ADVANTAGE OF USING AIRCRAFT-TYPE STAGES FOR CREATING "LATLAUNCH" LAUNCH VEHICLE

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Abstract. The demand for launching satellites into the Earth's orbit is constantly growing and becoming market-oriented. In 1985 only 37 out of 253 launched satellites were civilian, mainly owned by government institutions. Now more 2 700 satellites operate on the Earth's orbit and over 60% of them are privately owned. About 70% of newly launched satellites are light vehicles below 1 200 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. The orbit launch cost is the main brake to the commercial exploration of near-Earth space. The cost of launching of a small satellite weighing up to 100 kg may exceed 100 000 EUR per kg. 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. Attempts to create low-cost carriers are being made 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 1960s-1970s. The Institute of Aeronautics (AERTI) of the Riga Technical University MSTF is developing a LatLaunch aerospace system for launching small satellites into LEO. The project aim is to create a commercial launch system to deliver a payload to LEO at the lowest possible cost per kilogram. The research shows that the declared aim can be achieved through the discard of the first and probably the second stage of the conventional launch vehicle substituting them with reusable aircraft and aircraft-rocket stages. The study analyses the existing launch systems and proposes some new conceptual principles for development of them.

Keywords: launch, rocket, spacecraft.

1. Analysis of existing and prospective launch system technical and economic characteristics

Approaches to the low-budget system for launching microsatellites into low-earth orbit (LEO) LatLaunch (LatLaunch) concept formulation should base on the idea of achieving competitive advantages. The main one is the low cost of payload per kilogram launching [1].

The obvious way to reduce the launching cost is to create a truly reusable launch system. A reusable launch system (RLS or reusable launch vehicle, RLV) is a launch system that is capable of launching a payload into space more than once, as opposed to disposable ones where each launch vehicle (LV) is launched once and destructs in mission. The cost of RLV is not amortized for one flight, which makes it possible to reduce the launch cost [1].

Many RLS are currently under development. A fully reusable LVs do not currently exist. The ATV Space Shuttle came closest to this edge, but the cost and timing of preparing the used spacecraft for the next flight were comparable to the cost and timing of manufacturing a new LV. Below is a list of major projects underway to create at least partially RLV [2]:

1. Blue Origin New Shepard, USA;
2. SpaceX Falcon Reusable Grasshoppers, USA;
3. Armadillo Aerospace, Stig, USA;
4. Northrop Grumman Hybrid Launch Vehicle, USA;
5. McDonnell Douglas DC-X, USA;
6. XCOR Aerospace EZ-Rocket, USA;
7. ARCA Space Corporation Orizont, USA;
8. Masten Space Systems XA, USA;
9. The Spaceship Company Space Ship 2 Sat Launcher, USA;
10. Rocketplane Global, Inc XP, USA;
11. Scaled Composites Tier 1, USA;
12. Aerojet Rocketdyne RLS, USA;
13. Boeing X37 project, USA;
14. Orbital Sciences Prometeus, USA;
15. Sierra Nevada Corporation Dream Chaser, USA;

16. RLV-TD ISRO, India;
17. Airbus Defence and Space ADELINE, EU;
18. Skylon Reaction Engines Ltd, UK.

However, the above RLV projects (except for Falcon and partly X37) have not been implemented and are in the development stage, so the main competitors of LatLaunch in the next years will remain the existing disposable LVs, as well as the developing Falcon family.

The main providers of launch services are companies, which purchased exclusive rights to launch an entire LV. The data on the contract value for the launch of a particular spacecraft are confidential, therefore, the cost data available in the public domain contain significant contradictions. When analyzing the materials, it was guided either by close data given repeatedly without cross-references or by public commercial offers [3-15].

If it is necessary to launch a microsatellite without waiting for an accompanying payload, the most attractive is to launch it by Shtil-2 LV. In this case, with an average launch cost of 4-5 million USD, it will be able to put 70 kg into LEO, and the cost will be 54 100-71 400 USD per kg [3; 4].

All other LVs require a combined load, as they have a load of thousands kg, e.g., Dnepr (3 700 kg), which is usually put to LEO simultaneously about 30-40 satellites. That determines a long waiting period for an associated payload with similar orbit parameters. Dnepr's cost of launching 1 kg to LEO is 9-12 thousand USD [5; 6].

The Soyuz LV (Russia and ESA) delivers 7 020 kg of a payload to LEO at the price of 20-30 thousand USD per kg [8].

