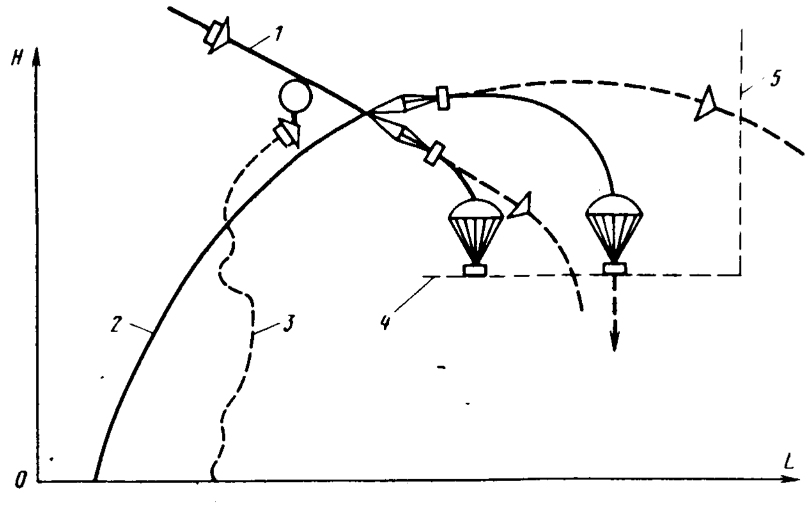


Figure 17.8. Comprehensive ballistic test diagram of the parachute system in the Earth's atmosphere:
1 - trajectory which models the Martian atmospheric entry; 2 - trajectory during the rocket launching; 3 - trajectory during the air balloon launching; 4 - conditional level of the Martian surface; 5 - conditional height above the Martian surface.

During the testing of a parachute system in the case of the spacecraft descending module Pioneer-Venus (USA), its mock-up, equipped with instruments and heat shields, was dropped off from the air balloon at the height of ~32 km in order to refine the systems of the parachute unfolding and stripping of the heat shield and also of the deactivation of the parachute after the 17-minute descent.

The parachute systems during such tests must be activated at the height of ~16 km, where the temperature and atmospheric density and also the velocity of the module correspond to the conditions of the Venus atmospheric entry, at the height of 67.6 km. The mock-up tests and the operation of the parachute system were monitored by using the movie cameras, placed on the air balloon and within the spacecraft mock-up. The observations of the mock-up module detachment were also conducted by ground means.

CHAPTER 18.

PHYSICAL MODELING OF THE SOFT LANDING OF THE SPACECRAFT AND ITS OPERATION ON THE PLANETARY SURFACE

The experimental testing of the landing and operation of the spacecraft is being conducted in accordance with the basic computed operational conditions which model the character of environmental actions and which include: /207

experimental work in selecting the design construction of the landing device, the selection of materials and rigidity of the shock-absorbing components;

the tests to evaluate that the parameters of the landing device correspond to the initially desired ones;

the tests to determine the strength of the module and of its components, applying its maximal mass and assuming the maximum relative angle between the landing device and the soil surface;

the tests to verify the device or module stability at the moment of contact with the surface, applying the uppermost and lowest positions of its center of mass, the maximum and minimum moments of inertia, the change of orientation angles in the device or module and the inclination angle of the surface;

the tests to determine the maximum impact load factors for the maximum and minimum mass of the module and zero relative angle between the landing gear and surface;

the tests to verify comprehensively the reliability of the onboard systems, in the presence of simulated planetary conditions. /208

The whole test process begins with a set of simple prototypes and mock-ups. Then the mock-up, with the standard operational devices, undergoes testing. The further testing (of the third-fifth type) is being conducted on the experimental device or module which in design is analogous to the standard operational one, with the mock-ups of all attached instrumentation, similar in size and weight and these are the finalized tests to substantiate the strength and stability of the module. The comprehensive testing is being conducted on the module with the functioning equipment in order to verify the operational state of the equipment, as one simulates the actual operational conditions of the module on the planet.

It is desirable to simulate as many as possible external factors at the earlier stages of testing. This would allow to find in time the design defects. An important question is the scope of tests and the minimal number or experiments of each type, after which, if one obtains positive results, one may assume and guarantee a sufficient reliability. It is necessary to take into consideration the inherent properties of design, the degree of novelty and complexity in the propulsion system design, the presence of reliable methods which are used for the theoretical and experimental work, the use of new materials and new technology, the ability for the multiple use of the components, the ability to regenerate the component after testing and also, one must take into account the importance of each assembly, system or aggregate for the viable operation of the module as a whole.

The test program must provide the operational modes which not only take into account the nominal, but also the limiting levels of the external and internal factor variation and the most unfavorable combination of such factors. In accordance with this, one determines the technical assignment and the test methods for each concrete case.

The module stability depends primarily on the landing speed, its construction design, its mass-inertia parameters, the planetary soil, coefficient of friction of the landing gear as it slides on the ground and the atmospheric parameters. The method of simulated landing by using the catapult technology was utilized in modeling the landing of Mars-6 module on the Martian surface.

The module landing was conducted on the planetary surface, with the surface inclination angles $\alpha=0 \ldots 10^{\circ}$, the carrying capacity $\sigma_c = 0.2 \cdot 10^5 \ldots 1.5 \cdot 10^7$ Pa, the horizontal velocity of $V_h=14.2 \ldots 28.5$ m/s, and vertical velocity $V_v=8.4 \ldots 13.1$ m/s. The test temperature before touchdown was $T=\pm 50^{\circ}C$.

The calculations during the module landing are shown in Table 18.1.

The module impacts against the planetary surface are absorbed by the variable thickness enclosure, enveloping the whole spacecraft shell.

The required relationships between the horizontal and vertical velocities, while the module makes contact with the ground, is provided by the change in the inclination angle of the directional catapult α. /209

In analyzing the module strength, the device sinking into the ground and operation of the equipment, one employs the simulated soils with the carrying capacity σ_c $(0.2 \ldots 1.0)\cdot 10^5 - (1.0 \ldots 1.5) \times 10^7$ Pa (aglophorite sand, foam concrete and concrete).

The required direction of the load factor is provided by turning the longitudinal axis X of the module with respect to vertical by

TABLE 18.1

Parameter	Computed cases during touchdown I	II	III	IV	V
Spacecraft approach angle to the planetary surface α^o,	21	22,5	28,5	37,5	42,5
Inclination angle of the guiding devices α_1^o,	18,5	18,5	18,5	37,5	37,5
Inclination angle of planetary surface α_2^o	2,5	4,0	10	0	5
Pitch angle θ^o,	180	120	20 - 60	60	20 60
Bank angle ψ^o,	0	90	0 170	0	0
Landing speed:	30	26,5	30	21,5	18
horizontal V_h, m/s	28,5	25,2	28,5	17	14,[illegible]
Vertical V_v, m/s	9,5	8,4	89,5	13,1	11

(Note: commas in tabulated material are equivalent to decimal points.)

θ angle and by turning it with respect to X axis by ψ angle. The necessary landing speed V_k of the module is simulated by using the acceleration system and shock absorbing devices. The shock absorbing capacity calculations involve the determination of cable diameter, number of coils in the cable and the operational range of relative cable stretching. By preliminary assignment of the cable diameter d, one determines the maximum P_{max} and minimum P_{min} load factors, taking into account the relative stretching ε.

The number of coil turns in the shock absorbing acceleration system is determined from the appropriate relationship ($i=P_{max}/2P_{c\,max}$) where $P_{c\,max}$ is the maximum load to which one cable is exposed.

The total movement of the shock absorbing assembly L and the work A are determined by using the following formulas

$$L = \frac{A}{\eta(P_{max} - P_{min})}; \quad A = \frac{l}{2} i (a_{max} - a_{min}), \tag{18.1}$$

where ℓ is the length of unloaded coil turn, a_{max} and a_{min} is the work of one running meter of the cable, in the presence of maximum and minimum relative stretching, ε_{max} and ε_{min}, respectively.

By varying ℓ, i, ε and d, one can select the rule of the acceleration force change P(t).

The tests are conducted in a certain sequence. The module which is prepared for testing is fastened by the belts to a cart which is in the extreme lower part of the directional guides. The cart together with the module is being lifted by electric crane into its extreme upper position and fastened by the electromechanical lock. The shock absorbing system is thermally stabilized.

The shock absorbing system is loaded by using the electric crane and the magnitude of this load is adjusted by using the tenso-dynamometer. As one attains the maximum force P_{max}, the electric lock unlocks and the amortization devices accelerate the cart with the module to a specific velocity V_k. At the end zone of the directional guides, the cart is stopped by a braking system and the module, because of inertia, continues its movement, impacting the analogue of the soil. During such tests, one determines the load factor, the speed at the end of accleration, the impact force and the speed photography is also employed. /210

In simulating landing by using the catapult approach, the tested device must have a specific speed V, which corresponds to the landing speed V_h, V_v. The equation of the device or module motion may be written in the following form

$$\frac{dV}{dt} = \frac{P(t)}{\Sigma M}, \tag{18.2}$$

where P(t) is the acceleration force, ΣM is the mass of moving objects which is determined by using the following formula

$$\Sigma M = M_{mod} + M_i + (1/3) M_{am},$$

M_{mod} - is the mass of the module, M_i is the mass of the cart and drive gear, M_{am} is the amortization gear mass.

