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LATLAUNCH AIR-LAUNCH SYSTEM FOR LOW-COST LAUNCHING OF SMALL SATELLITES INTO LOW EARTH ORBIT

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Abstract. The paper presents the air-launch system enabling the delivery of small satellites into low Earth orbit. One of the most important advantages of the concept is its cost. Generally, the paper proves that launching a carrier from an aerial platform (a movable launch pad) provides the whole range of competitive advantages. In particular, the total losses during the launch from an aerial platform will reduce by 20–35%, and the characteristic velocity of the maneuver will reduce by 4–7%.

Keywords: flight safety, LEO satellites, risk, algorithm, air-launch system.

Introduction

Launching a carrier from an aerial platform is the first step on the way to providing cost-effective delivery of micro-satellites into LEO. The demand for launching spacecraft into the Earth's orbit is constantly growing and becoming market-oriented.

In 1985 only 37 out of 253 launched spacecraft were civilian satellites mainly owned by government institutions (United Nations Office for Outer Space Affairs (UNOOSA) Online Index of Objects Launched into Outer Space), while, according to the data of the Union of Concerned Scientists, 1957 satellites were operated by the end of 2018 (Grego, 2019), and this number increased to 2062 by 31 March 2019 (Union of Concerned Scientists, 2005) (994 satellites by the end of 2012 (Satellite Industry Association, 2017), 1459 satellites in 2016 (Satellite Industry Association, 2017), over 51% of them are privately owned and only 21% are military or dual-purpose satellites (Satellite Industry Association, 2017; Pixalitics Ltd, 2018).

About 70% of newly launched satellites are light vehicles below 1200 kg, which are launched to LEO; and the proportion of this group tends to grow, while the average mass of a spacecraft tends to decrease (Satellite Industry Association, 2017).

It is generally accepted that the cost of a launch is the main limiting factor to the commercial exploration of near-Earth space. The cost of launching a big (approx. 4 tons) satellite into LEO is on average 30 000 to 40 000 EUR per kilo, and the cost of launching a small vehicle weighing up to 100 kilos may exceed 70 000 EUR per kilo (Jones, 2018).

Presently only 9 countries in the world, except ESA member countries, own carriers capable of delivering payload into the Earth's orbit: USA, Russia, Ukraine, Japan, India, PRC, Israel, North Korea and Iran.

1. Project idea

Attempts to create low-cost carriers are being made mainly by private companies in the USA and Japan: RocketLab (Electron) and SpaceX (various modifications of Falcon); IHI Aerospace (Epsilon, SS-520) and Mitsubishi Heavy Industry N-II.

However, in terms of conceptual approaches and main structural solutions, the proposed carriers do not differ fundamentally from those developed in the 1960's–1970's.

The Institute of Aeronautics (AERTI) of Riga Technical University is developing a LatLaunch aerospace system for launching small satellites into LEO.

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The aim of this project is to create a commercial launch system for delivering one kilo of payload into LEO at the lowest possible cost. The research shows that the declared aim can be achieved through the following activities:

1) to discard the first and probably the second stage of the traditional carrier rocket substituting them with reusable aircraft and aircraft rocket stages;
2) to use wing lift to the maximum to overcome the force of gravity;
3) to use atmospheric air to the maximum as an oxidant to reach maximum velocity and altitude;
4) to use aerodynamic control up to the maximum possible altitude to save control engine fuel and to discard the control of the thrust vector and correcting engines of the first stage;
5) to use an aerial platform for the launch, which will allow to carry out the launch over free or desert areas of the world ocean and perform the functions of a tracking station and a command and measurement complex at the moment of launching and during the delivery.

Preliminary mission concept is shown in Figure 1.