China has several commercially available LVs: CZ-2C (Long March 2C) with 3 850 kg LEO load and price 7 800 USD per kg [3, 7]; CZ-3BE with 12 000 kg and 5 800 USD per kg, CZ-4C with 4 200 kg, and 7 100 USD per kg [3, 5] and CZ-11 with 530 kg to LEO and kg price of 10 000 USD [3].

India also serves the global launch market. LV LVM3 can be ordered for 60 million USD and move 8 000 kg to LEO, so its price is 7 500 USD per kg. GSLV launches 5 000 kg and has a value of 47 million USD, which means 9 400 USD per kg, while PSLV can orbit 3 250 kg for 21 million USD and has a kg launching price of 6 500 USD [3].

ESA is represented by VEGA and Ariane LVs. VEGA can put to LEO 1 500 kg [7, 10] or 1 963 kg [3], the mission cost is 42 million [10] to 37 million USD [3]. Kg orbiting price is from 18.8 [3] to 23 [6] - 28 [10] thousand USD. In 2021, it is planned to launch the upgraded Vega-C, delivering to LEO 2 300 kg, which will reduce the price of the payload kg to 15 000 USD [11].

Ariane-5 with various upper stages moves 16-22 000 kg to LEO, with the cost of 160-220 million USD [12]. According to [3], the cost of a 20 000 kg mission is 178 million USD with the kg launching price of 8 900 USD. In 2022 Ariane-6 flight is planned. With the same load, its cost should be reduced to 94-117 million USD, which leads to a decrease in the kg launching price to 4 700-5 900 USD [3].

The leading provider of launch services currently is the USA. Active and promising LVs with a cost indicator of less than 20 000 USD per kg to LEO transportation, are represented by Antares, which have capability 6 600 kg to LEO and launch price 80-85 million USD (12 900 USD per kg), and Atlas-V, which can orbit a payload of 18 814 kg, while the cost of kg launching is 9 500 USD [3].

From the operating US LVs, the most attention deserves the Falcon 9, which can launch to LEO 22 800 kg and has 62 million USD price, and the Falcon Heavy, which can launch 63 800 kg for 90 million USD [13], which determines kg to LEO price, respectively, at USD 2 720 and 1410.

At the same time, it should be noted that it is of an advertising nature and is not technical documentation that defines the obligations to the flight customer [13]. If check the "Falcon users guide" [14], Section 3.3 "Mass Properties" in Fig. 3-2, it follows that the maximum payload should not exceed 11 000 kg with a maximum height of 2.8 m. Moreover, [14] contains a strict limitation: "Payloads should comply with the mass properties limitations given in Figure 3-2. Payloads can be accommodated as a mission unique service. Payload mass properties should be assessed for all items forward of the payload..., including any mission-unique payload adapters and separation systems. Mass property capabilities may be further constrained by mission-unique payload adapters, dispensers or separation systems", it does not mention any additional load capabilities, including for Falcon Heavy. Then, if someone is guided by the restrictions of [14], then Falcon 9 price for kg to LEO transport should be

5 636.4 USD, which seems much more realistic. Moreover, if the customer's satellite weighs not 11 tons but less, it will need to use the services of the operator, while the price per kg to LEO transfer will increase from 9 750 USD per kg for 100 kg satellite to 29 500 USD per kg for 10 kg CubeSat [15].

Table 1 shows summary data on the main technical and economic indicators of the launch of a payload into LEO by the main LVs.

Table 1

Comparative technical and economic indicators of existing launch systems

Launch vehicle, origin	Payload mass delivered to LEO, kg	Payload kg cost to LEO in USD when paying for the entire LV load
Shtil 2 (Russia)	70	54 100
Sojuz (ESA, Russia)	7 020	20 000
Dnepr (Ukrain)	3 700	10 000
CZ-3BE (PRC)	12 000	5 800
PSLV (India)	3 250	6 500
VEGA-C (ESA)	2 300	15 000
Ariane-6 (ESA)	20 000	4 700
Atlas-V (USA)	18 814	9 500
Falcon 9 [13]	22 800	2 720
Falcon 9 [14]	11 000	5 636
Falcon 9 - 300 kg payload [15]	300	6 167
Falcon 9 - 10 kg payload [15]	10	29 500

Obviously, when ordering a small load, the price per kilogram orbiting to LEO significantly increases and reaches 20 000 USD or more, as a result of which the launch cost approaches the same for non-budget systems.

Thus, when approaching the development of the concept of the LatLaunch LV, it is necessary to be clearly guided by the idea of achieving the main competitive advantage – reducing the cost of launching per 1 kg of payload into LEO to the level of the competitors' costs and lower when launching loads of hundreds of kilograms into orbit.

