Sounding Rocket Preliminary Design

Nickolay Zosimovych

Abstract — This article aims at studying sounding rocket (SR) class, or those vehicles launched from the earth and carrying out various experiments at high altitudes. After the end of their mission, rockets should deliver a payload to a given point or to give range.

In general, design tasks are divided into direct and inverse ones. This paper focuses on direct designing tasks. In this respect, the set values are maximum range, payload mass and restrictions on the sounding rocket construction. As a result, a SR general design technique is proposed. This technique includes selection of the SR initial parameters, the number of stages, the relative weights of the fuel components, the specific thrust of the engines for each stage and the initial transverse load on the SR. After selecting the fuel composition, engine design features and rocket as an object, the starting mass can be represented as a complex mathematical function. Moreover, predetermined maximum range, rocket payload, selected fuel components, structural design, materials, design parameters allow for a definite SR weight determination. Finally, overall and power characteristics of the SR are proposed within this technique. Such characteristics are accepted as design parameters.

Index Terms — Rocket, design, structure, parameter, sounding rocket, solid rocket motor (SRM), liquid rocket motor (LRM).

I. INTRODUCTION

Nowadays guided rockets have different classifications. In addition, their tasks and design procedures are varied significantly. Moreover, modern space technology is characterized by an increasing complexity of the tasks to be solved by modern space vehicles. The efficiency in solution of such problems significantly depends upon technical characteristics of the on-board engineering systems ensuring the functioning of the spacecraft. Due to the relatively small weight of modern scientific instruments, even “seemingly small increase of payload weight can significantly extend the program of research and experiments implemented by that spacecraft” [1].

The tasks and design procedures volumes may vary significantly. In this research studying sounding rocket (SR) class. That vehicle launched from the Earth and used for solving various experiments at the high altitude (Fig. 1). After a mission solution, rockets should deliver a payload to a given point or to a given range. In general, design tasks are divided into direct and inverse ones. In this publication namely the direct task was solved. As set values are accepted maximum range L_{max} , payload mass m_{pL} , and restrictions on the sounding rocket construction.

In practice, in the process of SR creating, instead of the costs, the SR starting mass is considered as a value proportional to the costs, under certain assumptions.

Fig. 1. The Maxus 5 sounding rocket on the launch pad at Esrange on 31 March 2003 [2].

The technique for solving the direct design problem provides for the following:

- initial selection of design parameters and fuel;
- SR weight analysis;
- SR ballistic design.

In the design process used tables and graphs obtained on the basis of experience and research design typical ballistic guided rockets developed previously [3]–[8]. It should be noted that the design process is iterative, i.e., the results repeatedly reviewed and refined. The proposed further design method may be used in carrying out experimental studies preceding the initial stages of the SR design.

Computer simulations processes enable rapid implementation of all design stages for varying conditions interactively.

In Fig. 2 shows a general design diagram of SR ballistic type. The content of the algorithms used by the modules shown in the diagram are described in subsequent publications [8]–[10].

Fig. 2. SR general design technique.

II. THE INITIAL STAGE OF SELECTING SOUNDING ROCKETS DESIGN PARAMETERS

A. Design Parameters

For the adopted mission program for the active part of trajectory, the SR speed at the end of this part and the flight range depend on the following parameters:

- 1) the number of stages n ;
- 2) the relative weights of fuel stages μ_{Ki} ;
- 3) the specific engine thrust for each stage P_{Spec_ni} ;
- 4) thrust-to-weight ratio steps, characterized by coefficients λ_{ni} ;
- 5) the initial load on the SR transverse midsection P_{M1} ;

In other words, the SR range can be represented as a next function:

$$L = f_1(\mu_{Ki}, P_{Spec_ni}, \lambda_{ni}, P_{M1}), i = 1, 2, \dots, n. \quad (1)$$

In this case, the following two factors must be taken into account:

1. The specific thrust of the engine P_{Spec_ni} depending on fuel composition, the engine design and magnitude of the pressure in the combustion chamber p_e , and in the outlet section of the nozzle p_a .