At the beginning of acceleration (t=0, L=0 and V=0); at the end of acceleration ($t=\tau$) the condition $V=V_k$ must be satisfied.

By integrating the equation (18.2) we will obtain

$$\left.\begin{aligned} V &= \frac{1}{\Sigma M}\int_0^t P(t)dt; \\ L &= \frac{1}{\Sigma M}\int_0^t \left[\int_0^t P(t)dt\right]dt \end{aligned}\right\} \tag{18.3}$$

and, correspondingly

$$\left.\begin{array}{l} V_k = \frac{1}{\Sigma M} \int\limits_0^T P(t) dt; \\ L_T = \frac{1}{\Sigma M} \int\limits_0^T [\int\limits_0^t P(t) dt] dt. \end{array}\right\} \quad (18.4)$$

The equation of motion of the module mock-up within the acceleration segment will have the following form

$$\ddot{x} - \frac{P(t)}{M_\Sigma} \quad g \sin \alpha + f(\dot{x}) g \cos \alpha_1 + \frac{c_x}{2M_\Sigma} \rho x^2 S = 0.$$

During the preliminary experimental testing, one determines the velocity V_k as a function of the mass of the moving objects $V_k = f(\Sigma M)$. It is necessary to keep in mind that during the acceleration, when the module speed increases up to V_k, the module is affected by the reverse load factor which is not desirable. Depending on the acceleration segment L and the developed speed V_k, the inverse load factor may be considerable. The magnitude of maximum overload during acceleration may be estimated in its first approximation by using the following formula

$$n_P^* \approx \frac{P(t)_{max}}{\Sigma M g}. \quad (18.5)$$ /211

It is desirable that during the acceleration, the reverse overloads should be minimal and should not exceed 10% of the work load factors which occur as the module impacts the soil.

The equation of module motion during braking may be written in the following form

$$M_a \frac{dV}{dt} = -F(t), \quad (18.6)$$

where F(t) is the braking force.

The load factor during the impact is

$$n(t) = \frac{1}{g} \left|\frac{dV}{dt}\right| = \frac{F(t)}{M_a g}. \quad (18.7)$$

The levels of overload make it possible to determine the active pulse acceleration and forces

$$\left.\begin{aligned} a(t) &= -gn(t); \\ F(t) &= \Sigma Mgn(t). \end{aligned}\right\}$$

The equation of motion (18.6) will be written in the following manner:

$$\frac{dV}{dt} = -gn(t).$$

The initial conditions for integration of the equation are:

$$t=0, \quad x=0, \quad V=\frac{dx}{dt}=V_k.$$

After integration, we will obtain

$$\left.\begin{aligned} V &= V_k - g\int_0^t n(t)dt; \\ x &= V_k t - g\int_0^t \left[\int n(t)dt\right]dt. \end{aligned}\right\} \tag{18.8}$$

At the moment in time $t=\tau$, when the impact load factor reaches its maximum ($n=n_{max}$), the speed of the module will decrease to zero.

The maximum overload n_{max} as a function of the impact velocity may be represented in the following form

$$n_{max} = \frac{V_k}{k_V g t_i}, \tag{18.9}$$ /212

where k_V is the coefficient which depends on the shape of the impact pulse, t_i is the duration of the frontal overload increase.

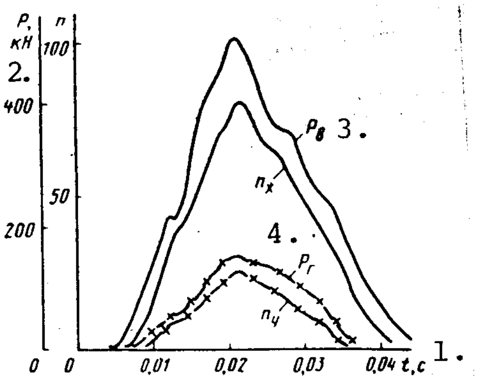


Figure 18.1. Vertical and horizontal loads P_v, P_h and of the longitudinal and transverse overloads n_x,n_y as a function of time t: V_k=21.5 m/s, $\theta=60^{\circ}$, $\alpha=37.5^{\circ}$ and $\psi=0$.

Key: 1. seconds; 2. kN; 3. P_v; 4. P_h

The k_V coefficient is determined by using the dimensionless curve for the impact pulse and utilizing the following equation

$$k_V = \int_0^1 \frac{n(t)}{n_{max}} d\frac{t}{t_i}.$$

The maximum braking path X_{max} will have the following form

$$x_{max} = k_x g n_{max} t_i^2 = \frac{k_x}{k_V} V_k t_i, \qquad (18.10)$$

where the k_x coefficient also depends on the shape of the imact pulse.

The impact of mock-ups with the soil were conducted for all calculated cases:

"bottom" - the overload n=130...170;

"top" - overload n=50...180;

"side" - overload n=70...110;

"bottom" into the sand - overload n=70.

The maximum collision force P_{max}=670 kN, the maximum overload n_{max}=180 and the coefficient of friction f_f=0.17... 0.28.

The strength of the module design, as a whole, was investigated by using three mock-ups with simulated equipment on board of actual mass. Preliminarily, each such prototype was subjected to mechanical testing. Five calculated cases were investigated (see Table 18.1). The collision between the mock-up and soil was at the point of attachment to the aerodynamic cone and at the point where the parachute compartment is attached. The landing speed was V_k=18...30 m/s (horizontal V_h=14.2...28.5 m/s, vertical V_v=8.4...13.1 m/s). The load factor along the X axis was $n_x \leq 80$, in the plane of impact $n_{side} \leq 180$. Figure 18.1 shows the P=f(t) and n=f_1(t) relationships. After the mock-up was jettisoned, one checked out the proper functioning of the protective enclosure detachment, etc.

The testing on the ground of the landing dynamics was terminated by carrying out a set of comprehensive tests, utilizing the catapult, the module with the functioning equipment which underwent separate check-out in terms of the climatic and mechanical factors. Earlier, the module as a whole was subjected to mechanical testing (vibration and centrifugal load factors). The complex and comprehensive tests were conducted for case IV (Table 18.1). The impact force was 630 kN, which corresponds to the load factor n=170. Before jettisoning the mock-up (using a catapult),a programmed time mechanism was remotely activated and all subsequent check-out of the equipment functioning was conducted using a standardized cyclogram. The catapult (Figures 18.2 and 18.3) makes it possible to simulate the module landing with large horizontal speeds (up to 28.5 m/s), with the peak impact velocities (up to 32 m/s), making it possible to determine the effect of such speed on the strength of the module and to test and refine its energy-absorbing components.

The catapult assembly consists of a moveable support, which mounts the directional guides at two fixed angles. A small cart moves along the inclined guides which have the interchangeable /213
supports to set up the tested module at different positions. The cart, while being in the upper starting position is locked by a electromechanical lock.

At the lower part of the guiding system, we have the braking system to absorb the cart energy. The braking stretch of the cart $L_T \leq 1.5$ m (the initial length of braking shock absorbing devices $L_0=1.5$ m, $\varepsilon=0.8 \ldots 1.0$).

The module landing was accomplished on the simulated soil, which was placed on the tensometric plate or on the aglophorite sand within the enclosure.

Technical Parameters of the Catapult Assembly

Mass of the tested objects, kg	500
Mass of the cart, kg	100
Maximum acceleration force, kN	85
Work of acceleration system, kJ	280
Peak impact velocity, m/s	32
Inclination angle of guides, rad	0.3-0.6
Inclination angle of the soil, rad	0-0.2

The simulation methods of landing by using the dynamic stand was developed to investigate the landing of Venera-9 - Venera-14 type of spacecraft, on the surface of Venus. The landing of the device was conducted with the maximum speed of $V_v=8.2$ m/s (probability of staying within the range P=0.997) on the planetary surface at the

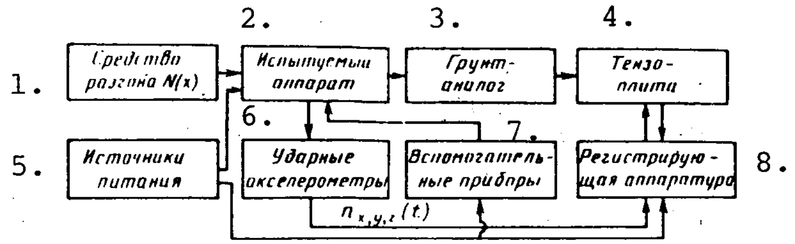


Figure 18.2. Block diagram of the catapult assembly.