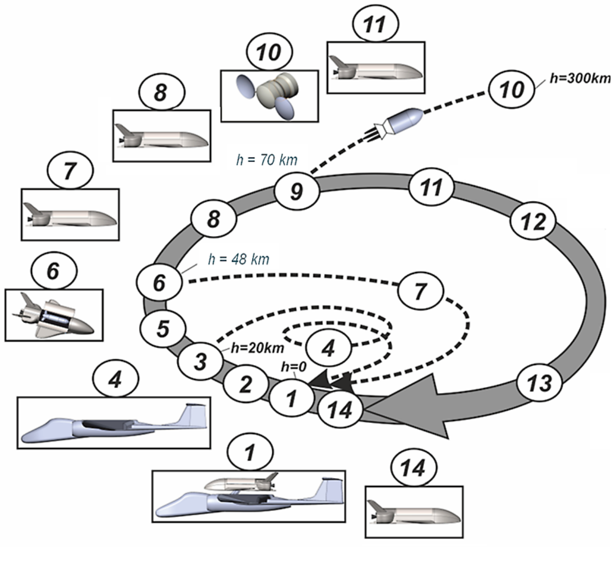

Figure 1. Preliminary LatLaunch launching system mission concept

In Figure 1: 1 – Take-off; 2 – Climb; 3 – Launch of the 2nd stage; 4 – 1st Stage mission control flight and landing; 5 – Supersonic / hypersonic stage(s) acceleration; 6 – Rocket plane stage launching; 7 – Supersonic / hypersonic stage(s) deceleration, descending and landing; 8 – Rocket plane stage acceleration; 9 – Rocket stage (or mini-shuttle) launching; 10 – Satellite separation; 11 – Rocket plane deceleration; 12 – Rocket plane aviation-type flight and descending; 13 – Rocket plane landing; 14 – System repair and start preparation.

2. Scheme of launching a satellite orbiting system from an aerial platform

At the earliest stage of LatLaunch system concept development, it was necessary to correctly make the first decision that would determine the further development of the low-cost launch system design. It was a decision about the place from which the carrier would be launched: from Earth like at standard launch sites, from a sea carrier like in Sea Launch project or from an aerial carrier like in the world's first private launch system called Pegasus (Orbital ATK — Northrop Grumman, USA).

One of the substantial negative factors during the launch from a ground-based launch site is the necessity of evacuating a large area for the whole launch period to avoid victims and destruction in case of possible emergency launch (Gapiński & Stefański, 2014; Stefański et al., 2014). In addition to that, a ground-based launch site (a launch pad) is costly due to the necessity of acquiring a large plot of land for the launch complex itself including a sufficient safety area around it and other above-ground structures (a refuelling complex with a storage facility for propellant components located at a safe distance from other facilities, an assembly and testing facility, a launch control centre, a command and measurement complex, etc.).

The problems related to the evacuation and securing of large areas, acquisition of a large plot of land and ensuring ground infrastructure was solved in Sea Launch project. A ship, which is used an assembly and testing facility, a launch control centre and a command and measurement complex, tows a launch pad created on the basis of a drilling platform and combined with a refuelling complex. They sail to the near-equatorial area of the world ocean, which is free from maritime traffic, and launch the carrier. In 1999 this project allowed to substantially increase the availability of space launches and reduce the costs.

However, is the sea launch the most effective way of launching?

The analysis shows the opposite. Launching a carrier from an aerial platform, i.e. a movable launch pad, has a whole range of competitive advantages.

Firstly, with a sufficient flight radius of the "platform aircraft", the carrier can be safely launched above the desert areas of the world ocean, while all the required assembly and testing infrastructure of the system is located in densely populated areas of Europe – places where the personnel permanently reside and work. This solution will substantially reduce not only the expenses for the construction of above-ground launch structures but also business travel expenses and infrastructure expenses related to the operation of such a system.

The choice of a certain launch area can be made several days before the launch, which excludes long and costly procedures of informing and securing the launch area and helps to reduce the related expenses.

Secondly, the use of an aerial platform is related to a number of technical and economic advantages of such a launch system. It is explained by the factors listed below.