It should be guided by the target indicator of realistic Falcon 9 pricing, i.e. 5 500 USD per kg, with a payload mass of 1 000 kg, then the LatLaunch mission price should not exceed 5.5 million USD, which is 11 times less than the price for the Falcon 9 flight.

A launch system, capable of satisfying such a requirement, must have a completely different technical concept than the existing ones.

2. Approaches to formation of the LatLaunch concept

The SciLab-6.1.0 mathematical model was developed to define proper approaches to the LV LatLaunch concept. Mass and power characteristics of budget launchers were analyzed. Based on the manufacturers' data, the characteristic velocity at the end of the operation of the corresponding stage was calculated by the Tsiolkovsky equation:

$$\Delta V = -v_{ex} \ln \frac{m_s}{m_{wf} + m_s},$$
(1)

where ΔV – characteristic velocity, m·s^{-1};
v_{ex} – working fluid exhaust velocity, m·s^{-1};
m_s – rest mass of the spacecraft stage (structure mass) with payload, kg;
m_{wf} – used working fluid mass, kg.

Based on the reference data from the ratio:

$$T = -v_{ex}G + S(P_{ex} - P_{atm}),$$
(2)

where G – working fluid mass flow rate, kg·s^{-1};
S – stage engines cross-section, m^2;

P_{ex} – working fluid exhaust pressure, $kg \cdot m^{-2}$;

P_{atm} – atmosphere pressure, $kg \cdot m^{-2}$, according to the reference data of the stage power plants, the data missing for the solution of equation (1) were determined.

Further, proceeding from the flight characteristics, a numerical method was used to solve the inverse Mieszczerski equation to determine the heading velocity of the stage:

$$\vec{V}(t) = \frac{1}{m_{wf} + m_s} \int \sum_{i=1}^{n} \|\vec{F}_i\| dt - \vec{v}_{ex} \int \frac{1}{m_{wf} + m_s} dm_{wf} \,, \tag{3}$$

where $\vec{V}(t)$ – launcher stage velocity vector, $m \cdot s^{-1}$;

t – stage operation time, s;

$\vec{F}_i$ – forces, acting on the launcher stage, N.

Also, to determine the flight coordinate along the corresponding stage trajectory (to calculate the flight distance), the solution of the equation of motion for variable mass body was carried out by integrating equation (3) over the operating time of the stage engine:

$$x(t) = \frac{1}{m_{wf} + m_s} \left[\int\int \sum_{i=1}^{n} \|\vec{F}_i\| dt dt \right] - \vec{v}_{ex} \left(\int\int \frac{1}{m_{wf} + m_s} dm_{wf} dt \right), \tag{4}$$

where $x(t)$ – launcher stage movement coordinate along the flight path, m.

Taking into account that in many cases the final speed of the stage is known, when the engine stops, the time of operation of the stage also stops, i.e. stage initial (ignition, start) and shutdown (cutoff) velocity alongside ignition and shutdown time, as well as the corresponding trajectory parameters (for example, flight altitude, stage movement in longitude and latitude), stage characteristics were determined by solving the equation of motion for the flight segment, obtained on the basis of equation (4):

$$x(t_1) - x(t_0) = \int_{t_0}^{t_1}\int_{t_0}^{t_1} \frac{1}{m_{wf} + m_s} \left[\sum_{i=1}^{n} \|\vec{F}_i\| \right] dt dt - \vec{v}_{ex} \int_{0}^{m_{wb}}\int_{t_0}^{t_1} \frac{1}{m_{wf} + m_s} dm_{wf} dt \,, \tag{5}$$

where $x(t_0)$ – launcher stage engine start position on flight path, m;

$x(t_1)$ – launcher stage engine shutdown position on flight path, m;

t_0 – stage engine start time, s;

t_1 – stage engine cutoff time, s.

While modelling, the hypotheses about the uniform mass flow rate of the working fluid by the engine of the stage and the use of the entire fuel supply of the stage were used. When preparing the initial data for the calculations, the main sources were used: [14], [16-21] and other sources. The results of modelling the stage characteristics of the main budget launchers are presented in Table 2.