Key: 1. Means used for acceleration N (x); 2. Tested module; 3. Simulated soil; 4. Tensometric plate; 5. Power sources; 6. Impact accelerometers; 7. Auxiliary instruments; 8. Recording equipment

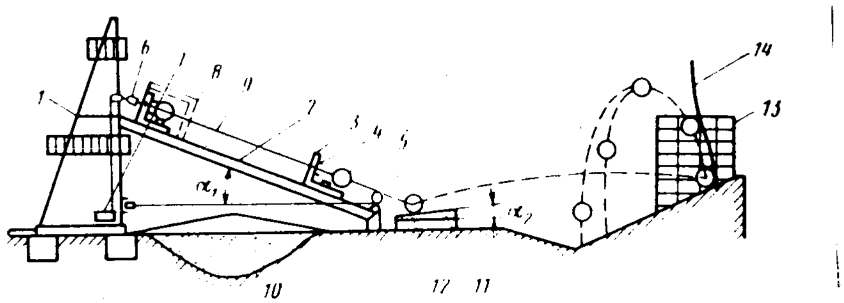


Figure 18.3. Fundamental schematic diagram of the catapult assembly:
1 - supports; 2 - directional guides; 3 - cart; 4 - module holder; 5 - module; 6 - lock; 7 - crane; 8 - thermostat; 9 - acceleration system; 10 - aglophorite sand; 11 - tensometric plate; 12 - simulated hard soil; 13 - protective wall; 14 - screen

angles $\alpha=0...10^{\circ}$ and the soil carrying capacity $\sigma_{lim}=0.2\cdot10^{5}...3\cdot10^{6}$ Pa, the ground being of different composition. The ambient temperature was 710-750 K. /214

Different calculated cases for the module landing:

Case 1 - landing on the horizontal surface: vertical speed $V_v=8.2$ m/s, horizontal speed $V_h=1...1.5$ m/s, longitudinal axis of the device prior to impact is vertical, $\theta=0$. In this case, we have the maximum overload at the center of mass of the device and at the peripheral points.

C-4

Case 2 - landing on the inclined surface: vertical speed V_v=8.2 m/s, longitudinal axis of the device before impact is close to vertical, $\theta=0...5^o$. In this case one studies the stability of the device, so that it would not flip over.

Case 3 - landing on the inclined surface: vertical velocity V_v=8.2 m/s, longitudinal axis of the device before impact is deflected by $\theta=-15^o$. In this case, one obtains the maximum compression of the shock-absorbing means.

Assuming that the energy absorbed by the ground is equal to zero, we can determine the maximum force, acting upon the module

$$F_{max}=\frac{MV_k^2}{2kH_{max}}+\frac{Mg_{pl}}{\eta_{wf}}, \tag{18.11}$$

where η_{wf} is the total work factor of the shock-absorbing component in the landing device

$$\eta_{wf}=f(S_a,R,T,H,H),$$

where S_a is the surface area of the shock-absorbing component, as it comes in contact with the ground, R is the specific destruction load.

The F_{max} is constrained by the maximum overload for the module during landing which is determined by taking into account the expression (18.11)

$$\eta_{max}=\frac{F_{max}}{Mg}=\frac{V_k^2}{2kgH_{max}}+\frac{g_{pl}}{\eta_{wf}g}. \tag{18.12}$$

To determine the friction force F_f near the zero velocity of skidding, one introduces the k_V factor which is the friction as a function of velocity, and then

$$F_f=f_f k_V P_g,$$

where P_g is the reaction of the ground and f_f is the coefficient of friction.

The force acting upon the landing gear of the module, taking into account the response of the ground, may be determined on the basis of the following equation

$$\bar{F}_a = S_K(a_1 + a_2 \cdot \dot{S} + a_3 \dot{S}^2)(\bar{S}/\dot{S}), \tag{18.13}$$

where a_1, a_2, a_3 are the parameters of the ground, taking into account the effect of movement, of the velocity and of the square of the velocity of the module on the ground resistance, S_k is the surface area of the shock-absorbing component in the landing module which is in the tangential motion with respect to the surface of the ground, S, S are the displacement and speed of the shock-absorbing component in the module.

The overloads, or load factors during the module landing may be determined by using the following formula

$$n = \frac{0.5MV_k^2 - W}{\eta_g h_g gM}, \tag{18.14}$$

where W is the energy, absorbed by the shock-absorbing component, η_g is the factor of total ground deformation, h_g is the depth of penetration into the ground.

The duration of impact pulse is determined in the following manner

$$t = \frac{2(h_g + H)}{V_k}.$$

The nature of the overload change during the impact is determined on the basis of the module equation of motion, as it makes contact with the ground

$$\ddot{x} - g + \frac{S_a}{M} xk + \frac{P_a}{M} = 0, \tag{18.15}$$

where x is the depth of sinking of the shock-absorbing component, k is the subsoil factor and P_a is the load on the component.

To account for the effect of all parameters of the module and of the ground, as they affect the landing and to evaluate the actual stability safety margin, is possible only experimentally.

The experimental testing is conducted by using the actual dynamic mock-ups in which the mass, coordinates of the center of masses and moments of inertia may be manipulated.

The mock-up is essentially a simplified rigid construction design with standard landing gear attached. The landing assembly is a thin-walled shell of toroidal shape, attached to the rigid body by means of a welded support. The mock-up body consists of the welded together, cylindrically shaped steel sheath and flanges. To obtain the required mass and to generate the inertia parameters, the mock-up body includes the ballasts which can be moved. At the wall of the body, near the center of mass, a beam is incorporated which provides hinged attachment of the moving system.

The tests are conducted by using a dynamic stand, ensuring that the mock-up has six degrees of freedom (Figure 18.4). The technical parameters of the stand are presented below.

Lifting capacity, kg	3200
Elevation height of the mock-up, m	30
Instrumental, computerized factor $n\tau$,s	7.7
Surface area size, m^2	4X5
Working pressurized area, m:	
diameter	2.7
height	2.5
Working area temperature, K	770
Nominal power output of the heater, kW	90
Power supply voltage of the heater, V	380
Nominal current, A	140
Height from the horizontal surface area to center of pressurized area, m	3.5
Height range for the adjustment of pressurized compartment, m	±0.5
Specific load of the heater, W/m^2	$1.47 \cdot 10^{-4}$

The stands consist of a drop system, trestle, the thermostat with heater, the shock-absorbing and tension devices, landing platform and the measuring and recording instrumentation.

The system for drop-off includes the rotationl boom pendulum suspension of the mock-up, the device to simulate planetary gravitational force which ensures a certain range for unloading and consequently decreases the effect of free fall, as the mock-up is being dropped. /216

The trestle is of steel construction and the horizongal displacement of the moveable part of the thermostat is accomplished by using rails. The trestle ensures the movement of the thermostat also in the vertical direction.

The thermostat consists of two semicylinders: a stationary one without bottom and a moveable one with disk bottom, mounted on a cart.

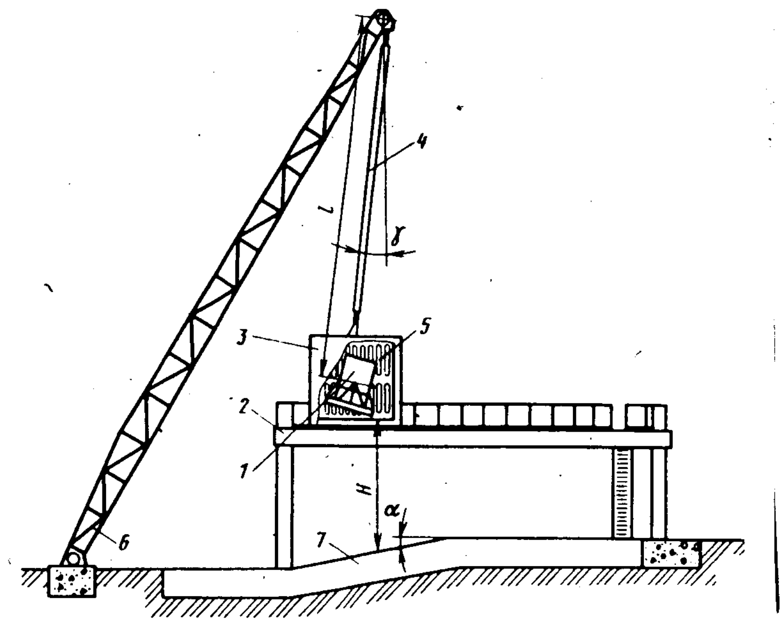


Figure 18.4. Basic schematic diagram of dynamic stand:
1 - mock-up which is being tested; 2 - trestle; 3 - thermostat;
4 - unloading system; 5 - heater; 6 - rotating boom;
7 - simulated ground

The thermostat enclosure has twin walls. The space between the walls is filled with kaoline wool. The heater, located inside of the constant temperature chamber consists of 54 heating elements, made of nichrome wire. The shock-absorbing assembly (rubber straps) are used to pull back the moving part of the thermostatic enclosure, at the moment when the mock-up is being dropped-off. The landing platform simulates the surface layer of the planet by using the analogues of the soils of different carrying capacities.

The stand makes it possible to simulate the free fall acceleration on the planet, it is possible to generate the required landing velocities, to change the initial position of the mock-up and the ground inclination angles, providing the required angles with respect to the planetary surface which is to be approached by the mock-up.

The thermostatic assembly recreates the thermal environment of the planetary atmosphere, as it affects the module at the terminal descent trajectory segment, according to the assigned rules T(t). /217

The contemporary concepts about the Venus surface present a broad range of types of soils which are to be analyzed during the studies on the ground, utilizing the new types of Venera spacecraft type: sandy surface areas (analogue is the aglophorite sand), fine crushed rock with sandy base, large rocky areas and solid ground.

On the whole, the surface of Venus is relatively smooth, the inclinations are basically within the range of 0-10°.

To evaluate the landing dynamics and operational viability of the module on the surface of Venus, the following models of the simulated soils have been developed which reflect the major parameters of Venus surface.