3. Determining the parameters of launching a satellite orbiting system from an aerial platform

To launch a satellite into LEO with a velocity V_{sat}, the carrier must develop a characteristic velocity of orbital manoeuvre ΔV that considerably exceeds V_{sat}; the velocity V_{sat} is the same for all types of launching, it is set through calculations when determining the payload (satellite) target orbit, and in the case of the Earth's orbits the velocity V_{sat} lies within the range between the first cosmic velocity of 7.8 km/s and the second cosmic velocity (escape velocity) of 11.2 km/s:

$$\Delta V = V_{sat} + \Delta V_g + \Delta V_a + \Delta V_c + \Delta V_p \pm \Delta V_{rot,} \tag{1}$$

where: ΔV – vector value of the required launcher velocity; V_{sat} – required orbit velocity of the satellite; ΔV_g – gravity velocity losses; ΔV_a – aerodynamic resistance velocity loses (drag loses); ΔV_c – velocity loses for the transformation of initial speed vector direction to the required orbit velocity vector direction (control velocity loses or steering velocity loses); ΔV_p – velocity losses for the compensation of engine thrust reduction in the atmosphere (engine pressure losses, compensating atmospheric pressure); ΔV_{rot} – projectioin of the Earth's rotation velocity vector.

The value of the modulus of velocity increment vector is a determining value in the calculation of the required propellant mass and necessary stage/system engine characteristics when solving an inverse problem – Tsiolkovsky equation:

$$\frac{M_i}{M_f} = e^{\frac{\Delta V}{V_e}}, \tag{2}$$

where: V_e – working body exhaust velocity, m/s; M_i – initial launching system/stage mass, kg; M_f – final launching system/stage mass, kg.

To compare the values of velocity losses in (1), let us consider Table 1 (Humble et al., 1995) where velocities are presented in m/s.

To determine the most efficient method of launching, let us analyse the functional dependencies of values determining the characteristic velocity ΔV in accordance with (1) for various methods of launching.

To perform the analysis, we will use the integration of the relevant part of the motion equation in terms of time, taking into account that the flight time t_{fin} is the useful motion of the carrier; the useful motion occurs from the point of launch 0 to the point of payload separation (or stage separation if the calculation is performed for a stage) t_{fin}.

The value of velocity losses for compensating the engine thrust reduction in the atmosphere (engine pressure losses, compensating atmospheric pressure), which is not presented in Table 1, is maximum for a launch at sea level altitude. It is possible to determine ΔV_p in a way described below.

An expression for rocket engine thrust at sea level T has the following form:

$$T = - V_e \cdot \frac{dM_i}{dt} + (P_a - P_e) \cdot S_{a,} \tag{3}$$

where: P_a – atmospheric pressure; P_e – pressure of thruster outgoing gases in vacuum conditions at the nozzle end; S_a – cross section area of the nozzle end.

Rocket engines used for the first stages of rocket carriers usually have combustion chamber pressures of about 20 MPa and nozzle expansion ratios (the ratio of the cross-section areas and critical section) within the range of 75 to 80. Thus, the working fluid pressure at the end of the engine nozzle in vacuum is 0.25 to 0.27 MPa, and at sea level it is resisted by an atmospheric pressure of 0.1013 MPa.

Let us determine the value of velocity loses due to counterpressure on the example of Saturn V carrier rocket (NASA, 1995).

The thrust of Rocketdyne F-1 engine of the first stage (with a total of 5 engines of the stage) at sea level is T = 6770 kN, while in vacuum it is T_v = 7776 kN (NASA, 2014). The difference in thrust $\Delta T = T - T_v$ is 1006 kN at sea level.

In accordance with Table 4-1 (NASA, 1995), Saturn V reached a velocity of Mach 1 at an altitude of 7800 m in 65 seconds after launching. Based on the data of the U.S. Standard Atmosphere, 1976, the sound velocity at an altitude of 7800 m is 308.91 m/s, the atmospheric pressure is 366.92 mbar, the ratio between the pressure at this altitude and pressure at sea level is 0.36212. Thus, in 65 seconds the flight velocity was 309 m/s, and the difference in thrust at an altitude of 7800 m $\Delta T_1 = T_1 - T_v$ was 364.3 kN.