Table 2

Results of the budget launcher characteristics modelling

Launcher	Sojuz	Dnepr	CZ-3BE	PSLV	Ariane 6	Atlas V	Falcon9 22.8t	Falcon9 11t
LEO max load LV mass, t	314.45	208.9	469.33	320	860	587.8	549	532
Mass factor (load to LV)	0.0223	0.0096	0.0245	0.0102	0.0252	0.0320	0.0415	0.0207
1 stage mass, t	177.69	161.52	180.9	89	618.4	233.5	421.3	421.3
1 stage propellant mass, t	156.24	147.9	164.4	73.2	574.4	213.2	395.7	395.7
1 stage thrust, kN	3354	4160	2961.6	4221	18600	6754	7605	7605
1 stage Vex, $m \cdot s^{-1}$	2569	2873	2556.2	2569	2731	2739	2765	2765
1 stage burning time, s	117	130	140.1	70	132.8	88.3	145	145
1 stage separation time, s	118	135	141.6	75	134	118	148	148
1 stage final ΔV, $m \cdot s^{-1}$	1951	3537	1102	667	3010	1233	3527	3769
1 stage final altitude, km	50	51	56	26	51	44	84	84
stage/launcher mass ratio	0.565	0.773	0.386	0.278	0.719	0.397	0.767	0.792

Table 2 (continued)

Launcher	Sojuz	Dnepr	CZ-3BE	PSLV	Ariane 6	Atlas V	Falcon9 22.8t	Falcon9 11t
stage/launcher ΔV ratio	0.203	0.372	0.115	0.069	0.313	0.130	0.433	0.362
2 stage mass, t	99.8	41.1	197.4	166.5	168.9	305.4	96.6	96.6
2 stage propellant mass, t	90.1	36.47	186.2	138.2	140	284.1	92.67	92.67
2 stage thrust, kN	792.5	775	2961.6	4800	1370	3827	981	981
2 stage Vex, m·s^{-1}	2500	3128	2556	2324	4246	3113	3412	3412
2 stage burning time, s	286	190	157.5	110	600	252	374	374
2 stage separation time, s	287.6	327	159	187	737	257	533	533
2 stage final ΔV, m·s^{-1}	3226	4594	2653	2119	3678	5165	4626	6634
2 stage final altitude, km	170	120	71	67	150	164	168	168
stage/launcher mass ratio	0.317	0.197	0.421	0.520	0.196	0.520	0.176	0.182
stage/launcher ΔV ratio	0.335	0.484	0.278	0.220	0.382	0.545	0.567	0.638
3 stage mass, t	27.76	3.07	55.96	48.72	43.76	23.29	-	-
3 stage propellant mass, t	25.4	1.91	49.4	42	31	20.83	-	-
3 stage thrust, kN	297.9	20.2	742	799	180	198.4	-	-
3 stage Vex, m·s^{-1}	3520	3030	2922	2873	4560	4505	-	-
3 stage burning time, s	270	1000	180.4	133	770	772	-	-
3 stage final ΔV, m·s^{-1}	4444	1365	2357	3064	2929	3075	-	-
3 stage final altitude, km	-	-	169.2	218	-	-	-	-
stage/launcher mass ratio	0.088	0.015	0.119	0.152	0.051	0.040	0.000	0.000
stage/launcher ΔV ratio	0.462	0.144	0.247	0.318	0.305	0.325	0.000	0.000
4 stage final ΔV, m·s^{-1}	-	-	3440	2183	-	-	-	-
5 stage final ΔV, m·s^{-1}	-	-	-	1600	-	-	-	-
payload to LEO	7000	2000	11500	3250	21650	18814	22800	11000
total ΔV, m·s^{-1}	9621	9496	9552	9634	9618	9474	8154	10404
LEO (ISS) need V, m·s^{-1}	7750	7750	7750	7550	7550	7550	7550	7550
ΔV loss, m·s^{-1}	1871	1746	1803	2084	2068	1924	604	2854*

* - excess is explained by the possible margin for braking for the 1st stage return.

3. Results of modeling observation and formulation of the principles of the LV concept

The final formulation of the principles for creating the LV LatLunch occurred as a result of the analysis of modeling data based on the concept of the characteristic velocity of the orbital maneuver ΔV. The essence of the ΔV concept is that for launching a satellite into orbit with a velocity V_{sat}, which is the same for all launch methods, it is set through calculations when determining the satellite target orbit, and in the case of the Earth's orbit it lies between the minimal orbiting velocity of 7.8 km·s^{-1} and the escape velocity of 11.2 km·s^{-1}:

$$\Delta V = V_{sat} + \Delta V_g + \Delta V_d + \Delta V_c + \Delta V_p \pm \Delta V_{rot}, \qquad (6)$$

where ΔV – vector value of the required launcher velocity, m·s^{-1};
  V_{sat} – required orbit velocity, m·s^{-1};
  ΔV_g – gravity velocity losses, m·s^{-1};
  ΔV_d – aerodynamic resistance velocity loses (drag loses), m·s^{-1};
  ΔV_c – velocity loses for transformation of the initial speed vector direction to the required orbit velocity vector direction (control velocity loses or steering velocity loses), m·s^{-1};
  ΔV_p – velocity losses for compensation of the engine thrust reduction in the atmosphere (engine pressure losses, compensating atmospheric pressure), m·s^{-1};
  ΔV_{rot} – projection of the Earth's rotation velocity vector.