Model A. The area with aglophorite sand and the inclination angle α=10°. The carrying capacity of such surface is $(0.2...1.0)\cdot 10^5$ Pa. To exclude the edge effects, the aglophorite sand thickness is not less than 1 m. The base of the area is made of concrete or of foam concrete of no less than 0.5 m thickness.

Model B. Model A is used as a basis. On the surface of the aglophorite sand, one scatters fine crushed rock of 30-80 mm diameter, of particles (50% of 30-50 mm size and the other 50% of 50-80 mm size), in other words δ=30...80 mm, with the carrying capacity of no less than $1\cdot 10^7$ Pa (crushed, annealed bricks).

Model C. Model B is used as a basis. On such surface, one places the rocks made of foam concrete and of granite, the height of which are 150-200 mm, and which are 250-300 mm cross-sectionally. The distance between such rocks is about 100 mm. The carrying capacity of the granite is $(1...3)\cdot 10^8$ Pa.

Model D. The soil surface area has two variants: α_1=0 and α_2=10°. It consists of the foam concrete blocks with carrying capacity $(2...3)\cdot 10^6$ Pa, and the height of no less than 500 mm. The blocks are stacked on a concrete base of no less than 50 mm in thickness. The blocks are cemented together, using the solution of the same strength capacity.

The simulted soil during such experimental studies of the mock-up landing is thermally regulated. The soil temperature is determined by using the thermal sensors, embedded into the landing soil at a depth of 50-100 mm. Depending on the type of tests, the mock-up drop-offs are conducted over the appropriate model of the simulated ground. Before each type of testing, the mock-up is being weighed, the coordinates of the centers of mass and the moments of inertia, with respect to the three orthogonal axes are determined. The mock-up is joined together with the suspension system which simulates the planetary gravitational force, and such system incorporates the electromechanical lock. The mock-up is placed on a platform and the rubber amortization system, by using cranes and straps, is activated, until the necessary unloading force is attained. Then the mock-up is lifted into the working area of the temperature-regulated enclosure, placed at a given height H.

The spatial orientation of the mock-up is attained by adjusting its position with respect to the suspension: the mock-up X axis coincides with the direction of the vector of gravity, by rotating it around the X axis, one establishes the yaw angle ϕ and by rotating it around the Y axis - the pitch angle ϕ. After that, one adjusts the assigned bank angle ψ. Now one couples the moveable part of the thermally regulated enclosure and the stationary one, and then heats the mock-up in the air (or hydrogen) environment, within the enclosure.

After attaining the assigned temperature mode, the remote /218
controlled separation of the thermostatically controlled enclosure with the rolling back of its moveable parts, one activates the soft landing system.

First one activates the lock of the pendulum system. As the pendulum moves from its vertical position, the lock is activated and the mock-up is dropped off. The released mock-up, which has a specific horizontal speed V_h, begins its drop. As the mock-up drops down, the unloading force increases, reaching a necessary level at the moment of its contact with the surface.

During the mock-up soft landing, the fluctuations in the unloading force begin to appear, which is due to the inertia in the amortization device. The longitudinal oscillations of the amortization device results in some change of the force associated with the weight loss (by 5-8%). The latitudinal oscillations change the direction of the force, which results in the landing distortion, because of the change in the speed of motion. To evaluate the nature of the mock-up movement with the amortization device, one can utilize the dimensionless additive velocity $\bar{V} = \frac{V_a}{V_h}$ which is characterized by the change in the mock-up kinetic energy, being acted upon by the amortization device [6]

$$\bar{V} = \sqrt{\frac{2P_0}{Ml\omega^2}\sum_{n=1}^{\infty}\frac{\dfrac{\sin \pi/2n\,\frac{\omega_0}{\omega}}{\left(n\frac{\omega_0}{\omega}\right)^2}-1}{1-\left(\frac{n\omega_0}{\omega}\right)^2}}, \qquad (18.16)$$

where P_0 is the force associated with the weight loss and ω is the angular frequency of the acceleration pulse.

The following parameters are being measured during the on-the-ground experimental studies of the landing dynamics: the pitch and yaw angles, the bank angle and the load factors in three orthogonal directions, the angular velocities and acceleration of the mock-up rotation, the deformation in the components of the landing device and the stresses in the construction design components, the trajectories of motion.

The process of contact between the mock-up and the surface of the simulated ground on the landing platform is fixed in two mutually perpendicular planes against the coordinate grid background, by using the movie camera equipment which operates at 380-460 frames per second, making it possible to determine the trajectory of motion of the center of masses of the device x(t), y(t), z(t), the inclination angles of the device axes $\theta(t)$, $\phi(t)$, $\psi(t)$, the angular velocities $\omega_x(t)$, $\omega_y(t)$, $\omega_z(t)$ and the angular accelerations around the axes $\dot{\omega}_x(t)$, $\dot{\omega}_y(t)$ and $\dot{\omega}_z(t)$. To obtain the qualitative picture of the device behavior during landing, one utilizes a TV camera and radio receiver. After landing, one photographs the mock-up and the deformation in the simulated soil region. One measures the residual deformations of the shock-absorbing shell, the destruction of the simulated layer of the ground, and also the sinking of the landing gear into the soil, etc.

The operational modeling of the planetary transportation means (PTM) - of the planetary self-propelled devices and of the planetary flight devices is being carried out in order to elucidate their operational parameters, to estimate the reliability and functional viability of specific subassemblies, systems and of the devices as a whole, as one simulates the actual operational conditions. For the automated PTM, in addition, of great importance is acquiring by the crew on the ground of the control techniques, taking into /219
account the specifics of the transmitting and receiving devices within the command communication line. During the optimization of PTM on the ground, it is necessary to reproduce thoroughly all major factors which are acting upon these devices, during the stage of its delivery to the planet and during the functioning of such devices on the planetary surface [12, 15]. It is reasonable to start such tests at the earliest stages of the design work, during the periods of design selection, the development of instrumentation and the design of the general appearance of the PTM.

We have presented earlier the scale of some of the physical quantities which are significant during modeling of the dynamics of the PTM movements. Of practical interest are the following cases:

a) $K_E=1$, $K_\rho=1$, $K_\ell=1/K_j$ ($K_\ell=1.6$ in the case of the Moon and $K_\ell=2/5$ in the case of Mars and Mercury. In such case, one must manufacture the dynamically similar model of the planetary vehicle, by using the materials of the actual design work (titanium and aluminum alloys, steel), by changing proportionally the size of all model components and the relief of the planetary surface. In such case, the linear speed of movement of the PTM model will be $K_V=1$, in other words, the model will have the same speeds which the actual planetary vehicle (planetary airborne device) will have. This simplifies significantly the experimental studies and the recalculation, aiming at the actual operational conditions. Let us assume that the movement of such PTM model is being recorded on a film, with the actual distances being $1/K_j$ and which is faster than the normal ones by a factor of K_j. Then, in reviewing the film with normal speeds, one can determine the actual process of the planetary vehicle movements across the actual relief of the planetary surface, and the decrease in linear size of the device and the region of its operation by a factor of K_j as compared to the actual conditions makes it possible to test the PTM models on specially designed test ranges, which simulate the planetary relief, the properties of the soil, the actual illumination and thermal conditions.

b) $K_E=1$, $K_\rho=1$, $K_\rho=1/K_j$ ($K_\rho=1/6$ in the case of the Moon and $K_\rho=$ $=2/5$ in the case of Mars and Mercury. Here we encounter a contradictory requirement, namely, employing the actual size of the dynamically similar planetary vehicle model, it is necessary to obtain the total mass of the model which is smaller by K_j factor than the actual one. In order to satisfy this requirement, one must construct the model of the PTM drive which is of actual size, employing the actual construction materials, while the remaining components of the model (useful payload, instrumental compartments, a part of the construction design, etc.) must be decreased considerably, but only in such a way that the total mass of the obtained planetary vehicle model would correspond to the $K_m=1/K_j$ scale, and the model would have the actual centering of the mass, which is inherent in the actual planetary vehicle.

The specifics of the actual dynamically similar PTM model are: angular acceleration increase by a factor of K_j, the increased linear and angular velocities and power output by a factor of $\sqrt{K_j}$, the actual energies (work mode), associated with the stresses, rigidities and forces, acting upon the model, the test time which is shorter compared to the actual time by a factor of $\sqrt{K_j}$ and the mass, mass density and moments of inertia of the model, decrease by a factor of K_j.

The dynamically similar model, in contrast to the actual PTM mock-up (see below) does not require any specific mechanism for "unloading" part of its weight by $[1-(K_j)^{-1}]$ which significantly simplifies the design of the test range for the comprehensive verification of the actual self-propelled chassis in the course of ground testing: in determining the actual safety margins of the drive and propulsion units, the interaction with the ground, the /220 obstacles which are to be overcome, the thrust parameters and energy expended, the parameters of smooth movements and stability of movement. The tests of such model are also interesting from the point of view of crew training, trying to elucidate the dangerous situations, to develop the techniques and methods for the remote control of the planetary vehicles.