The relative weight of fuel in the stages μ_{Ki} be expressed by the duty ratio of the first stage fuel μ_{K1} , namely [11]:

$$\begin{cases} \mu_{K2} = \chi_1 \mu_{K1} \\ \mu_{K3} = \chi_2 \mu_{K2} = \chi_1 \chi_2 \mu_{K1} \\ \dots \\ \mu_{Kn} = \chi_{n-1} \mu_{Kn-1} = \chi_1 \chi_2 \dots \chi_{n-1} \mu_{K1} \end{cases},$$

where $\chi_w = \frac{\mu_{Ki+1}}{\mu_{Ki}}$ - the weight ratio of adjacent stages.

Then we can represent the function (1) in a new form:

$$L = f_1(\mu_{K1}, \chi_{i-1}, \lambda_{ni}, p_{Ki}, p_{ai}, P_{M1}), i = 1, 2, \dots, n. \quad (2)$$

The same relationship can be viewed as an inverse function of the form:

$$\mu_{K1} = \bar{f}_1(L, \chi_{i-1}, \lambda_{ni}, p_{Ki}, p_{ai}, P_{M1}), i = 1, 2, \dots, n. \quad (3)$$

After choosing the composition of the fuel, the design features of the engines and the SR as a whole, the starting mass can be represented approximately as a function. Accordingly, for SR with liquid rocket engines (LRE) and

solid rocket engines (SRE):

$$m_{01} = f_2(m_{PL}, \mu_{Ki}, \lambda_{ni}), i = 1, 2, \dots, n. \quad (4)$$

$$m_{01} = f_3(m_{PL}, \mu_{Ki}, p_{Ki}, p_{ai}, \bar{l}_{3i}, u_i), i = 1, 2, \dots, n. \quad (5)$$

From the analysis of the SR coefficient of thrust-weight ratio with the SRE should be as function of values m_{0i} , P_{Spec_ni} , $\bar{l}_{3i}$, λ_{ni} , μ_{Ki} , $(u\rho_T)$ and do not an independent parameter:

$$\lambda_{ni} = \frac{m_{0i}}{P_{Spec_ni}} \cdot \frac{1}{u_i \rho_{Ti}} \cdot \frac{1}{K_s \bar{l}_{3i}} \cdot \left(\frac{\psi_i \bar{l}_{3i}}{m_{0i} \mu_{Ki}} \right)^{\frac{2}{3}}.$$

Initial transverse load on the midsection for SR with the SRE can be represented as:

$$P_{M1} = m_{01}^3 \sqrt{\frac{\psi_1 \bar{l}_1}{m_{01} \mu_{K1}}}.$$

Thus, the SR starting mass with SRE can be considered as a function:

$$m_{0i} = f_3(m_{PL}, L, \chi_{i-1}, p_{Ki}, p_{ai}, \bar{l}_{3i}, u_i, \rho_{Ti}), i = 1, 2, \dots, n. \quad (6)$$

In the expression (6), except for setting the maximum mission range of payload masses and - the type of fuel selected, constructive scheme and materials for SR design and manufacture, includes parameters that may uniquely identify the SR weight, sizes and thrust. They are all accepted as are design parameters.

For SR with LRE, the design parameters are: $n, \chi_{i-1}, \rho_{Ti}, \lambda_{ni}, p_{Ki}, p_{ai}, P_{M1}$, and for SR with SRE are $n, \chi_{i-1}, p_{Ki}, p_{ai}, \bar{l}_{3i}, (u\rho_T)$.

2. Therefore, it should find the optimum combination of design parameters by the minimum starting SR mass criterion. However, due to the first of the factors noted above, we choose the fuel.

