The Concept and Theoretical Performance of Vertical Rocket Launcher Aircraft

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Abstract: A concept of a new type of military aircraft called rocket launcher aircraft (RLA) is presented. RLA is effectively a reusable first stage of a two-stage military rocket. The second stage called drone launched short range rocket (DLSRR) is disposable. DLSRR is discussed in author's previous work and the present work can be viewed as a sequel to afore article. The function of an RLA is to raise one or more DLSRR to an altitude of up to 100 km and to supply them with initial velocity of up to 2,355 m/s. The DLSRRs are fired at high initial velocity and altitude. This enables them to reach targets at the distances of hundreds of kilometers at much lower cost than conventional short range rockets. RLA returns to the base within 6.5 to 8 minutes of its launch. It should be able to perform two to four sorties per hour and 30 to 50 sorties per day. RLA has one or more primary rocket engines for liftoff and acceleration. Most RLAs have one or more auxiliary propeller engines for landing. Some RLAs use their primary rocket engines for landing. Light RLAs may be able to land using only a parachute. A great variety of RLAs with liftoff mass ranging from 1 ton to over 1,000 tons is possible. Some RLAs have disposable fuel tanks. In this work we calculate performance of a light (10 tons), medium (55 tons) and heavy (390 tons) RLAs. Every RLA should be capable of both vertical and forward-leaning trajectories. In this work, we focus on vertical ones. We hope that, Rocket Launcher Aircraft hold a great promise for the future.

Keywords: Military Aircraft, Reusable First Stage, Short Range Ballistic Missiles, Rocket Artillery

1. Introduction

In this work we discuss a concept of a new class of military aircraft. Rocket Launcher Aircraft (RLA) is a rocket-powered aircraft which raises drone launched short range rockets (DLSRR) to a high altitude and fires them with high initial velocity. Then RLA returns to its base ready for refuelling and another sortie. Effectively, an RLA is a reusable first stage of a military rocket. DLSRR are discussed in the authors article [1] and the present work can be viewed as a sequel to [1].

The trajectory of any RLA can either be vertical or inclined forward. In this work we only consider vertical trajectories for RLA. Vertical trajectories enable faster return to base and thus faster reuse. Simpler RLAs should be able to fly 4 sorties per hour. The three RLAs we are considering in this work have DLSRR release altitude of 40 km to 100 km. They have DLSRR release velocity of 1,200 m/s to 2,355 m/s.

DLSRRs are fired at relatively high initial altitude and with relatively high initial velocity. For RLAs moving on a trajectory inclined forward, DLSRR's engines can apply thrust in the same direction in which a DLSRR is moving. For RLA on vertical trajectory, DLSRR starts out with an initial vertical velocity. The thrust of DLSRR's engines must be inclined forward.

DLSRRs can deliver non-nuclear warheads to a target at much lower cost than conventional short range ballistic rockets. First, DLSRR's engine must provide lower $\triangle v$ than a conventional ballistic rocket in order to reach a target at the same range. This is due to DLSRR's head start both in terms of initial altitude and initial velocity. Second, as DLSRR flies through rarified air, it experiences less heating and needs less protection than a conventional rocket.

It may be noticed that conventional fighters and bombers also deliver munitions to a relatively long distance at a relatively low cost. Unlike RLA, conventional fighters and

bombers expose themselves to antiaircraft fire, which may make their operations inefficient. RLA never flies into hostile territory.

In Section 1, we describe the state of the art rockets. In Subsection 1.1, we describe rocket systems capable of making precise strikes to the range of 80 km to 600 km also mentioning some longer range missiles. For ranges of 156 km to 600 km, the costs are from $2,700 per kg to $6,700 per kg depending on the missile. Thus we establish the need for a less expensive system. In Subsection 1.2, we describe space rockets with reusable first stage.

In Section 2, we introduce the physics of vertical rocket flight. We present the equations describing the rocket trajectory. We also introduce the concept of Specific Impulse – an important concept for rocket development. The material in Section 2 is used to write programs and perform calculations with results presented in subsequent sections.

In Section 3, we describe engines used by RLA. In Subsection 3.1, we describe solid propellant rockets. We discuss propellant composition, fuel grain shape, and propellant grain cost. In Subsection 3.2, we describe liquid propellant rocket engines. We describe liquid propellants and their cost. In Subsection 3.3, we describe auxiliary rocketprop engines used during landing.

In Section 4, we describe Rocket Launcher Aircraft. In Subsection 4.1, we present the variety of parameters for different RLAs and their trajectories. In Subsection 4.2, we present three RLAs we are considering in this work – Light, Medium, and Heavy. In Subsection 4.3, we present time lines and performances for vertical sorties by the three aforementioned RLAs. In Subsection 4.3, we summarize our results. In Section 5, we present Conclusions and discuss further directions for research.