C) $K_E=1$, $K_\ell=1$, $K_\rho=1$. In this case, the modeling is accomplished by using the PTM mock-ups manufactured by using the actual construction design materials and employing the actual masses and size. The retention of similarity in employing such modeling is possible only by artificially creating on the Earth the planetary gravity acceleration ($K_j=1$). This may be achieved by several methods. The only strict method of obtaining the planetary gravity acceleration is to place the PTM mock-up into the airborne laboratory. Less rigid, but simpler from the technical point of view are various methods of "unloading" in which the correlation is being obeyed in terms of the actual devices only in the mock-up weight which is being acted upon by the gravitational forces of the Earth (the mock-up must be "unloaded" by $[1-K_j^{-1}]$ of its total weight. In this case, it would be reasonable that the selected method of unloading would be applied to the largest number of PTM mock-up components. The mock-up may be unloaded by using counterweights (for the basic components of the mock-up), by using the aerodynamic braking, by employing the hydropneumatic mechanism, by using the attached air balloon under the mock-up, thus creating an air cushion under the mock-up, or by using the inclined wall, tilted at α angle (in optimizing the lunar vehicle, $\alpha=80.6^\circ$, in optimizing the Martian vehicle $\alpha=66.6^\circ$), etc.

Since the PTM mock-up is continuously in contact with the unloading mechanism by using a system of cables, the zone of its maneuvering is limited by the size of the test equipment (of the test range). In testing the mock-up of the planetary vehicle on an inclined wall, difficulties are encountered to simulate the relief of the region being investigated, and the properties of the porous, deforming soil of the planet. However, in order to acquire proper habits in the PTM remote control, particularly in overcoming different obstacles on the way, and to evaluate the static and dynamic stability of the device and its reliability, it is quite reasonable to utilize a circular, inclined wall. By providing at the end, during tests on the ground employing PTM mock-ups, the scale of $K_j=1$, one can manage that the remaining derivatives of the scale under consideration would also be equal to unity.

It has been noted earlier (Chapter 16) that the spacecraft functioning on the planetary surface is affected by the physical and mechanical properties of its ground. The conditions of similarity [15] require the development and application during testing of the natural models (case B), and natural mock-ups (case C) of the special model soils which differ in terms of their properties from the actual planetary surface layer (for example, the density of such soils must be lower than the planetery surface by a factor of K_j). /221

But because to design the perfectly similar model soils, or to "unload" each particle of the natural soil, is rather complicated the error estimation was carried out, associated with the use during the model testings, as per B and C cases, the natural simulated soils which correspond in terms of their properties to the present-day concepts about the planetary surface layers. It has been established that the magnitude of error increases with the increase of the surface layer depth. On the surface of the ground, the error in determining its settling properties and the normal pressure are practically absent. At the depth of 0.5 m, as the module presses on the ground with the force of $0.5 \cdot 10^5$ Pa, and the soil density of 1200 kg/m^3, this error reaches 9.5% as one optimizes the lunar vehicles and 6.5%, as one optimizes the Martian vehicles. Consequently, the use during the ground testing of simulated soils without unloading, which are similar to the planetary soils, is quite permissible. In accordance with the similarity criteria, the same simulated soils must also be used in testing the similar PTM models which are dynamically similar but decreased in size (case a).

CHAPTER 19.

COMPARSION OF THE FINE-TUNING AND OPTIMIZATION RESULTS WITH THE CALCULATIONS AND DATA OF THE SPACECRAFT FLIGHT TESTS

The flight tests (FT) are preceded by the prognosticated results /221 by employing the mathematical modeling, by using the analysis and processing the available data, by employing the diagnostics of failures and by analyzing possible emergency situations.

The basic method of test results prognostication are methods which are based on the description of mathematical models with complex physical parameters, by employing the generalization approach and by processing the experimental data on the ground.

In prognistication, one may employ the function of the following type as a mathematical model

$$y(t>t_p)=f[ay(t_p);\ x;\ \varepsilon(t>t_p)],$$

where $y(t>t_p)$ is the prognosticated parameter at the t time period which follows the preceding time period t_p; $y(t_p)$ is the basic parameter of the prognosticated process; x are the random factors, affecting the prognostication; $\varepsilon(t>t_p)$ is the prognostication error and a is the coefficient of transformation.

One should point out the three variants of prognosticated processes:

the process takes place without any significant change;

at the end of the previous process, a new approach has been developing or some new, earlier undetectable properties have been discovered;

simulataneously with the process under investigation, the development of a completely new process or phenomenon has been discovered.

The prognostication methods differ not only in terms of their /222 structure, but also in terms of the availability of scientifically substantiated data. In the majority of such methods, the concept of the appropriate model parameter extrapolation is being used. The extrapolation techniques may be based on the following principles: the description of the most reliable type of model, based on the available statistical data; the selection of the type of approximation function, based on the analysis of physical parameters of the whole process.

During the experimental ground testing, it is difficult to provide for the spacecraft the simulated weightlessness, significant vacuum conditions and the combination of dynamic factors involved. All this requires flight testing during the operational use. In parallel with such flight testing, one should also carry out the comprehensive testing of the onboard systems and the systems on the ground. On the basis of such data, one refines the degree of correlation between the stands and flight testing and as a result, some tests on the ground may be corrected.

The program of the spacecraft fligth testing must include the following: the purpose and goal of testing, the amount of testing, the methodical plan which includes and types and order of tests, the computer programs for processing and analyzing the measured results, the substantiated requirements in terms of reliability, the substantiated operational parameters of the module, the required volume of technical documentation and the reports on the flight tests which have been conducted.

The flight test program must be constructed in such a way as to make it possible, by using it, to bring about and realize the standard external load factors for all major computed operational modes. Keeping in mind the random character of a number of external factors and the great cost of flight experiments, the calculation and conditions of computing the load factors must always be properly planned.

The flight tests however also have their own drawbacks. The major ones are: high cost, somewhat lower amount of information obtained because of the limited number of measuring devices and telemetry channels, the limited possibility of controlling the physical processes, compared to tests on the ground.

The flight test is a final stage of the spacecraft design, in the course of which one completely removes all inaccuracies which have been determined while optimization work was done on the ground and one also refines the tactical and technical parameters of the spacecraft and of its systems. The FT purpose is to verify the spacecraft functioning while in use and to execute all operational functions, in accordance with the technical assignments, to verify the compatibility of all spacecraft subassemblies during the standard use, to verify whether the experimental optimization on the ground is sufficient and effective, to refine the operational conditions, to substantiate the fact that all requirements in terms of reliability and appropriate paramters are satisfied, which were obtained during experimental optimization on the ground, prior to the acutal use.

By using the FT, one also substantiates and determines the tactical, technical and other parameters, namely: the actual parameters and scattering of values with respect to the nominal ones, /223 the possible emergency situations, the trajectory parameters, the parameters of the propulsion unit operation, the parameters which characterize the dynamics of the separation of various stages and various components, the load factors along the spacecraft periphery, the frequencies and shapes of internal elastic oscillations and their logarithmic decrements, the parameters of the control system operation and of other systems, the calculated cases of load applications and response of the module to the external actions, the proper selection of the spacecraft dynamic schematic, the agreement between the module parameters obtained during the experimental studies on the ground and the flight parameters, the proper selection of the composition and range of the measured parameters, the definition of proper placement of primary converting devices and the list definition of the control and measuring equipment.

On the basis of FT, one sometimes may have to do some additional work on the construction design components or on some other spacecraft systems. After conducting such additional work, the experimental verification of the proper resolution of the problem must be carried out on the ground.

The measured results of the parameters during the flight tests make it possible to evaluate the standardized functions of the design loads in the module which is being investigated, taking into account the specifically assigned reliability levels.

The probability that the highest levels of the load function H_{im}, obtained during the test of m devices, will be exceeded even by one order of magnitude in the subsequent tests of k number of such devices, will be on the order of

$$P = \left\{ H_{im}(k) > H_{im}(n) \right\} = 1 - \frac{n}{n+k} .$$

It is clear that in having a small number of n, this probability will be rather high, in other words, only the information from a large number n of start-ups may be used as a basis for quantititive evaluation of possible maximum values of loads applied $H_i(t)$ or $H_{ij}(t)$.

Figures 19.1 and 19.2 show the relationships of the axial and transverse load factors (n_x, n_y, n_z) as a function of time, during tests on the ground and the FT calculations, as the device enters the Venus or Martian atmosphere.

By comparing these curves, one can see that in the case of devices which enter the dense atmosphere of Venus, the maximum load factors in the operational and calculation instances correspond to the load factors obtained theoretically (the deviation is not more than 5%).

In the operational case, the limiting pulse I_{lim} and the measure of integral load factor effect M_{ℓ} during ground tests is greater than in the case of calculations ($\theta_{en}=20\pm5^{\circ}$). The relationships which were obtained by means of the downward computations are equidistant to the relationships obtained on the basis of operational data. The excess of limiting pulse I_{lim} and of the integral load factor effect, are explained by the appropriate power output of the centrifugal assembly unit, with the resulting decrease in the overload gradient. A greater rigidity in testing device facilitates the increase of its reliability without any significant increase in the mass of the device. Because of increased logarithmic decrement of fluctuations during the experiment, the quantitative and qualitative modeling of the actual event is attained by double simulation of decreasing /224 oscillations.

The results obtained for the devices which enter the rarified atmosphere of Mars (see Figure 19.2) indicate that the experimental relationships n=f(t) of the operational and computed cases correspond to the theoretical data for analogous cases of the computed limiting load ($\theta_{en}=-14\pm4^{\circ}$). Consequently, the magnitude of the overload n, the C gradient, the limiting pulse I_{lim} and the integral overload factor M_{ℓ} during testing are identical to the theoretical results.

