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SPACE ROCKET FOR AIR-ROCKET SYSTEM

Koncepcja rakiety kosmicznej dla zestawu rakietowo-lotniczego do wynoszenia ładunków na orbitę okołoziemską

Abstract: *The traditional and most used method of launching payloads into Earth orbit is to launch a carrier rocket from the surface. An alternative method of space transport is launching payloads into Earth's orbit using air-rocket (air-assisted) systems. The concept of mixed space transport involves launching a space rocket with a payload from an aircraft or other reusable platform-carrier at a specific altitude above the Earth's surface. The air-rocket system enables the launch of small satellites while reducing ground infrastructure and costs. Such a method seems promising in the context of relatively cheap, mobile and responsive small payload launch systems. It is an interesting proposition and beneficial especially for countries without convenient conditions to build their own spaceport. The paper analyses the fundamental requirements for the design of rockets used in this type of system and performs preliminary calculations of a conceptual two- and three-stage rocket capable of lifting a payload of 10 kg into a zero-inclination orbit at an altitude of 500 km. The analyses carried out were based on available research reports in this area.*
Key words: air-assisted rocket launch system, space rocket, rocket design.

Streszczenie: *Tradycyjnym i najczęściej wykorzystywanym sposobem wynoszenia ładunków na orbitę ziemską jest start rakiety nosiciela z powierzchni Ziemi. Wzmożone zainteresowanie eksploracją przestrzeni kosmicznej oraz gwałtowny rozwój przemysłu zajmującego się produkcją małych satelitów wymagają dedykowanych systemów wynoszenia ładunków dostosowanych do indywidualnych potrzeb klienta. Alternatywnym sposobem transportu kosmicznego jest wynoszenie ładunków na orbity Ziemi za pomocą lotniczo-rakietowych systemów. Koncepcja mieszanego transportu kosmicznego zakłada start rakiety kosmicznej z ładunkiem użytecznym z samolotu lub innej platformy – nosiciela wielokrotnego użytku na określonej wysokości nad powierzchnią Ziemi. W artykule dokonano analizy wymagań stawianych konstrukcji rakiet wykorzystywanych w tego rodzaju systemach, a także wykonano wstępne obliczenia koncepcyjnej rakiety dwu- i trzystopniowej zdolnej do wyniesienia*

ładunku o masie 10 kg, na orbitę o zerowej inklinacji i wysokości 500 km. Przeprowadzone analizy wykonano w oparciu o dostępne raporty z badań w tym zakresie.
Słowa kluczowe: lotniczo-rakietowy system, rakieta kosmiczna, projektowanie rakiet

1. Introduction

Launching a carrier rocket from the surface is the traditional and most used method of launching payloads into Earth orbit. An alternative space transport method is launching payloads into Earth's orbit using air-rocket (air-assisted) systems [1-5]. The concept of mixed space transport involves launching a space rocket with a payload from an aircraft or other reusable platform-carrier at a specific altitude above the Earth's surface. The air-rocket system enables the launch of small satellites while reducing ground infrastructure and costs. Such a method seems promising in the context of relatively cheap, mobile and responsive small payload launch systems. It is an interesting and beneficial proposition, especially for countries without convenient conditions to build their own spaceport [6-12]. The paper analyses the fundamental requirements for the design of rockets used in this type of system. What is more, it performs preliminary calculations of a conceptual two- and three-stage rocket capable of lifting a payload of 10 kg into a zero-inclination orbit at an altitude of 500 km. The analyses carried out were based on available research reports in this area.

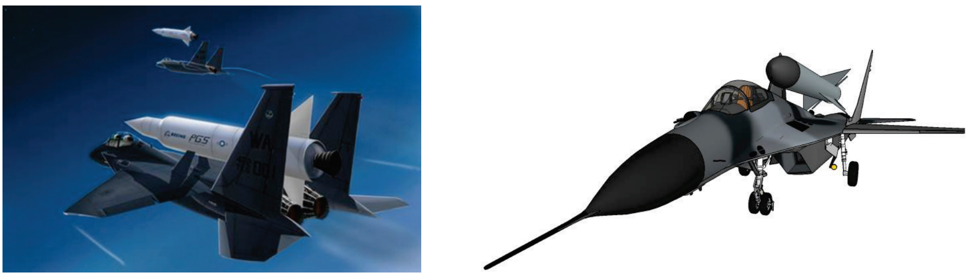


Fig. 1. Air-assisted rocket systems: on the left Boeing concept [6] and the right MUT version [7-12]

2. Characteristics of space launch rockets

The fundamental component of the space launch system, regardless of the concept (traditional or alternative), is the space rocket, whose task is to transport the payload to a specific orbit. The rocket's characteristics are significantly influenced by the variability of its flight conditions due to its movement through different layers of the atmosphere. This makes it very important to determine the optimal flight conditions for the rocket. This issue usually boils down to determining the optimal operation programmes for the propulsion unit and controlling the direction of the velocity vector. As a prerequisite for determining the programme of operation of the propulsion unit, it is necessary to determine the target orbit to which the payload is to be launched, which is usually the basis for determining the rocket parameters. The basic parameter characterising a space rocket is the payload mass that the rocket can lift into orbit at a given altitude and inclination. In addition, the number of stages, the velocity gain achieved, the dimensions, launch mass and stage separation conditions are also important parameters.

Knowing the target orbit makes it possible to determine the required incremental rocket velocity needed to lift a payload of a given mass. Optimalisation of the maximum incremental velocity allows us to determine rocket parameters. One way to increase the achievable velocity is to divide the rocket into stages. The stages of a split rocket, each of which usually has its own propulsion unit, are placed on top of the other to form the rocket's airframe. Once the fuel has burned out, the stages are discarded one by one, thus reducing the mass of the remaining rocket. Using the same amount of thrust, the use of split stages enables a higher speed gain than a single-stage rocket.