B. Fuels Selection and Their Characteristics Determination

Among the few energy sources that are used to create jet propulsion, chemical molecular fuel still has the widest application.

In the SR design process necessary to select the optimum composition of existing components of fuels on the following criteria [12]:

- 1) standard specific thrust P_{Spec_st} ;
- 2) thermodynamic parameters of the combustion products (gas constant R , temperature T , the adiabatic index k or isentropic index n_{iz});
- 3) average fuel density ρ_f ;
- 4) the burning rate (for SRE) u , or the residence time of the combustion products in the chamber (for LRE) τ ;
- 5) the characteristics of mechanical strength of solid fuels with regard to their rheology;
- 6) cost characteristics of fuel components.

C. Fuels Selection and Their characteristics Determination

The standard specific thrust is a characteristic energy of

fuel capacity. Theoretical specific thrust is standard at equilibrium flow of combustion products and at high pressures in the combustion chamber $p_{ch} = 40 \text{ bar}$ and at the nozzle exit $p_a = 1.0 \text{ bar}$. This operation of the nozzle is assumed in design mode.

The specific thrust values for various fuel components are obtained as a result of thermodynamic calculations, then that results are tabulated and entropy diagrams are calculated too. To account for losses in the combustion chambers, a standard specific thrust P_{Spec_st} to be reduced by 4-5%. Thus, the standard thrust shown as:

$$P_{Spec_st}^{br} = (0.95 \div 0.96) P_{Spec_st}. \quad (7)$$

For solid fuels containing aluminum, it is necessary to take into account additional losses in specific thrust caused by the two-phase gas flow, i.e., in presence of gaseous combustion products of particulate matter Al_2O_3 .

The standard specific thrust in this case is calculated by the following formula [8]:

$$P_{Spec_st}^{br} = P_{Spec_st} [(1 - 17a + 0.009a^2) \times 10^{-2}], \quad (8)$$

where a – the aluminum content is in weight percent.

In case of the absence of tables and entropy diagrams, the engine specific thrust at the calculated mode of the nozzle operation can be calculated by the following empirical dependencies [8]:

a) for liquid fuels:

$$P_{Spec_st}^{calc} = P_{Spec_st} + 21 + 0.76p_{ch} - 0.003p_{ch}^2 - 70p_a + 25p_a^2; \quad (9)$$

b) for solid propellants:

$$P_{Spec_st}^{calc} = P_{Spec_st} + 19.4 + 0.76p_{ch} - 0.003p_{ch}^2 - 70p_a + 25p_a^2. \quad (10)$$

In formulas (9) and (10), the specific thrust is measured in seconds, and the pressure in the combustion chamber and at the nozzle exit are in bars.

The specific thrust in a vacuum is given by [8]:

$$P_{Spec_vac} = P_{Spec}^{calc} + \frac{RT}{g_0^2 P_{Spec}^{calc}} \left(\frac{p_a}{p_{ch}} \right)^{\frac{k-1}{k}}, \quad (11)$$

where R – the gas constant; T – combustion temperature; k – adiabatic exponent.

The calculated dependence for determining specific thrust at any altitude has the next form:

$$P_{Spec_vac} = P_{Spec}^{calc} + \frac{RT}{g_0^2 P_{Spec}^{calc}} \left(\frac{p_a}{p_{ch}} \right)^{\frac{k-1}{k}} \left(1 - \frac{p_h}{p_a} \right), \quad (12)$$

where p_h – the air pressure at an altitude above the surface of the Earth.

Specific thrust on the ground is calculated at the atmospheric pressure of $p_N = 1.01$. To calculate the function

$\left(\frac{p_a}{p_{ch}} \right)^{\frac{k-1}{k}}$ designer may use the table at Ref. [13].

The combustion temperature of the fuel can be calculated by the formula [13]:

$$T = T_{st} + 1.12(p_{ch} - 40), \quad (13)$$

where T_{st} – standard combustion temperature determined by the results of thermodynamic calculations; p_{ch} – pressure in the combustion chamber (bar).