The magnitudes of transverse overloads n_y, n_z, of the frequencies f and of the angle variations ϕ during the spacecraft experimental studies correlate quite well with the computed data.

All curves in terms of the flight testing are within the area of possible computed overloads and do not exceed the operational overload limits, in other words, the FT loads do not exceed the overloads which were determined during the study of the device on the ground.

As one can see, by comparing the test results by using the methods developed, and employing the computational methods and flight tests, one can draw a conclusion as to the applicability of these methods not only to evaluate the tests, but also to study the basic dynamics of the device, as it enters the atmosphere.

During the landing of the mock-up of a device of Venera-14 type, it first impacts against the ground with the upper point of its shock-absorbing shell, acquiring rotational motion. In the course of 0.4-0.6 s it changes its pitch angle from θ_0 to θ_{max}. At the next moment in time, the mock-up impacts the ground with the lower /225 point of the shock-absorbing enclosure. At this moment, the upper point is above the ground. During the time of the lower point enclosure impact, the mock-up may either change direction of its rotation and then the upper point will again touch the ground and the mock-up will acquire a stable position, or such change may not occur and then the mock-up, without touching the ground by the upper point, will tilt over.

The results of studies indicate that the mock-up is stable on the simulated models of the soil in the case of A, B, C models in all assigned test modes and on the platform of D model - only by using specific modes (see Chapter 16). The stability on simulated soils of the A, B, C models is associated with the fact that a considerable part of the landing device kinetic energy is absorbed by the ground.

The increase of landing speed to V_{max}=1.1 V_{nom} affects only slightly (~6%) the pitch angle θ_{max}, as the mock-up is dropped on the simulated soils of A, B, C models.

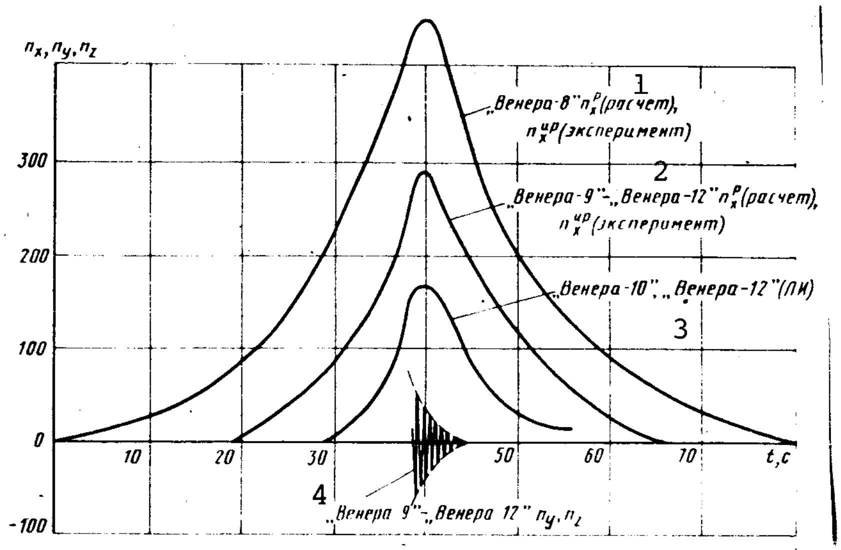


Figure 19.1. Overload factors n as a function of time during flight tests (FT) by using the calculations and by maximizing the operation of Venera devices on the ground.

Key: 1. Venera-8 n_x^p(calculations), n_x^{up}(experiment);

2. Venera-9 - Venera-12 n_x^p(calculations), n_x^{up}(experiment);

3. Venera-10 - Venera-12 (FT);

4. Venera-9 - Venera-12 n_y, n_z

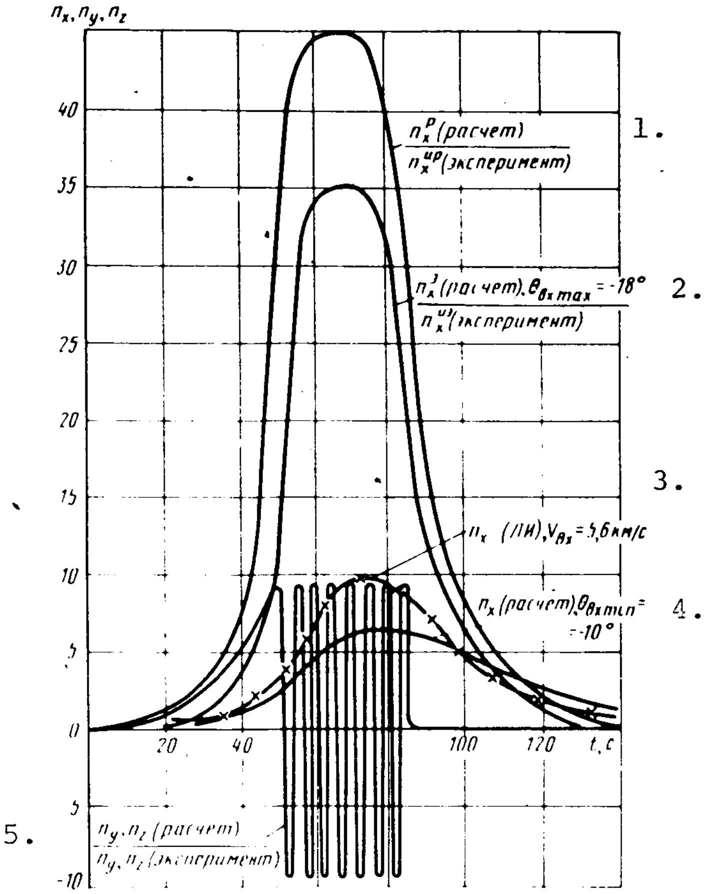


Figure 19.2. Overload factor n as a function of time t during the test flight (FT), by using the calculations and testing the Mars module on the ground.

Key: 1. n_x^p(calculations)/n_x^{up}(experiment);

2. n_x^j(calculations), $\theta_{en\ max} = 18^o$/n_x^{up}(experiment);

3. n_x(FT),V_{en}=5.6 km/s; 4. n_x(calculations), $\theta_{en\ min} = 10^o$;

5. n_y, n_z(calculations)/n_y, n_x(experiment)

Figure 19.3 shows the maximum pitch angle as a function of initial mock-up orientation $\theta_{max} = f(\theta_0)$, when the landing velocity is constant V_{nom}. The tests results indicate that the end position /226 of the mock-up depends significantly on the approach angle toward the landing site and on the carrying capacity of the soil. For example, in landing the mock-up on the test site in the case of

D model, when $\theta \geq 0$ and $V \geq V_{nom}$, the mock-up will tilt over (Figure 19.3).

Figure 19.4 shows the anglular velocity ω_z at the moment of the first impact of the mock-up with the soil, as a function of the carrying capacity of the ground. With the increase of the carrying capacity (model D) the stability of the mock-up will decrease. The experimental and theoretical results, studying the effect of the moment of inertia I_z, of the vertical velocity V_v, of the height of the center of masses x_{cm}, of the gravity acceleration g and of the mass M, affecting the landing module, indicate that the increase of the landing module mass M and the increase in the gravity acceleration g will improve the stability of the device. The increase of vertical velocity during the landing and the moment of inertia with respect to the transverse axis of the device will reflect negatively on the stability. The increase of dimensionless $\bar{R}=R/c_{cm}$ quantity (by increasing the radius R or by decreasing the center of height of the device center of mass x_{cm}) results in improved stability and increased of permissible landing velocity V.

The density of planetary atmosphere has a complicated effect on the dynamics of soft landing [34]. On one hand, the increase of buoyancy force results in the deterioration of the module stability, on the other hand, the high atmospheric density results in the appearance of virtual mass, which is equivalent to the increase in the forces of inertia of the device and of its mass.

Figure 19.5 shows the comparison of calculated overload factor n as a function of the magnitudes of compression of the shock-absorbing shell and the experimental data. As the shock-absorbing shell is compressed by 140-145 mm, the overload factor does not exceed $n \leq 200$. In the case of extremal velocities and the imact angles with a rigid soil by the D model, a part of the energy is absorbed by the mock-up design components - the struts, side frame, which results in the increase of the load factor ($n \geq 200$) and residual deformations.

The calculations of landing on hard ground correlate satisfactorily with the experimental data. The disagreement between experimental and theoretical data does not exceed 10%, and is used as a safety factor which is the result of permissible tolerances which were used during the theoretical studies, some constraints which were imposed on the number of tests and some errors in measuring the parameters.

In the spacecraft Venera-13 and Venera-14, the measurement of impact overload factors which act on the spacecraft during its landing on the surface was accomplished by using the modified measuring system Bison-M. The comprehensive system Bison was used /227
during the flights of Venera-11 and Venera-12 to obtain the data

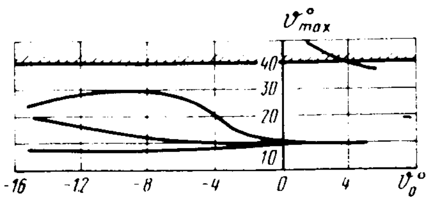


Figure 19.3. Maximum pitch angle as a function of initial orientation of the mock-up $\theta_{max}=f(\theta_0)$.