An additional benefit of dividing the rocket into more stages is that it simplifies the issue of developing an optimal programme of work for the propulsion unit. The air-rocket system provides the space rocket with a launch from less dense layers of the atmosphere and a non-zero initial velocity. This gives a certain advantage compared to rocket launches from the Earth's surface. On the other hand, regardless of the method, the rocket launch takes place in an atmosphere with a significant density compared to a vacuum, which involves significant drag on movement. To reduce the velocity loss of the rocket, its flight path is programmed to rise as quickly as possible beyond the dense layers of the atmosphere. For this reason, the rocket's thrust at launch should be as high as possible. As the ceiling increases, the thrust of the rocket should decrease at the expense of a higher velocity of the emitted gas stream. Such programming of the thrust of a single engine is a complex issue. Introducing the division of the rocket into several stages with separate propulsion units makes it possible to adapt the characteristics of the engines of the individual stages to the conditions in which they will operate.

The performance of a rocket can be expressed in terms of the payload mass that can be carried into the target orbit. Space rockets used to date are characterised by a percentage ratio of payload mass to total rocket mass between 1% and 4% [1]. The value of this ratio depends on, among other things, the characteristics of the propulsion unit, the mass of the rocket, the launch site, the altitude of the target orbit and the mission profile.

3. Conceptual design of a space rocket

The initial conceptual design is for developing a space rocket equipped with a three-stage solid-fuel propulsion system capable of launching satellites weighing 10 kg into Low Earth Orbit (LEO) using a combat aircraft [1, 7].

The design of a space rocket is a very complex issue and requires considering many factors affecting its design. Most often, space rockets are designed based on specialised programmes. The rocket design should start with analysing the target orbit and determining the initial rocket parameters. To this end, computer simulations of the flight trajectory are carried out, considering varying environmental conditions. This, in turn, makes it possible to determine the velocity losses and begin the process of optimising the rocket subassemblies to achieve the velocity gain required for the payload launch mission. The values of the basic parameters that a rocket should approximately have to perform a mission to LEO successfully have been determined based on a study by M. W. van Kersten [1].

3.1. Assumptions for an air and missile system

The lack of space rockets designed for launching payloads in the 10 kg category has led to the research being based on slightly larger and heavier existing rockets and test reports. This approach seems sufficient at the conceptual design stage. The study adopted rocket drop conditions consistent with the design of the air-rocket system being implemented at the Military University of Technology in Warsaw (MUT)[1]: drop ceiling $\mathrm{h_0} = 15\,\mathrm{km}$, airdrop velocity of the combat aircraft (rocket carrier) $\mathrm{v_0} = 250\,\mathrm{m/s}$, approximately equal to the initial velocity of the rocket, rocket launch angle $\gamma_0 = 50°$, rocket total mass $\mathrm{M_0} = 930$ kg, including payload $\mathrm{M_L} = 10$ kg, rocket length $\mathrm{L_R} = 5.4$ m and diameter $\mathrm{D_R} = 0.56$ m [1, 7].

In this paper, the analysis of the propulsion unit was mainly carried out in terms of solid-fuel rocket engines. Using such engines has certain limitations compared to liquid fuel engines, but also advantages in terms of simplicity of construction, ease of operation and lower production costs. The contemporary Pegasus XL rockets demonstrate the viability of using solid-fuel rocket engines in an air-to-rocket system [13, 14, 15].

Analysis of propellants using NASA Chemical Equilibrium Applications (CEA) [1] showed marginal (<1%) differences in performance for different types of solid fuels [1]. Hence, it is reasonable to consider a single propellant. The most common fuel used in solid rocket engines is hydroxyl terminated polybutadiene (HTPB), a polybutadiene terminated with functional hydroxyl groups with a composition of 19% aluminium, 12% HTBP and 69% AP (aammonium perchlorate), and this propellant was adopted in this work.

3.2. Target orbit

The Low Earth Orbit (LEO) group occurs in the altitude range of 180 to 2 000 km. Most research and many weather satellites in space orbit closed orbits from this group. The

[1] The paper is the result of the University Research Project entitled: *Airborne-Rocket Launch System for Delivering Satellite Payloads into Low Earth Orbit – Feasibility Study*, funded by the Polish Ministry of National Defense and conducted at the Military University of Technology (MUT) in years 2018-2022 [7-12].

choice of target orbit also involves considering the launch site of the rocket. In the case of air-rocket systems, it is advantageous to conduct launches over vast areas of water bodies. The geographical location of Poland makes it possible to conduct such a launch from the Baltic Sea, which has been considered in this work. Considering the above considerations, a circular target orbit of altitude $\mathrm{h} = 500$ km and zero inclination has been assumed for the Earth radius $\mathrm{R_z} = 6371$ km, the radius of the target orbit $\mathrm{r} = 6871$ km.

3.3. Rocket flight dynamics

The magnitude of forces acting on a rocket significantly affects its design. Associated with the forces are the overloads that occur during a rocket flight. The forces acting on a rocket of mass M moving at a speed of v include rocket thrust-T, gravity-F_G and aerodynamic forces such, as drag-F_D and lift-F_L [1].

Assuming:

$$F_D = C_D \cdot \frac{\rho \cdot v^2 \cdot A_f}{2} \tag{1}$$

$$F_L = C_L \cdot \frac{\rho \cdot v^2 \cdot A_f}{2} \tag{2}$$

where A_f - reference surface area of aerodynamic forces and moment, C_D - drag coefficient, C_L - lift coefficient.