Formula (13) is valid under the temperature condition: $3000 \text{ } ^\circ K < T_{st} < 3500 \text{ } ^\circ K$.

The gas constant R and the adiabatic exponent k are weakly dependent on the pressure in the combustion chamber. Therefore, only their standard values R_{st} and k_{st} are used for calculations.

The density of the liquid fuel is calculated by the formula [10]:

$$R_{st} = \frac{(1+K)\rho_{ox}\rho_f}{\rho_{ox}+K\rho_f}, \quad (14)$$

where ρ_{ox} – oxidant density, $\frac{kg}{m^3}$; ρ_f – fuel density, $\frac{kg}{m^3}$; K – ratio of the second consumption of oxidizer and fuel agents.

The solid propellant density is determined from the next expression:

$$\rho_f = \sum_{j=1}^{j=n} \rho_j g_j, \quad (15)$$

where ρ_j – is the density of the j – source component; g_j – j - weight fraction of component ($j=1,2,...,n$).

The density of modern solid propellants covers a relatively narrow range: from 1680 to 1800 $\frac{kg}{m^3}$.

The most important characteristic of solid propellants is their burning rate. The burning rate is determined by the nature of the fuel, the ratio of its components, and essentially depends on external factors: the pressure in the combustion chamber p_{ch} and the initial temperature of solid propellant grain T_{PG} .

The dependence of burning rate on pressure in the chamber is expressed by the empirical expressions [9]:

$$u = ap_{ch}^v, \quad (16)$$

$$u = b + ap_{ch}^v. \quad (17)$$

The dependence of the burning rate on the initial temperature of the solid propellant grain is characterized by the temperature coefficient of the burning rate:

$$\alpha_T = \frac{\Delta u}{\Delta T_{PG}}, \quad (18)$$

where Δu – change in combustion rate; ΔT_{PG} – interval changes the initial solid propellant grain temperature.

For modern fuels $\alpha_T = 0.02 - 0.03$ [10].

For solid rocket propellants, in addition to the chemical composition, it is necessary to select the solid propellant grain shape. When choosing a solid propellant grain shape, consider:

1. The combustion surface remains constant during engine operation.
2. Large second gas flow.
3. Large fill factor of the combustion chamber with propellant.
4. Simplicity of manufacturing technology and installation of the solid propellant grain.

In practice frequently used solid propellant grain telescopic form, solid propellant grain with longitudinal slits and with star-grain channel (transitioning) (Fig. 3, [10]).

Fig. 3. Cross-sectional Views of Different Solid Propellant Grain Configurations. The outer black line is the motor casing, red colored area is the propellant grain, and white areas denote the region into which the gas generated flows. The STS RSRM refers to the Space Shuttle Redesigned Solid Rocket Motor [10].

The burning of such solid propellant grains occurs on the internal surfaces, while the external, and sometimes end surfaces, are covered with an inert reservation. The solid propellant grain may be bonded to the inner surface of the combustion chamber.

The main geometric characteristics of the solid propellant grain:

- 1) the outer diameter of the propellant grain d_g ;
- 2) relative solid propellant grain length:

$$\bar{l}_g = \frac{l_g}{d_g}, \quad (19)$$

where l_g – length of the charge;

- 3) the relative diameter of the inner channel (for solid propellant grain with slots and star-grain channels):

$$\bar{d}_{sg} = \frac{d_{sg}}{d_g}, \quad (20)$$

Where d_{sg} – the inner channel diameter;

- 4) the relative length of the slits to the charge with longitudinal slits (Fig. 4):

$$\bar{h}_{sg} = \frac{h}{d_g} = 0.3\bar{h}_{sg} - 0.3, \quad (21)$$

During ballistic design of SR with SRMs, the difference

between the outer and inner diameters of the cylindrical part of the combustion chamber and the outer diameter of the charge can be neglected. Therefore, for the n -th stage, we can take

$$d_{gi} = d_i. \quad (22)$$

Fig. 4. Solid propellant grain with longitudinal slots.