Table 2 shows that the total ΔV loss is about 20% of the ΔV, which is developed by LV engines. Reducing losses will reduce the required mass of fuel, engine thrust, structure material consumption and the cost of the LV. Let us consider the constituent elements of losses.

The velocity loss component entering into equation (6) - the gravity losses ΔV_g described as:

$$\Delta V_g = \int_0^{t_{fin}} g \sin \theta dt \tag{7}$$

where g – gravitational acceleration, m·s^{-2};
       θ – angle between the launcher trajectory and horizon, degree.

Entering in (7) acceleration of gravity, acting on the LV, mainly depends on the latitude of the launch site and the distance from the Earth's surface. According to [22], the gravitation acceleration can be approximately calculated as:

$$g = 9.780327(1 + 5.3024 \cdot 10^{-3} \sin^2(\varphi) - 5.8 \cdot 10^{-6} \sin^2(2\varphi)) - 3.086 \cdot 10^{-6} h, \tag{8}$$

where φ – geographic latitude, degree;
       h – height above sea level, m.

Consequently, the closer to the equator and higher the starting point is the fewer gravitational losses, and with a horizontal start, they are reduced to zero.

Entering into equation (6) losses for the aerodynamic drag ΔV_d are the integral over the flight time of the ratio of the aerodynamic drag force F_d to the vehicle mass:

$$\Delta V_d = \int_{t_0}^{t_1} \frac{F_d}{m_{wf} + m_s} dt . \tag{9}$$

The aerodynamic drag force F_d, from equation (9) is determined as follows:

$$F_d = 0.5 \cdot \rho \cdot v^2 \cdot C_x \cdot S, \tag{10}$$

where ρ – air density, kg·m^{-3};
       v – LV velocity relative to the flow of air, m·s^{-1};
       C_x – aerodynamic drag coefficient;
       S – characteristic area of the LV, its cross-section to the airflow velocity vector, m^2.

Air density is significantly dependent on the altitude. According to [23], the standard air density at sea level is 1.2250 kg·m^{-3}, while at geometric altitudes of 10, 25 and 50 km the density is respectively 0.41351, 0.039466 and 0.001026 kg·m^{-3}. Therefore, all other things being equal, in accordance with (9) and (10), when the LV is launched from a platform at an altitude of 10, 25 and 50 km, the aerodynamic losses will be 3, 30 and 1200 times less, respectively.

The same applies to engine backpressure losses ΔV_p, which fall with altitude, while the engine thrust and the specific impulse increase.

The more the control losses ΔV_c decrease, the more the trajectory at the start coincides with the final flight trajectory, and also the smaller disturbance (wind, precipitation, uneven air density) has to be compensated for by the thrust of attitude control vernier engines, that consume the fuel supply and "eat up" the LV mass.

Therefore, the minimum control losses will be at the launch of the carrier in the orbital plane at the maximum altitude and with the minimum angle of attack.

When analyzing the data obtained, it was found that the greatest losses relate to the first stages of the carrier, with 90% of the losses occurring during the flight to the Kármán line (altitude < 100 km).

Conclusions

The conducted research allows to formulate 3 conceptual principles, guided by which, when designing the LatLaunch carrier, one can achieve a significant reduction in the cost of launch:
1. Reducing the LV mass and cost by reducing the mass of the required fuel by using an aircraft platform for launching and launching the LV from a course coinciding with the plane of the mission's orbit from an altitude of about 10 - 15 km, from an area as close to the equator as possible (using the linear component of the Earth's rotation velocity);
2. Reducing the LV mass and cost due to the use of atmospheric air as an oxidizer for 1 and possibly 2 stages to overcome the main aerodynamic and gravity drag, and accelerate to a velocity of 2-

3 km·s^{-1}, the use of wing lift to climb to an altitude of about 45-50 km, and aerodynamic control to high altitudes;

3. The use of fully reusable winged jet-powered supersonic and hypersonic aircraft-type stages and the rocket engine-powered hypersonic aircraft-type stage for further launch of a non-recoverable rocket booster with a payload.

With a decrease in the mass of the propellant required for the flight, this cost is not simply "deducted" from the cost of the mission, but the LV weight decreases, and, consequently, the cost of its structures. The principle of multiple uses of most of the mission equipment allows depreciating its cost in several flights, which will lead to a significant launching price reduction.

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