The following formula can describe the equation of motion of the rocket:

$$M(t) \cdot \frac{dv}{dt} = T(t) \cdot cos\,\alpha - M(t) \cdot g \cdot sin\,\gamma - C_D \cdot \frac{\rho \cdot v^2 \cdot A_f}{2} \tag{3}$$

$$M(t) \cdot v \cdot \frac{d\gamma}{dt} = T(t) \cdot sin\,\alpha - M(t) \cdot g \cdot cos\,\gamma - C_L \cdot \frac{\rho \cdot v^2 \cdot A_f}{2} \tag{4}$$

3.4. Number of rocket stages

The next stage of rocket design is to determine the number of stages. Reference was made to the basic equation characterising the rocket in the absence of the influence of the gravitational field to analyse the dependence of the number of stages of the rocket on the speed increment achieved. This relationship is called the Tsiolkovsky equation and makes the rocket's velocity increment dependent ΔV on the effective velocity of the outflow of gases in vacuum $v_{eq_{vac}}$ and the ratio of the initial mass M_0 to final mass of the rocket M_b (after fuel burnout), denoted as R_M.

$$\Delta V = v_{eq_{vac}} \cdot ln\frac{M_0}{M_b} = v_{eq_{vac}} \cdot ln \cdot R_M \tag{5}$$

Assuming an initial rocket mass of 930 kg (including structure of 93 kg and a payload of 10 kg), and an effective gas jet velocity of 3048 m/s, the velocity gain for the single-stage rocket was calculated: $\Delta V = 6707$ m/s.

To compare the effect of the division of the rocket into stages, on the value of the speed gains achieved, the speed gains of two- and three-stage rockets with the same initial parameters as for a single-stage rocket were calculated. The division of the rocket into stages was considered according to two criteria. In the first case, equal structural and fuel masses were assumed, while in the second equal structural and payload coefficient of all rocket stages. For the first case, the calculated velocity gain of a two-stage rocket for the mass distribution shown above is: $\Delta V = 8250\ m/s$, while for the three-stage rocket: $\Delta V = 9082\ m/s$. For the second case, the calculated velocity increment of the two-stage rocket is: $\Delta V = 9987$ m/s, while for a three-stage rocket, the speed increment obtained is: $\Delta V = 11025\ m/s$.

Regarding the achieved speed gains, the calculations show a clear advantage of two- and three-stage rockets over single-stage rockets. In addition, the more optimal method of mass distribution was the one that assumes equality of the mass ratio and the structural and payload coefficients.

Dividing a single-stage rocket into two stages generates the greatest increase in rocket speed gain. Splitting the same rocket into three stages generates a further increase in speed gain, which is less than in the first case. Increasing the number of stages beyond three is inefficient. As the number of stages of a rocket increases, the complexity of the whole structure increases, contributing to higher costs and lower reliability. For this reason, the most used rockets are two- and three-stage rockets. Hence, the remainder of this paper mainly considers such rocket configurations.

3.5. Increase in space rocket velocity

A space rocket design strictly depends on the speed increments ΔVrequired for the payload launch mission into the target orbit. Putting a satellite into orbit involves giving it the appropriate speed specific to the target orbit - the v_{sat}. To carry out a satellite launch mission, it is not sufficient to give the rocket a terminal velocity equal to that of the satellite in that orbit. This is due to the presence of velocity losses during the rocket's flight. The speed that has to be given to the rocket in order to get the payload into orbit is called the rocket velocity increment ΔV and has a higher value than the final velocity of the rocket. The velocity increment ΔV required to lift a payload into one of the LEO groups is between 8000 and 10000 m/s [16].

Factors affecting the required speed gain of a rocket include losses due to gravitational interaction Δv_g, aerodynamic drag Δv_{drag} and rocket control Δv_{cont}. In addition, the speed increase ΔV is also affected by the rotational velocity of the Earth v_{rot} and the non-zero initial velocity of the rocket v_0, which is one of the advantages of the systems [17].

Taking into account the above-mentioned components make it possible to determine the value of the increase in rocket speed ΔV required to perform the payload launch mission into the target orbit according to the following relation:

$$\Delta V = v_{sat} + \Delta v_g + \Delta v_{drag} + \Delta v_{ster} - v_0 \pm v_{rot} \tag{6}$$

Taking these factors into account makes it possible to determine the value of the speed increment ΔV required to carry out the mission. Assuming a rocket launch over the Baltic Sea basin in an easterly direction, the required rocket speed increment is:

$$\Delta V = 7614 + 1673 + 176 + 62 - 250 - 276 = 8999\ \frac{m}{s} \tag{7}$$

Comparing with the speed increment obtained from the tests quoted above, the determined increment is larger by $366\ m/s$. This difference is due to the adoption of different rocket launch sites and target orbit altitude.

4. Rocket designs

This section presents the basic parameters and characteristics of the designed space rocket for the 10 kg payload category satellite. Performances and propulsion characteristics are included in table and figures below.

4.1. Space rocket with a mass of 1064 kg

Table 1

Parameters and performance of the optimised rockets for different launch altitudes and initial speeds

Rocket parameters/performances	Altitudes [km], initial speeds [m/s]		
	10 km 250 m/s	**15 km 200 m/s**	**15 km 400 m/s**
Total rocket mass [kg]	1225.0	1064.0	807.3
Total rocket length [m]	5.478	5.438	5.388
Rocket diameter [m]	0.6649	0.5947	0.5156
Time of a fairing separation [s]	109.0	94.6	86.9
Vacuum thrust [kN]			
I stage	34.22	32.11	23.11
II stage	7.825	5.611	4.347
III stage	2.579	2.425	2.031

Table 1 cont.