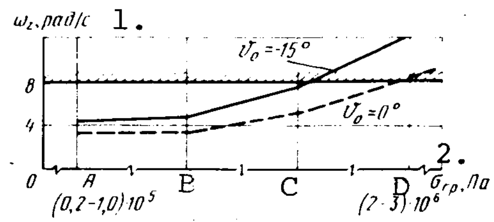


Figure 19.4. Angular velocity ω_z at the moment of the first impact of the mock-up against the surface as a function of the carrying capacity of the simulated ground and V_{nom} = =const.

Key: 1. rad/s; 2. β_{gr}, Pa

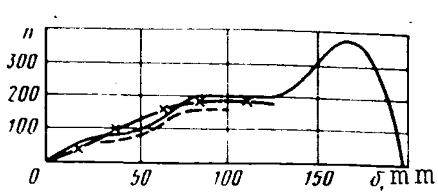


Figure 19.5. Comparison of calculated overload factor n and the experimental data as a function of the shock-absorbing compression of the shell
x----x experiment (NKU); ____ calculated; ----experimental (+T)

in regard to the overload factors during the entry of the device into the atmosphere. The Bison-M system could function on the Venera-13 and Venera-14 in two operational modes: a) in measuring the aerodynamic over-load factors at the height range between 100 and 65 km and b) to measure the impact overloads as the device made contact with the ground.

The system for recording the impact pulse load factors incorporated the following components: the primary converters of the impact pulse accelerations (the piezo sensors of AVS-03404 type), the sensor of inertia of DP-M300-74 type; the electron assembly, consisting of control block, conversion, processing and data accumulation blocks.

The sensors were placed near the center of masses of the descending device and their sensitivity axes were oriented in the following manner: along the spacecraft X axis - the DP_X inertia sensor and one of the AVS_X piezo sensors, and along the Y axis - the second AVS_Y piezo sensor.

In accordance with the functional logic of the measuring system in the Bison-M system, after recording the load factors during aerodynamic braking and storing this data, the system was converted to the second operational mode and was measuring the impact accelerations.

As a result, it was possible to obtain the magnitudes of the impact load factors with the time interval between the experimental points of 1 ms.

The data about the change in time of the axial component of the impact load factor $n_{x\,im}(\tau)$ during the landing of Venera-13 and Venera-4 is shown in Figure 19.6.

After processing the experimental data, determined during the flight tests (FT) it was possible to obtain the true picture of the impact processes in the case of the landing of both devices (the dot-and-dash line and a solid line in Figure 19.6, pos. 2). The lateral overloads $n_{yim}(\tau)$ which were recorded during the landing of Venera-13 and Venera-14 were not significant (Figure 19.7, pos. 2), enabling us to make the assumption that the devices have landed virtually vertically. The data obtained by using the DP_x sensor (Figure 19.7, pos. 1) and distorted somewhat because of inertia forces, may be used to estimate the maximum axial impact overload and the nature of the impact process.

To ensure the verification of the experimental data reliability, a program of studies on the ground was developed, the purpose of which was to determine the level and nature of the loads on the device during its landing, in order to tune-up properly the Bison-M system in the necessary measuring range, to obtain the calibration data about the impact processes by using the drop-off tests on the simulated soil of Venus and to verify the reliability and correctness of operational logics and of the measuring system. /228

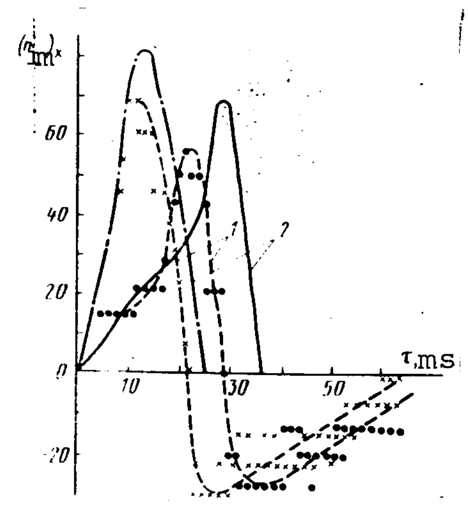


Figure 19.6. Measured results of the impact overload factors during the Venera-13 and Venera-14 landings.
1 - experimental curves of the axial impact overloads, constructed on the basis of telemetric data; 2 - true overloads obtained after proper processing of the curves (1).

An actual mock-up prototype was subjected to tests which had the mass and inertia parameters of the Venera-13 and Venera-14 devices, mounting the Bison-M system (Figure 19.8).

The following models of simulated landing sites were used: aglophorite sand (porous soil, compressed), aglophorite sand with crushed rock, particle size 30-80 mm and foam concrete.

During testing, the mock-up was dropped off on the horizintal and inclined sites (the inclination angle varied between 0 and 10°), using different impact velocities (4-8.5 m/s) and different angular positions of the device.

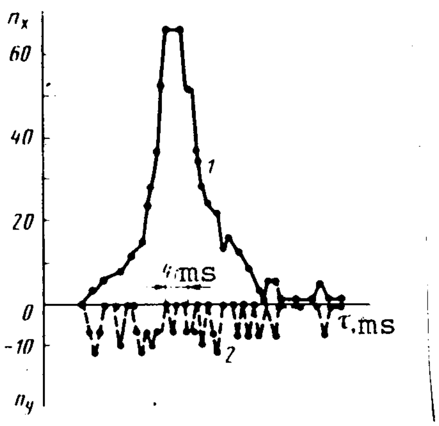


Figure 19.7. Impact, recorded by the DP_x sensor - the axial impact of Venera-14 (1) during soft landing and the lateral impact load factors, recorded by AVS_y sensor on Venera-13 during its soft landing (2).

Regardless of measurements which were made by Bison-M assembly, the overload impact factors were measured by using the standardized laboratory equipment and the readings from this equipment were assumed to be the reference readings (the data was recorded digitally and by two coordinate recording systems).

In comparing the test results, as obtained by Bison-M measuring system, some differences were observed in terms of durataion and amplitude of the impact pulses.

The reason for such a difference in the input signal is the rather low input impedance of the gain and measuring circuit [51].

The comparison of the recorded impact processes and their analysis made it possible to test the efficiency of data processing, recorded by the Bison-M system, and to obtain the true parameters of the impact pulse ($n_{x\,im}$, τ_{im}) by employing geometric construction (Figure 19.9)). /229

By analyzing a large number of experimental data, obtained on the ground, a good correlation was obtained between the data obtained by the above-described method, utilizing the Bison-M equipment, and the reference data. The data obtained during the flight tests was processed analogously and has defined the impact parameters during the landing of Venera-13 and Venera-14 with a high degree of reliability (see Figure 19.6):

for Venera-13 $(n_{x\,im})_{max}$=81, and τ_{im}=25 ms;

for Venera-14 $(n_{x\,im})_{max}$=68, and τ_{im}=36 ms.

Let us note that during the landing of both spacecraft on the surface of Venus, after the impact against the ground, the devices have bounced.

Since the tests on the ground have shown that the deformation of the landing gear will not take place in the presence of the impact load factors at the levels obtained during the flight experiment (see Figure 19.6, 19.7) the landing device, in its first approximation, may be viewed as a solid body, interacting with the yielding landing surface. The braking process of the landing device by the ground was recorded during a time period of 60 ms, by the measuring equipment Bison-M.

Figure 19.10 shows a good correlation between the impact pulse obtained during the vertical drop-off of the spacecraft mock-up on the compressed aglophorite sand (curve 2) with the speed of ~8 m/s (which corresponds to the landing speed on Venus (~7.2 m/s) and the data, /230
recorded during the Venera-13 landing on the planetary surface (marked off by the dots, pos. 1). The landing speed of Venera-13 is somewhat greater - ~7.6 m/s, which explains the somewhat higher maximum overload factor.

Although the frontal impact process during Venera-13 landing was not fully recorded, one might claim that it practically coincides with the front end of curve 2, because an apparent agreement between the curves is displayed after the maximum overload application. In addition, by analyzing Figure 19.10, one can see that in the area of the Venera-13 landing, the soil was approximately of the same kind and with the same parameters as in the case of the aglophorite sand. (Aglophorite is a porous, compressed sand)

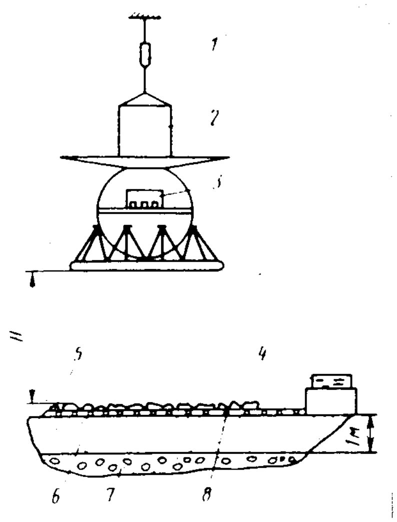


Figure 19.8. Schematic diagram of the drop-off tests on the ground of the natural spacecraft mock-up prototype: 1 - lock; 2 - mock-up; 3 - Bison system; 4 - Control and measuring instrument; 5 - foam concrete blocks; 6 - aglophorite sand; 7 - concrete; 8 - crushed rock.