The total operating time of the engines of the 1st stage was 160 seconds. Thus, a final velocity of 2388.9 m/s was reached at an altitude of 68400 m where the atmospheric pressure was 0.065465 mbar, the ratio between the pres-

Table 1. The value of losses for various launch systems

Launch vehicle	Orbit: h_p x h_a (km) / inclination (deg)	Vsat LEO	ΔVg	ΔVa	ΔVc	ΔVrot	ΔV
Ariane A-44L	170 x 170 / 70	7802	1576	135	38	–413	9138
Atlas l	149 x 607 / 27.4	7946	1395	110	167	–375	9243
Delta 7925	175 x 319 / 33.9	7842	1150	136	33	–347	8814
Space Shuttle	196 x 278 / 28.5	7794	1222	107	358	–395	9086
Saturn V	176 x 176 / 28.5	7798	1534	40	243	–348	9267
Titan IV/ Centaur	157 x 463 / 28.6	7896	1442	156	65	–352	9207

sure at this altitude and the pressure at sea level equalled to 0.000064609 and the difference in thrust $\Delta T_2 = T_2 - T_v$ was 0.067 kN.

According to NASA, 2014, the propellant mass flow for the engines of the first stage was 13011.3 kg/s. Therefore, with a system launch mass of 2970 t, 845700 kg of propellant were used during 65 seconds of flight, and the mass of the remaining propellant was 2124300 kg. The average mass of the system was 2547000 kg. During 160 seconds of flight, 2081808 kg of propellant were used, and the average mass of the system was 1929096 kg.

Then

$$\Delta V_p = \int_0^t \frac{\Delta T(t)}{M(t)} dt \,, \tag{4}$$

where M(t) is the function of launch system mass change.

It is possible to receive an approximate value of expression (4) through the mean values obtained for 160 seconds of flight: the mean value of thrust difference 503034 kN and the mean value of system mass 1929096 kg. As a result, the design value of velocity losses due to counterpressure will be 41.7 m/s. It can be affirmed that the received value is close enough to the truth because the function of atmospheric pressure value related to altitude is determined by an exponential dependence, while the counterpressure appears at an altitude of 48000 m as 0.1% of the value at sea level. At the same time, as the operation of the stage is coming to an end, the acceleration is increasing reaching a maximum of 38.97 m/s^2 in the 160th second of the flight due to the burnout of propellant reserves – with a value of 10 m/s^2 at the point of launching. An altitude of 48000 m is reached approximately 22 seconds before the end of the flight, when 86% of the flight time have elapsed.

Thus, the total counterpressure losses may reach 30–40 m/s, taking into account that 60–70% of this value are related to altitudes up to 10–12 km from the point of launching. Consequently, when launching from an aerial platform, velocity losses will reduce by 18–28 m/s at an altitude of 10–12 km.

The gravity losses ΔV_g determine the increase of characteristic velocity required for supporting the carrier within the gravitational field (velocity losses due to the effect of gravity).

$$\Delta V_g = \int_0^{t_{fin}} g \sin\theta dt \,, \tag{5}$$

where: g – gravitational acceleration; θ – angle between the launcher trajectory and horizon.

The value of this integral can be determined as follows:

$$\Delta V_g = (g \cdot \sin\theta)_{\mathrm{miv}} \cdot t_{fin}, \tag{6}$$

where: $(g \cdot \sin\theta)_{\mathrm{miv}}$ – mean integral value by flight time.

Thus, gravity losses depend on the value of free fall acceleration, average flight path curvature (more exactly, on the degree of proximity to the horizontal flight and flight time).

Free fall acceleration acting on a launch system mainly depends on the latitude of the launch site and the distance from the Earth's surface. According the GRS80 (Geodetic Reference System, 1980) adopted by XVII General Assembly of the IUGG, Canberra, 1979, the gravitation acceleration can be approximately calculated as:

$$g = 9{,}780327 \cdot (1 + 0{,}0053024 \cdot \sin^2(\varphi) - 0{,}0000058 \cdot \sin^2(2 \cdot \varphi)) - 0{,}000003086 \cdot h, \tag{7}$$

where: φ – geographic latitude; h – height above sea level (m).

So, when carrying out a launch from an aerial platform at an altitude of 10 km under other equal conditions, the reduction of gravity losses will make up 0.315% when compared to a ground start. At an altitude of 12 km it will be 0.379%, which will make up 4.4 m/s to 5.25 m/s if applied to the mean value of this characteristic (1386.5 m/s), presented in Table 1.

Actually, the effect will be substantially higher because during a vertical launch sin $\theta = 1$ and during a horizontal launch sin $\theta = 0$.