Rocket parameters/performances	Altitudes [km], initial speeds [m/s]		
	10 km 250 m/s	**15 km 200 m/s**	**15 km 400 m/s**
Specific impulse [s]			
I stage	288.5	277.9	290.1
II stage	293.2	292.9	300.6
III stage	315.5	310.5	307.7
Mass of propellant [kg]			
I stage	844.7	765.4	577.7
II stage	196.1	134.1	103.3
II stage	38.54	35.28	24.44
Length [m]			
I stage	2.117	2.112	2.222
II stage	1.011	1.0676	1.0318
III stage	0.7257	0.6549	0.5759
Case diameter [m]			
I stage	0.6649	0.5947	0.5156
II stage	0.4958	0.3852	0.3358
III stage	0.2722	0.2813	0.2590
Mass [kg]			
I stage	72.29	69.47	48.24
II stage	20.39	10.48	9.518
III stage	5.195	4.445	3.278
Exhaust nozzle diameter [m]			
I stage	0.3752	0.2600	0.3540
II stage	0.1657	0.1966	0.1655
III stage	0.1612	0.1498	0.1460
Engine burn time [s]			
I stage	67.82	63.09	66.88
II stage	69.97	66.65	68.01
I stage	44.901	43.025	35.283
Separation time [s]			
I stage	8.756	1.808	7.740
II stage	452.1	521.1	558.5
Nozzle exit pressure [bar]			
I stage	0.2128	0.4888	0.1625
II stage	0.23297	0.11838	0.11525
III stage	0.5656	0.06705	0.06173
Pressure in combustion chamber I [bar]			
I stage	44.23	51.94	37.99
II stage	64.42	33.95	55.15
III stage	88.10	68.68	51.51

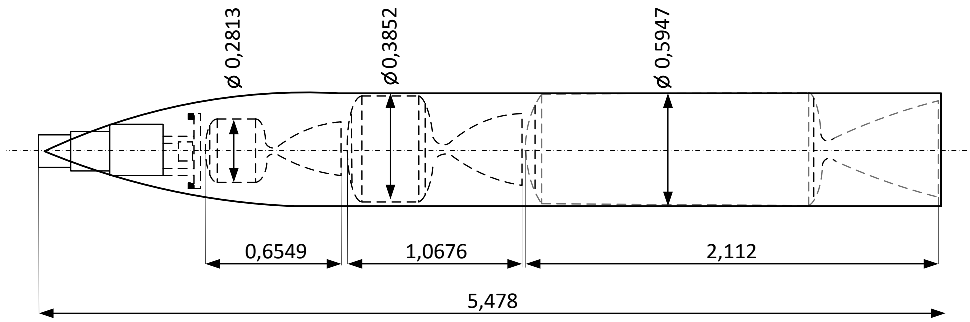


Fig. 2. Geometry of a three-stage rocket with a mass of 1064 kg, designed for a separation altitude at 15 km and initial velocity of 200 m/s (Table 10)

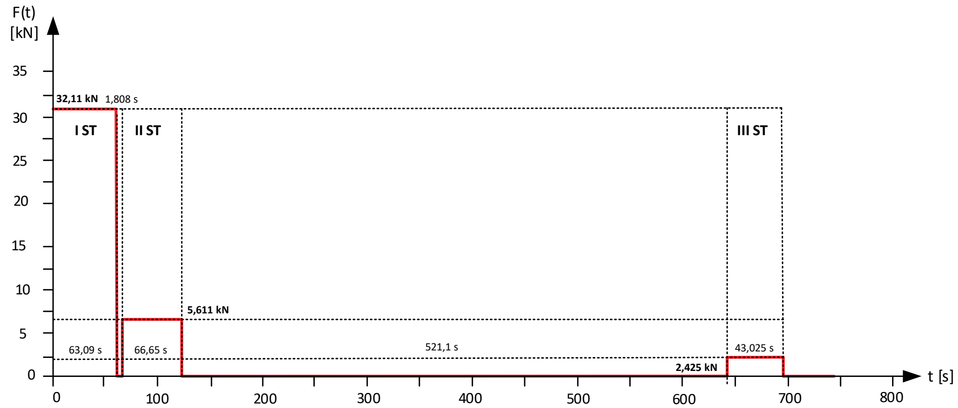


Fig. 3. Thrust variation over time of a three-stage rocket with a mass of 1064 kg