The relative diameter of the inner channel (if the absence of constraints on the strength of the solid propellant) is chosen from the next condition:

$$\frac{A_{free}}{A_{crit}} = \frac{d_g^2}{d_{crit}^2} \leq 1.5, \quad (23)$$

where A_{free} – area of the fuel-free cross section of the combustion chamber before starting the engine; A_{crit} and d_{crit} – area and diameter of the nozzle critical section.

For a solid propellant grain bonded to a combustion chamber, provided that there are limitations on its strength, the internal diameter of the channel d_g is selected from the ratio:

$$d_K = (0.3 \div 0.5)d_g. \quad (24)$$

III. RESULTS AND DISCUSSION

As a first approximation, it is possible to use the graphs that describe the dependence of the SR range on the ratio of the starting mass to the payload mass and the number of stages, calculating the SR starting mass for a given mission range for a different number of stages (Fig. 5-7). In this case, the most profitable number of stages can be selected.

Fig. 5. Range versus $\frac{m_0}{m_{PL}}$ for single-stage SR.

Fig. 6. Dependence mission range from $\frac{m_0}{m_{PL}}$ and the number of stages with Solid Rocket Motors.

Fig. 7. Solid propellant grain with longitudinal slots Dependence mission ranges from $\frac{m_0}{m_{PL}}$ and the number of stages SR with Liquid Rocket Motor.

IV. CONCLUSION

Analysis charts (Fig. 5-7) are leads to the following conclusions:

1. For a single-stage SR, a significant increase in flight speed V_{max} and range L occurs only until $\frac{m_0}{m_{PL}} \approx 50$; with a relative weight of the fuel $\mu_k = 0.90 \div 0.92$ single-stage SR with LREs can reach mission ranges 5,000÷6,000 km.
2. For single-stage SR with SRMs, the possible mission ranges are much lower, since the specific thrust of modern solid propellant rocket engines is 10-15% less than the specific thrust of the LRE and SR with SRMs having lower fuel filling values 11.
3. It can be considered that ballistic possibilities for single stage SR and for 2-stages SR with solid rocket propellants with practically identical.
4. In the range of distances, single-stage and two-stage SRs with LREs have practically the same starting mass ($\frac{m_0}{m_{PL}} \approx 6 \div 7$); in the same mission ranges, the same starting masses ($\frac{m_0}{m_{PL}} \approx 7$) have two- and three-stage SRs with solid propellant rocket engines.
5. Two-stage SRs with LREs can provide flight for almost any range, including intercontinental ones, at $\frac{m_0}{m_{PL}} \approx 50$ SR are able to bring a payload into a basic circular orbit.
6. Three-stage SRs with LREs, starting from ranges $L \approx 2,000 \div 3,000$ km, have a slightly lower starting mass

than two-stage ones. However, the gain in the starting mass does not exceed 10%.

7. Increasing the SR with SRM number of stages over 3 does not significantly increase in flight distance or decrease the starting weight.
8. Increasing the SR number of stages with SRM in excess of 3 results in no significant increase in flight distance or the reduction of the starting weight.
9. SR reliability with the number of stages is markedly reduced.
10. Given these factors, the most advantageous number of stages n for SR is usually one less than the number of stages at which it has a starting mass.
11. After selecting the number of stages, from the graph it is easy to find the first approximation of the SR starting mass m_0 .