In addition to that, it is obvious that with all else being equal, the flight time in case of launching from an aviation platform will decrease. Being at an altitude of 10 – 12000 m, a platform will help to reduce the flight time by the value required for reaching this altitude. Moreover, an aerial platform itself has some initial velocity, which will allow to save the flight time by the value of time required for reaching such a velocity.

The value ΔV_a characterizing aerodynamic losses is conditioned by thrust consumption for overcoming the force caused by atmospheric air flow. During the acceleration of a carrier rocket, the force of air resistance is the most significant value.

The force of air resistance F_a is determined as follows:

$$F_a = ½ \cdot \rho \cdot v^2 \cdot C_x \cdot S, \tag{8}$$

where: ρ – air density; v – carrier velocity relative to the flow of air (the effect of wind can be neglected as the wind speed is too low relative to the carrier velocity; so, this term will be equal to the carrier velocity relative to the Earth); C_x – aerodynamic drag coefficient; S – characteristic section area (the normal carrier section to the carrier velocity vector relative to the air flow).

In this case, the air density can be expressed based on the ideal gas equation:

$$\rho = \frac{P \cdot M}{R \cdot T}, \tag{9}$$

where: P – absolute atmospheric pressure; T – absolute atmospheric temperature; R – ideal gas constant; M – air molar mass.

In compliance with the U.S. Standard Atmosphere, 1976, in the troposphere the temperature T and pressure P change along with the increase of altitude according to the following dependencies:

$$T = T_0 - L \cdot h, \tag{10}$$

where: T_0 – standard temperature at sea level (altitude $h = 0$, $T_0 = 288.15$ K); L – temperature lapse rate =

6.5 K/km for altitudes up to 11 km, then L = 0 K/km up to an altitude of 20 km, after which L = –1 K/km up to an altitude of 32 km (U.S. Standard Atmosphere, 1976); h – altitude above sea level, km.

$$P = P_0 \left(1 - \frac{L \cdot h}{T_0}\right)^{\frac{g \cdot M}{R \cdot L}}, \tag{11}$$

where: P_0 – pressure at sea level; g – acceleration due to gravity at sea level.

Thus:

$$\rho = \frac{M \cdot P_0 \left(1 - \frac{L \cdot h}{T_0}\right)^{\frac{g \cdot M}{R \cdot L}}}{R \cdot \left(T_0 - L \cdot h\right)}. \tag{12}$$

The U.S. Standard Atmosphere (1976), assumes the following values of constants contained in expressions 4–7: P_0 = 101325 Pa, sea level standard pressure; T_0 = 288.15 K, sea level standard temperature (15^0C); g = 9.80665 m/s^2, gravitational constant; L = 6.5 K/km, temperature lapse rate; R = 8.31432 J/mol·K, gas constant; M = 28.9644 g/mol, molecular weight of dry air.

When using the values of the specified constants, expression (12) acquires the following form, which is true up to an altitude of 11 km (the density is expressed in grams per cubic metre):

$$\rho \frac{2.93482 \cdot 10^6 - \left(0.02256 \cdot h\right)^{5.2559}}{2395.7713 - 54.04308 \cdot h}. \tag{12.1}$$

It is obvious that the density decreases as the altitude grows. The value of degree index in the numeral of expression (7) for altitudes from 11 km to 20 km will increase up to 34.163194. Thus, the density drop rate will increase along with the increase of altitude.

Based on the data in the U.S. Standard Atmosphere (1976), the air density at sea level is 1.2250 kg/m^3, while at geometric altitudes of 10 km, 11 km and 12 km the density is 0.41351 kg/m^3, 0.36480 kg/m^3 and 0.31194 kg/m^3 respectively.

Thus, in accordance with equation (8), with all else being equal, the aerodynamic drag force during the launch of a carrier at an altitude of 12 km will decrease by 3.927 times in comparison with the drag force during the launch from the Earth's surface at sea level.