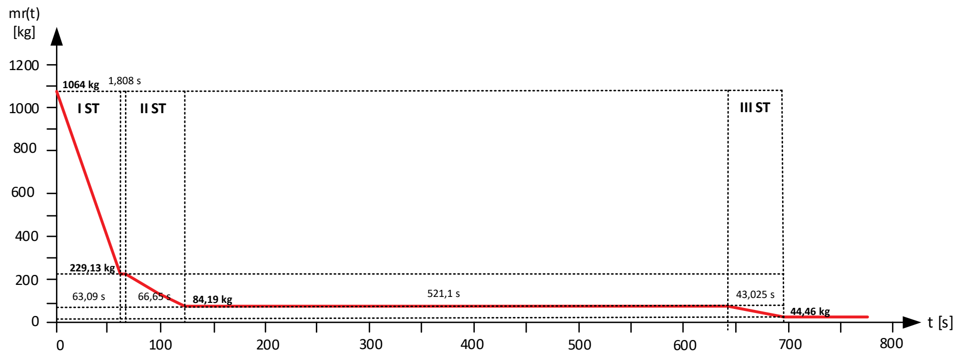
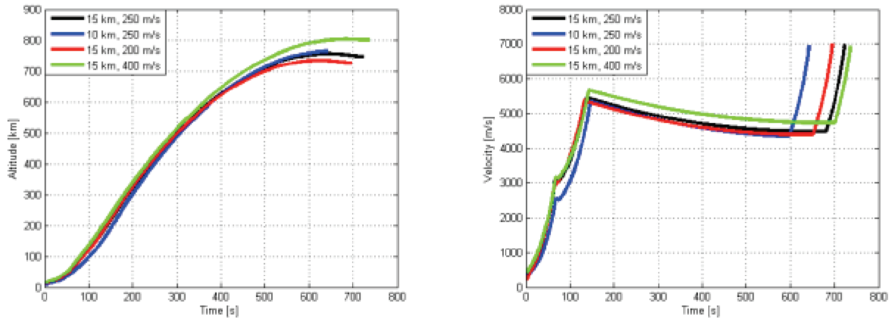
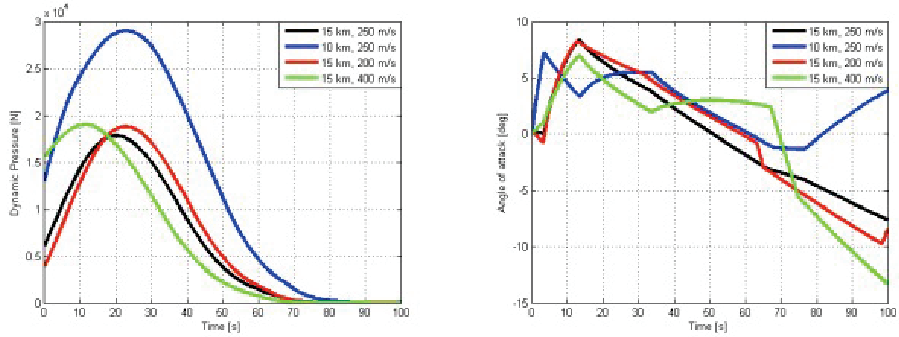


Fig. 4. Mass change over time of a three-stage rocket with a mass of 1064 kg



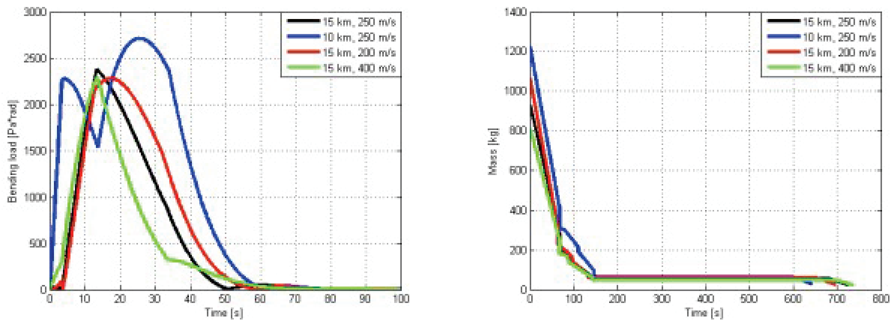
(a) Altitude versus time (b) Velocity versus time

Fig. 5. Altitude and velocity versus time for aircraft-launched rockets at selected altitudes and initial velocity



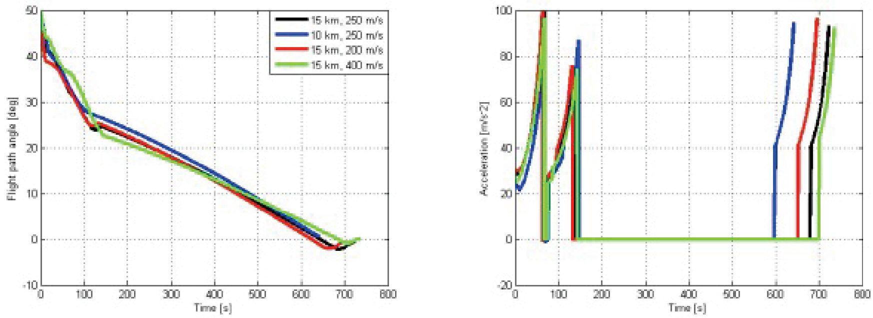
(a) Dynamic pressure versus time (b) Angle of attack versus time

Fig. 6. Dynamic pressure and angle of attack versus time for aircraft-launched rockets at selected altitudes and initial velocity



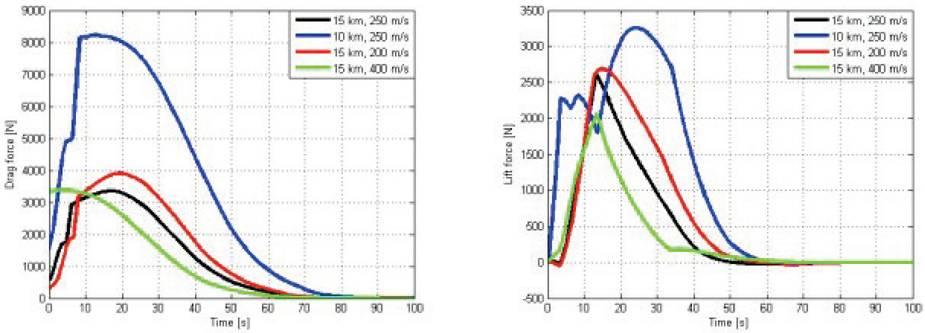
(a) Bending load versus time (b) Mass versus time

Fig. 7. Bending load and rocket mass versus time for rockets launched from aircraft at specific cases