REFERENCES

- [1] G. Seibert. (2006). The History of Sounding Rockets and Their Contribution to European Space Research. *ESA Publications Division, Netherlands*. Available online: https://www.esa.int/esapub/hsr/HSR_38.pdf.
- [2] R. Matthew Tetlow. (2003). Commercial Launch Vehicle Design and Predictive Guidance Development. *The University of Adelaide, Australia*. Available online: <https://www.digital.library.adelaide.edu.au>.
- [3] N. Zosimovych, *Commercial Launch Vehicle Design*, LAP LAMBERT Academic Publishing, 2016.
- [4] R.F. Appazov, S.S. Lavrov, V.P. Mishin, *Ballistika Upravlyayemykh Raket Dalnego Deystviya*. Science Publisher, Moscow, 1966.
- [5] R.F. Appazov, O.G. Sytin, *Metody Proektirovaniya Traektoriy Nositeley i Sputnikov Zemli*. Science Publisher, Moscow, 1987.
- [6] A.A. Lebedev, V.T. Bobronnikov, M.N. Krasilschikov, *Statisticheskaya Dinamika i Optimizatsiya Upravleniya Letatelnykh Apparatov*. Mashinostroeniye Publisher, Moscow, 1985.
- [7] T.W. Lee, *Aerospace Propulsion*. John Wiley&Sons, USA, 2014.
- [8] V.P. Mishin, V.K. Bezverby, B.N. Pankratov, *Osnovy Proektirovaniya Letatelnykh Apparatov (Transportnye sistemy)*. Textbook, Mashinostroeniye Publisher, Moscow, 1985.
- [9] P.M. Sforza, *Theory of Aerospace Propulsion*. Elsevier, UK, 2017.
- [10] G.P. Sutton, O. Biblarz, *Rocket Propulsion Elements*. 8 ed., John Wiley&Sons, Canada, 2017.
- [11] N. Zosimovych. (April 2017). Modeling of Spacecraft Centre Mass Motion Stabilization System, *Int. Ref. Journ. of Eng. and Science (IRJES)*. [Online]. 7(4). pp. 34-41. Available online: https://www.researchgate.net/publication/316273577_Modeling_of_Spacecraft_Centre_Mass_Motion_Stabilization_System.
- [12] N. Zosimovych, "Sounding Rockets Scattering Factors and their Influence on Motion Accuracy", presented at the 4th Int. Conf. on Aeronautical Materials and Aerospace Eng. (AMAE 2020), Chengdu, China, May 14-17, 2020.
- [13] *Tablicy standartnoy atmosfery*, GOCT Standard 4401-1964.

Dr Nikolay Zosimovych was born in former Soviet Union, on Ukraine Republic, Ukrainian, 1962. Graduated his all degrees (Bac., Ms and Ph. D), Ph.D. in Aerospace Engineering (1998) at Moscow Aviation Institute (National Research University), Moscow, Russian Federation.
Areas of scientific Activity:
• Designs and manufacture equipment, specialized systems and components that are supplying to the aerospace and mechatronic fields.

•Parametrical optimization, computation and structural designing of drones and aerospace flying vehicles.

Professional Experience:

2020-now: School of Robotics and Intelligent Manufacturing Ecosystem, Xi'an Jiaotong-Liverpool University, Suzhou, Jiangsu Province, China.

Professor.

2017-2020: Department of Mechatronics Engineering, School of Engineering, Shantou University, Shantou, Guangdong, China.

Professor.

2015–2017: Department of Aeronautical and Mechanical Engineering,
Faculty of Computing Engineering and Science, University of South
Wales, Pontypridd, UK.

Senior Lecturer (Fixed Term).

2013-2015: Department of Geoinformation and Space Systems, Sergey
Korolyov Zhytomyr Military Institute, Zhytomyr, Ukraine.

Professor.

Bibliography: 5 books, 128 scientific publications, 14 scientific and
technical reports and 3 methodical manuals.

Advisor: 38 MS's, 1 current MS, 1 BTech and 1 PhD students.

HONG KONG SOCIETY OF MECHANICAL ENGINEERS. ID:
m20180507001.

Honorary XJTLU Association, XJTLU Administration, 25/01/2021 to
30/11/2023, Associate Number: 711623.