As the aerodynamic drag force F_a is equal to the product of the launch system mass M_l and acceleration gained by the launch system under the action of this force a_a,

$$F_a = M_1 \cdot a_a. \tag{13}$$

Thus:

$$a_a = \frac{\rho \cdot v^2 \cdot C_x \cdot S}{2 \cdot M_1}. \tag{14}$$

The value of velocity losses due to the effect of aerodynamic drag can be obtained by integrating the expression for a_a in terms of the time of action of this force. For the period of time Δt, it is expressed as follows:

$$\Delta V_a = \int_0^{\Delta t} a_a(t) dt. \tag{15}$$

It is obvious that as a result of deriving equation (14) in the form of a time dependence for the integration in (15), there will appear 3 terms that depend on the time of action of the force, i.e. on the flight time. These are velocity, altitude (in the expression for air density) and launch system mass. However, there (12) is a tendency for ΔV_a to decrease along with the increase of the launch altitude (decrease of air density), which can be proved, for example, through comparing expressions (15) for altitudes 0 km and 12 km when $\Delta t \to 0$ (with a small change of altitude).

Then the air density is set as a constant and factored out of the integral:

$$\Delta V_{a,\Delta t \to 0} = \rho \cdot \frac{C_x \cdot S}{2} \int_0^{\Delta t \to 0} \frac{v^2(t)}{M_t(t)} dt. \tag{16}$$

This allows to conclude that with all else being equal the initial value of velocity losses due to aerodynamic drag during the launch from Earth at sea level will be 3.927 times higher than that during the launch from an altitude of 12 km.

Taking into account the data presented in Table 1, it is possible to estimate the decrease of launch system velocity losses due to aerodynamic drag when launching from an aerial platform at an altitude of 10–12 km. A decrease by (90–100) m/s can be expected.

When assessing ΔVc, it is necessary to note that during the launch from the surface of the Earth the initial vector of system velocity is normally directed towards the final vector of spacecraft velocity.

In case of launching from an aerial platform, the initial and final velocity vectors are collinear.

The launch from an aerial platform implies the absence of control losses related to the alteration of the rocket flight path from the vertical plane to the horizontal one.

In addition to that, during the launch from Earth when the latitude of the launch point is fixed, control losses are related to the necessity of carrying out a turn to alter the flight path of the carrier directing it to the orbit plane at a certain inclination.

During the launch from an aerial platform, the flight path can be altered by the carrier aircraft before the launch system is launched, which reduces the energy consumption of the launch system to zero.

The two above mentioned path manoeuvres as well as the expenditures on flight and path stabilization at the initial stage of flight (compensation of wind load, cloud load, local aerodynamic effects of atmospheric density fluctuations) form a great portion of the value of control losses. So, it is possible to conclude that during the air-launch the value ΔVc will decrease by (80–90) % reducing the velocity losses by 30 m/s to 200 m/s.

In addition to the above listed factors that decrease velocity losses during the launch from an aerial platform when compared to the launch from Earth, it is necessary to mention two more factors:

1) firstly, an aerial platform allows to carry out a launch from a region with the minimum acceptable latitude in accordance with the conditions of reaching the satellite flight path in the optimal way, i.e. to use the Earth's rotational velocity to the maximum extent possible;
2) secondly, during the launch from an aerial platform, the initial velocity of the launch system will not equal zero like in case of launching from Earth; it will be equal to the velocity of the aerial platform, i.e. about 230–250 m/s.

Conclusions

The analysis allows to conclude that the scheme of launching a satellite orbiting system from an aerial platform has indisputable advantages over a ground launch or a sea launch.

This can be explained with the fact that, in comparison with a ground launch or a sea launch, during the launch from an aerial platform the characteristic velocity of orbital manoeuvre for delivering a satellite into LEO ΔV decreases by 370 m/s at the lowest estimate, while at the highest estimate this decrease may reach over 580 m/s.

Thus, the total losses during the launch from an aerial platform will decrease by (20–35) %, and the characteristic velocity of the manoeuvre will decrease by (4–7) %.

Most losses fall to the 1st stage, which is the heaviest and the most expensive part of the carrier. Therefore, the above specified numbers can be mostly attributed to the first stage of the carrier, for which the characteristic velocity will decrease by (15–25) % that will make it possible either to start using much cheaper propellants with a lower specific impulse (for example, solid propellants) without detriment to the system mass, or to reduce the propellant reserve and, consequently, the required engine thrust, mass and cost of the launch system.

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