(a) Flight path angle versus time (b) Acceleration versus time

Fig. 8. Flight path angle and acceleration versus time for aircraft-launched rockets at selected cases launch velocity



(a) Drag force versus time (b) Lift force versus time

Fig. 9. Drag and lift force over time for rockets launched from aircraft at selected cases

4.2. Rocket design – own concept

In order to check the validity of the assumptions made, verification calculations were performed. Based on the conceptual rocket data developed within the ALOSS project [18], basic rocket parameters were calculated. The results of the performed calculations differ slightly from the parameters presented in the report. The discrepancies of the obtained parameters are the largest for the rocket's third stage. This is due to considering the time-varying gas discharge rate and thrust in the ALOSS study. The observed discrepancies do not significantly affect the rocket characteristics and are acceptable at the conceptual design stage. Using the dependencies and assumptions presented in the earlier chapters, the approximate parameters of a two- and three-stage rocket capable of launching a payload into a zero-inclination orbit and an altitude of 500 km were determined. The assumed initial parameters for the calculations presented in Table 2 are based on the results of M. W. van Kersten's research [1].

Table 2

Preliminary parameters for rocket calculations

Rocket parameters	**Stage of the two-stage rocket**		**Stage of the three-stage rocket**		
	I	**II**	**I**	**II**	**III**
Combustion chamber pressure [Pa]	6000000	6000000	6000000	4000000	6000000
Pressure in the nozzle outlet section [Pa]	17680	6174	17680	8734	7174
Engine burn time [s]	60	70	60	60	40
Engine diameter [m]	0.500	0.350	0.560	0.400	0.265
Nozzle exit diameter [m]	0.380	0.220	0.375	0.205	0.141
Propellant (designation)	HTPB	HTPB	HTPB	HTPB	HTPB
Density of propellant [kg/m^3]	1850	1850	1850	1850	1850
Specific heat ratio [ ]	1.1251	1.1251	1.1251	1.1235	1.1251
Molecular mass [kg/mol]	0.029655	0.029655	0.029655	0.029506	0.029655
Combustion chamber temperature [K]	3552.93	3552.93	3552.93	3511.13	3552.93

Considering the mass and geometry limitations of the rocket adopted in the paper, the calculations resulted in the basic parameters of a two- and three-stage rocket. The results of the calculations based on the above parameters are shown in Table 3.

Table 3

General characteristics of the conceptual two- and three-stage rocket

Rocket parameter or performances	**Stage of two-stage rocket**		**Stage of three-stage rocket**		
	I	**II**	**I**	**II**	**III**
Gas mass flow rate [kg/s]	11.247	1.566	10.953	1.697	0.737
Characteristic velocity [m/s]	1575.314	1575.314	1575.314	1570.785	1575.314
Thrust coefficient [ ]	1.856	1.965	1.856	1.891	1.950

Table 3 cont.

Rocket parameter or performances	Stage of two-stage rocket		Stage of three-stage rocket		
	I	II	I	II	III
Gas discharge velocity [m/s]	2923.08	3095.03	2923.08	2970.19	3072.27
Effective gas discharge velocity in vacuum [m/s]	3101.36	3244.82	3101.36	3140.05	3225.71
Thrust impulse [kNs]	2010.92	323.84	1958.35	295.89	87.54
Specific impulse [s]	292	310	298	303	313
Specific impulse in vacuum [s]	310	324	316	320	329
Vacuum thrust [kN]	34.881	5.084	33.970	5.329	2.378
Vacuum real thrust [kN]	32.659	4.760	31.806	4.990	2.227
Mass of propellant [kg]	695.074	112.972	676.903	104.883	30.378
Case thickness [m]	0.003	0.002	0.003	0.002	0.002
Case length [m]	2.055	0.681	1.596	0.485	0.319
Case mass [kg]	15.297	2.851	16.752	2.140	0.936
Mass of insulation layer [kg]	8.591	2.116	7.663	1.840	0.802
Igniter mass [kg]	1.145	0.300	1.123	0.284	0.114
Nozzle inlet diameter [m]	0.250	0.175	0.280	0.200	0.133
Nozzle throat diameter [m]	0.061	0.023	0.061	0.029	0.016
Nozzle expansion coefficient	38	92	38	50	81
Length of the convergent part of the nozzle [m]	0.163	0.131	0.190	0.148	0.101
Half of the nozzle throat divergence angle [deg]	31.0	34.7	31.0	32.2	34.7
Half of the nozzle exit divergence angle [deg]	7.7	6.8	7.7	7.5	6.8
Length of the divergent part of the nozzle [m]	0.464	0.326	0.458	0.271	0.208
Nozzle mass [kg]	28.928	2.797	28.170	3.023	0.285
Dry engine mass (without fuel) [kg]	62.919	9.403	62.623	8.497	2.491
Length of intermediate support structure [m]	1.339	-	0.519	0.828	-

Table 3 cont.