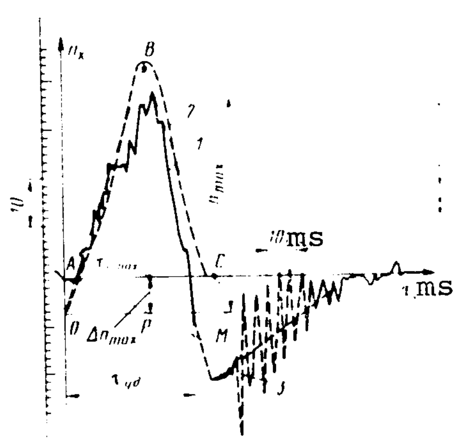


Figure 19.9. Schematic depiction of the impact oscillogram processing, as recorded by Bison-M system (1); 2 - refrence laboratory data; 3 - recording of after-effect pulse; ABCMOP - reference points of graphic representation.

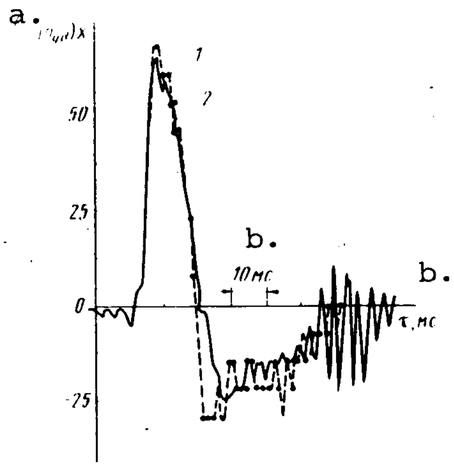


Figure 19.10. Comparison of impact pulse, obtained during Venera-13 landing (1) with the results of ground tests (2) where a mock-up was used. The measurements were conducted by Bison-M system.
Key: a. $(n_{im})_x$; b. ms

The expanses of space are unlimited. The human race finds itself at this time only at the beginning of its study and mastery. Scientists intend to obtain some new information about celestial bodies in the nearest time - at the periodic approach of Halley's comet to the Sun. The scientists of many countries are meticulously preparing themselves to meet this rare event. For example, the European Space Agency (ESA) specialists have develped a probe, according to Giotto program, which has a mass of 49.2 kg and the new device - Planet-A, of 10-15 kg mass has been developed in Japan. /231

In our country, as early as 1984, the unique international space project Vega was initiated and in the development of scientific equipment, the experts from the People's Republic of Bulgaria, the German Democratic Republic, the Viet Nam People's Republic, Polish People's Republic, Czechoslovak Socialist Republic, Austria, France and the Federal Republic of Germany took part. The Vega project contemplates the flight of the interplanetary probe in March of 1986, at a distance of ~10,000 km from the comet's nucleus, in other words a flight through its gaseous and dust atmosphere (its coma).

The scientific studies and measurements will be carried out by a variety of instruments, the total mass of which on the probe exceeds 130 kg. Since the relative speed of the spacecraft flight in the vicinity of the comet will reach 78 km/s, and the moment of encounter will be quite short (~0.1 s) the control functions of all scientific instruments will be conducted autonomously, by the control systems on board.

All three subprograms supplement each other, aiming at the direct study of Halley's comet, namely: to study the structure of the nucleus, of the comet's plasma and atmosphere, to determine the size, mass, temperature, chemical and physical composition of the nucleus, the magnitude of magnetic and electric fields, to elucidate the causes of explosions and expulsion from the nucleus of dust particles and of gas, the determination of interaction between the comet and the solar wind, etc.

The flight programs within the framework of the Vega project provides for the two identical spacecraft which first will fly toward Venus and then, after flying aroung the Sun and trajectory correction - the flight in the direction toward Halley's comet. At the present time, the problems related to the long-term study of the atmosphere of Venus have been solved for the first time in history by using the automated Venus balloons. The two aerial balloons of 3.4 m diameter and 21 kg mass have each conducted its own unique research work in the Venus atmosphere between the 11th and 15th of June, 1985, drifting across the Venus sky at heights of 53-55 km in the equatorial region of the planet. It is also known that the descending modules from Vega-1 and Vega-2, after successful soft landing, have completed all planned scientific research programs by direct study of the surface and of the soil properties in the new regions of the planet.

Naturally, while brining about such multifaceted space program, which is the Soviet project Vega, the already available extensive experience in both modeling of the spacecraft functioning in various flight stages and optimization of scientific instruments on the ground, of all systems and of the spacecraft as a whole, were put to use.

Such approach makes it possible to increase appreciably the reliability and efficiency of the programs which are provided by the whole project, to develop extremely "clever", "enduring" and universal space robots.

Appendix. Scale of quantities which were used in modeling spacecraft dynamics of descent and landing

/232-/233

Parameter	Scale designation	Separation in vacuum: Defining formula	Separation in vacuum: $K_\rho=1$; $K_E=1$, $K_l=1/2$	$K_l=1/10$	Landing on the ground and movement: Defining formula	Moon $K_j=6$, $K_l=1/6$, $K_E=1$	Flight through the atmosphere: Mars, Mercury $K_j=5/2$: $K_l=2/5$, $K_E=1$	$K_l=2/5$, $K_E=2/5$	Venus, $K_j=1$, H=0 km: $K_l=1$, $K_E=1$	$K_l=1/15$, $K_E=1$	Venus, $K_j\approx1$: H=54 km, $K_l=2$, $K_E=2$	H=30 km, $K_l=1$, $K_E=1/10$	Mars, H=0 km, $K_j=5/2$: $K_l=2/5$, $K_E=1$	$K_l=1$, $K_E=5/2$	Jupiter H=-20km $K_j=3/8$: $K_l=1/3$, $K_E=1/2$
Time	K_t	$K_l(K_\rho/K_E)^{1/2}$	1/2	1/10	$K_l^{1/2}K_j^{-1/2}$	1/6	2/5	2/5	1	$(1/15)^{1/2}$	$(2)^{1/2}$	1	2/5	$(2/5)^{1/2}$	$(8/9)^{1/2}$
Acceleration	K_a	$K_E(K_\rho K_l)^{-1}$	2	10	–	–	–	–	–	–	–	–	–	–	–
Mass density	K_ρ	–	–	–	$K_E(K_lK_j)^{-1}$	1	1	2/5	1	15	1	1/10	1	1	4
Mass	K_M	$K_\rho K_l^3$	1/8	10^{-3}	$K_E K_l^2 K_j^{-1}$	1/216	8/125	16/625	1	1/225	8	1/10	8/125	1	4/27
Force	K_F	$K_E K_l^2$	1/4	10^{-2}	$K_E K_l^2$	1/36	4/25	8/125	1	1/225	8	1/10	4/25	5/2	1/18
Moment of inertia	K_I	$K_\rho K_l^5$	2^{-5}	10^{-5}	$K_E K_l^4 K_j^{-1}$	$(1/6)^5$	$(2/5)^5$	$(2/5)^6$	1	$1/(15)^4$	32	1/10	$(2/5)^5$	1	$4/(3)^5$
Energy	$K_Э$	$K_E K_l^3$	1/8	10^{-3}	$K_E K_l^3$	1/216	8/125	16/625	1	$1/(15)^3$	16	1/10	8/125	5/2	1/54
Pressure (intensity)	K_p	K_E	1	1	K_E	1	1	2/5	1	1	2	1/10	1	5/2	1/2
Linear velocity	K_V	$(K_E/K_\rho)^{1/2}$	1	1	$(K_lK_j)^{1/2}$	1	1	1	1	$(1/15)^{1/2}$	$(2)^{1/2}$	1	1	$(5/2)^{1/2}$	$(1/8)^{1/2}$
Angular velocity	$K_{\dot\theta}$	$(1/K_l)(K_E/K_\rho)^{1/2}$	2	10	$K_j^{1/2}K_l^{-1/2}$	6	2,5	2,5	1	$(15)^{1/2}$	$(1/2)^{1/2}$	1	2,5	$(5/2)^{1/2}$	$(9/8)^{1/2}$
Angular acceleration	$K_{\ddot\theta}$	$K_E(K_\rho K_l^2)^{-1}$	4	100	$K_jK_l^{-1}$	36	25/4	25/4	1	15	1/2	1	25/4	5/2	9/8
Linear stiffness	K_C	K_EK_l	1/2	1/10	K_EK_l	1/6	2/5	4/25	1	1/15	4	1/10	2/5	5/2	1/6
Power	K_N	$K_EK_l^2(K_E/K_\rho)^{1/2}$	1/4	10^{-2}	$K_E(K_jK_l^3)^{1/2}$	1/36	4/25	8/125	1	$1/225\sqrt{15}$	$8\cdot(2)^{1/2}$	1/10	4/25	$(125/8)^{1/2}$	$1/36\sqrt{2}$

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ABSTRACT

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~~This work presents~~ A comparative study is presented of the methods developed for the calculation of thermal contact resistance between two surfaces submitted to a perpendicular heat flux. Several factors affecting this resistance are analised and a brief historical of the works in this field is made, spotting the methods of interest for spacial applications. These are compared to experimental data so as to establish the most proper method for the couplings of the first Brazilian satellite.

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16. Abstract
The book considers the fundamental aspects of modeling of the spacecraft characteristics by using the computing means. Particular attention is devoted to the design studies, the description of physical appearance of the spacecraft and simulated modeling of spacecraft systems. The fundamental questions of organizing the on-the-ground spacecraft testing and the methods of mathematical modeling are being presented.

The book aims at design engineers, spacecraft designers and test personnel.

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