Rocket parameter or performances	Stage of two-stage rocket		Stage of three-stage rocket		
	I	II	I	II	III
Half of the nozzle exit divergence angle [deg]	7.7	6.8	7.7	7.5	6.8
Length of the divergent part of the nozzle [m]	0.464	0.326	0.458	0.271	0.208
Nozzle mass [kg]	28.928	2.797	28.170	3.023	0.285
Dry engine mass (without fuel) [kg]	62.919	9.403	62.623	8.497	2.491
Length of intermediate support structure [m]	1.339	-	0.519	0.828	-
Mass of intermediate support structure [kg]	16.665	-	6.932	6.598	-
Stage length [m]	2.683	1.139	2.244	0.903	0.628
Stage total mass [kg]	774.7	122.4	746.5	120.0	32.9
Design factor [ ]	0.122	0.130	0.1088	0.1258	0.2496
Load factor [ ]	0.177	0.081	0.2251	0.4249	0.2594
Mass ratio [ ]	3.935	5.119	3.6692	2.5873	2.4745
Speed increment [m/s]	4004	5054	3878	2881	2840
Payload compartment diameter [m]	0.5000		0.4000		
Nose section length [m]	1.1500		1.1500		
Nose section mass [kg]	34.887		31.191		
Total rocket length [m]	5.1727		5.2264		
Total rocket launch mass [kg]	931.9		930.5		
Speed increment [m/s]	9058		9599		

The determined velocity increments of the conceptual rockets are sufficient to perform the mission of launching the payload into the target orbit. Moreover, the speed increment obtained for the three-stage rocket makes it possible to increase the mass of the payload to 15 kg.

In order to reduce the cost of developing new rocket engines, an analysis was made of the feasibility of using available solutions in the propulsion units of conceptual rockets [19, 20, 21]. During the analysis of available solid-fuel rocket motors, significant difficulties were encountered in fitting off-the-shelf motor solutions to simultaneously maintain the size and weight constraints of the rocket adopted in the paper and meet the speed gain requirements. The available small solid fuel rocket motors are usually spherical in shape, which translates into a larger diameter. For this reason, it has not been possible to fit motors that maintain the dimensional constraints adopted in the manuscript.

The analysis of the geometry and mass of the rocket was based on a simplified design (Fig. 10). As a result of the previous consideration of the achievable speed increments, the concept of using a single-stage rocket was rejected. Hence, the focus in the following article was mainly on two- or three-stage designs.

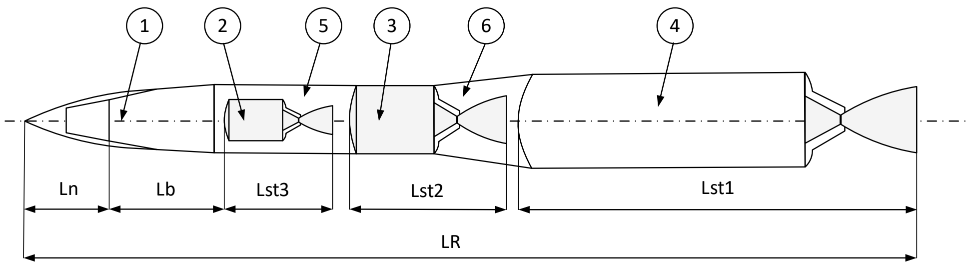


Fig. 10. The three-stage rocket design adopted in the paper: 1 - nose section; 2, 3, 4 - rocket stages (III, II, I); 5, 6 - interstage support structures

It is assumed that the rocket consists of two or three stages, placed one behind the other and terminated by a nose section. The solid-fuel rocket engines form the individual stages of the rocket, and their bodies, joined by interstage support structures, also form the fuselage support structure.

In rockets with small payloads that take off at a significant angle, the design of an additional carrier wing is mostly ignored [1, 22]. This is due to the small benefit of equipping the rocket with a load-bearing wing compared to the significant increase in fuselage complexity and cost. Based on the available test reports, a wingless design was decided upon.

The analysis of the rocket's solid geometry comes down to a consideration of the fuselage structure. The shape of this component has a significant impact on the aerodynamic characteristics of the rocket. Taking into account that the fuselage houses the other rocket assemblies and takes all the loads from external and internal forces, its design should be characterised by maximum usable volume, with the most favourable aerodynamic shapes and the required strength and stiffness [22].

The most common is a hull with a slender rotating body shape. The nose section is often conical, pyramidal, or rounded, while the rest is cylindrical. Parabolic and pyramidal nose sections have the least resistance and the largest interior volume, with equal lengths and diameters. Slightly inferior characteristics are presented by the conical nasal part. This is due to a relatively sharp edge at the junction with the cylindrical part, which interferes with the free flow of air [1, 23].

Using Solid Edge software, models of the conceptual rockets were made based on the determined parameters and are shown in the figures below.

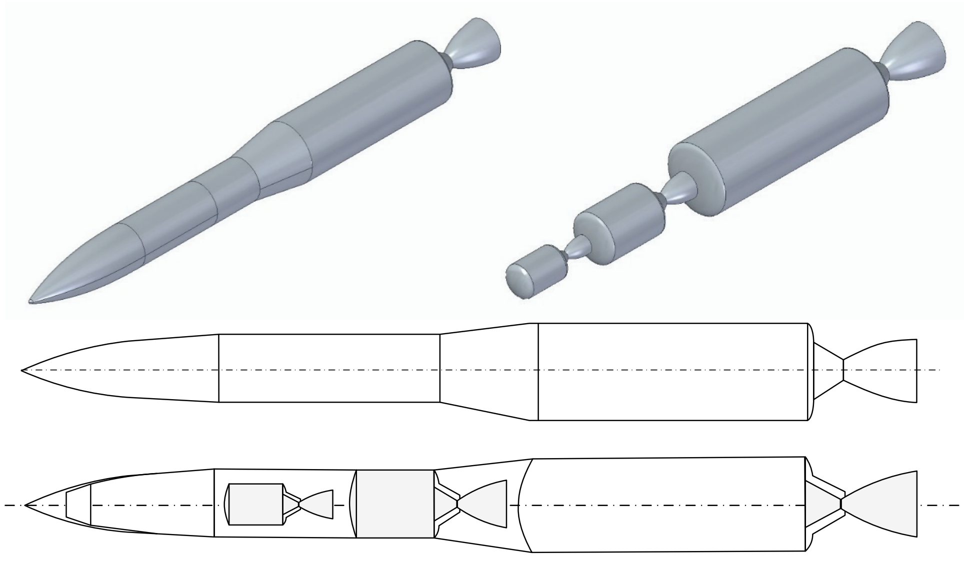


Fig. 11. Overviews and drawings of the designed conceptual three-stage rocket

In the case of the two-stage rocket concept, no suitable solid-fuel rocket engines were found to achieve the required speed gain, with a rocket launch mass not exceeding 1,000 kg. In contrast, for the three-stage rocket concept, the combination of using three engines: STAR 30BP, STAR 27 and STAR 13B (Fig. 11) seems to be the most promising. The use of these models results in a reduction in length while increasing the diameter and launch mass of the vehicle.

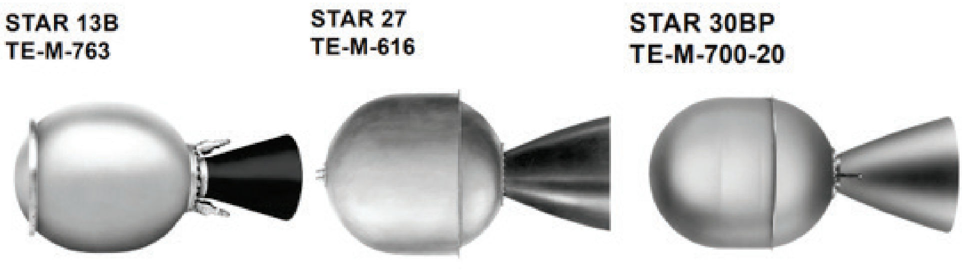


Fig. 12. The motor models used in the three-stage rocket concept [24]

Based on the technical data of the above-described engines in the Northrop Grumman catalogue [24], the basic parameters of the rocket were determined. The mass of the interstages and the dimensions of the nose section were assumed to be the same as for the three-stage rocket presented earlier.

Table 4

Parameters of a three-stage concept rocket using models of available solid-fuel rocket engines

Rocket parameters and performances	Stage		
	I	II	III
Rocket engine – motor [22]	STAR 30BP	STAR 27	STAR 13B
Maximum pressure in combustion chamber [Pa]	4102382	3881750	6446601
Working operating time [s]	55	37.3	16.1
Motor diameter [m]	0.762	0.693	0.344
Nozzle output diameter [m]	0.584	0.485	0.203
Propellant (designation)	TP-H-3062	TP-H-3135	TP-H-3340
Gas discharge velocity [m/s]	2892.95	2850.63	2810.86
Thrust impulse [kNs]	1461.040	950.985	115.876
Specific impulse [s]	295	291	287
Maximum thrust [kN]	30.893	28.202	9.608
Mass of propellant [kg]	505.034	333.605	41.224
Case mass [kg]	13.832	10.703	2.540
Nozzle throat diameter [m]	0.068	0.069	0.030
Nozzle expansion coefficient	73.70	48.80	49.80
Nozzle mass [kg]	15.330	9.250	1.680
Engine mass (without fuel) [kg]	37.690	27.480	5.800
Mass of intermediate support structure [kg]	6.932	6.598	-
Length of nose section [m]	1.1500		
Mass of nose section [kg]	31,191		
Stage length [m]	1.5062	1.2370	0.6378
Total stage mass[kg]	549.653	367.687	47.029
Design factor	0.1026	0.0927	0.2430
Load factor	0.7688	0.1767	0.1928
Mass ratio	2.0299	4.3685	2.7372
Velocity increment [m/s]	2048	4203	2830
Total motor length [m]	4.731		
Total motor mass [kg]	995.5		
Overall velocity increment [m/s]	9081		

The results demonstrate the feasibility of using the available solid-fuel rocket motor designs for the space rockets under consideration. However, inaccuracies in the results obtained, arising from the adoption of several simplifications during the preparation of the calculation method, should be considered.

5. Summary and final conclusions

Preliminary analyses have positively verified the possibility of developing a rocket dedicated to the payload launch system developed at MUT. According to the analysis, the three-stage rocket configuration can successfully perform payload launch missions to the target orbit. The limited values of the specific impulses generated by the solid-fuel rocket engines considered in the context of the propulsion units create significant difficulties for the two-stage concept. Such a concept, due to the reduced number of propulsion systems, requires the use of more fuel and an increase in the operating time of the engines. An analysis of the available models of small rocket engines has shown that meeting such requirements, even considering the design of new engines, can be considerably difficult with the current state of technology. From this point of view, the three-stage concept seems to be more advantageous. With less restrictive requirements for the engines of the propulsion units of such a rocket, it is possible to use the available solutions, which will significantly reduce its development and deployment time.

The considerations carried out are an introduction to the detailed design of a rocket dedicated to the system concept evaluated at MUT. During the work, difficulties were encountered due to the complexity of the issue of designing a space rocket. For this reason, the analyses presented in this paper were performed assuming a significant degree of simplification. Assuming a constant in time value of thrust and mass flow rate of gases may influence the overestimation of rocket performances and characteristics.

The work is under construction. In further stages, separate attention should be given to issues concerning the detailed analysis of the design of the individual rocket assemblies with respect to strength conditions.

In the next stage of the work, the rocket's mission profile should be simulated based on the developed models to obtain accurate data on the velocity losses and overloads that occur during flight.

It also seems reasonable to analyse the use of liquid fuel engines in rocket propulsion units. Higher values of the specific impulses generated may benefit the rocket's characteristics, especially when considering this type of engine in the context of higher stages.

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