2. State of the Art Rockets

2.1. Artillery Rockets

For reader's convenience, we summarize information on some characteristics of artillery rockets. Short range ballistic missiles can make precise strikes at ranges from 100 km to 1,000 km. Some of the modern systems are extremely expensive. Cost per 1 kg of munitions delivered to the target varies between $1,500 and $13,300 depending on the range. The measure of expense for a missile system at a given range is the cost per unit delivered weight. More detailed information on the costs for solid propellant rockets (such as GMLRS, ATACMS, Trident (see [2, 3, 4, 5])) can be found in Table 1 of author's work [1].

Liquid propellant rockets are less expensive, but they require about an hour to be prepared for firing. The information on the costs of liquid propellant rockets [6] (such as Dong-Feng 15 [7], Scud C [8], Agni II [9]) can be found in Table 2 of author's work [1]. As can be seen from the aforementioned table, the cost of delivering munitions by short range ballistic missiles is very high. The costs of long range missiles are relatively

low for their range. For example, delivering 1 kg of munitions to a range of 7,840 km by Trident II missile is only twice as expensive as delivering 1 kg of munitions to a range of 300 km by ATACMS missile.

2.2. Rockets With Reusable First Stage

From the first space launches of the late 1950s until now, launching payload into orbit has been very expensive. The primary cost comes from the fact that until recently, no launch vehicle has been reusable. Propellant and oxidizer make up under 1% of space launch cost [10]. There have been many projects of reusable spaceships dating back at least to 1960s, but none of them have been successful [11]. On December 21, 2015, Space X made a huge step in history, when the first stage of Falcon 9 spacecraft returned to the launching pad [11].

Currently Space X can deliver payload to Low Earth Orbit at $4,530 per kg, which is much less expensive than the cost that can be suggested by other companies. Space X plans to reduce that price to $3,200 per kg in the near future [11]. One Falcon 9 launch costs $62 million, about $300,000 of which is the cost of fuel [12]. Several other companies and national agencies have plans for producing their own space vehicles with a reusable first stage [13].

Reusability of space vehicles is still in its infancy. Full reusability would bring down the cost of space transportation by a great margin, which is still unknown. Reusable spacecraft would issue a dawn of Space Colonization and the beginning of the true Space Age. In this work we discuss the application of rocketry with reusable first stage to military technology. This technology holds great potential for both military and civilian applications.

3. The Physics of Vertical Rocket Flight

This section describes the equations of motion of a vertically flying rocket. Equations for rocket's altitude and vertical velocity are derived. The equations presented in this section are instrumental for design of MatLab programs that are used to obtain the results of the subsequent sections. In the derivation of the results below we can use one-dimensional model, which is a simplified version of the two-dimensional model used in our work on DLSRR [1].
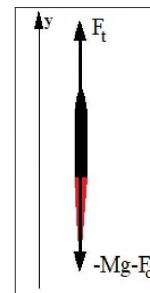

Figure 1. *Forces acting on a firing rocket.*

Three forces acting on a rocket during the powered part of a vertical ascent are presented in Figure 1. The gravitational force is pointed down, and its magnitude is $M(t)g$. The air resistance or drag force F_d is also pointed down during the ascent. Its magnitude is

$$F_d = C_d(\mathcal{M}) \ \frac{\rho v^2 A}{2} = C_d(\mathcal{M}) \ \mathcal{M}^2 \ \frac{\rho v_s^2 A}{2}, \qquad (1)$$

where $C_d(\mathcal{M})$ is the Mach–number dependent drag coefficient, v is the velocity, v_s is the speed of sound, $\mathcal{M}$ is the Mach number, i.e. $\mathcal{M} = v/v_s$, ρ is the air density, and A is the base area of the rocket. Let $M(t)$ be the rocket mass at time t. Then thrust force denoted by $F_t(t)$ acting up is

$$F_t(t) = \dot{M}(t)v_e, \qquad (2)$$

where overdot is the notation for time-derivative and $\dot{M}(t)$ is the fuel burning rate. For an RLA using pure rocket propulsion, v_e is the exhaust velocity. For an air-breathing RLA, v_e is the *effective exhaust velocity*. Generally, the effective exhaust velocity for air-breathing engines is 8,000 m/s to 45,000 m/s. These effective velocities are not attained by any gas in these engines. Their actual jets have velocity of 500 m/s – 700 m/s.

For a pure rocket propulsion, v_e slightly increases as the rocket rises out of dense atmospheric layers. For some rocket engines, the fuel burning rate changes during the burn time. In our case we assume an almost steady burning rate

$$\dot{M}(t) = \frac{M f_p}{t_b} \qquad \text{for time } 0 \le t \le t_b, \qquad (3)$$

where M is the mass of the rocket with propellant, t_b is the *burn time* and f_p is the *propellant mass fraction* given by

$$f_p = \frac{\text{Propellant mass}}{\text{Combined mass of propellant, rocket, and payload}}. \qquad (4)$$

Since we know the forces acting on the rocket, we calculate its trajectory via the MatLab programs *VFirstStage.m* and *VRetroFire.m* , which are described at the end of this section.

The change in velocity produced by the rocket engine is

$$v_r = \int_0^{t_b} \frac{F_t(t)}{M(t)} \, dt. \qquad (5)$$

According to *Tsiolkowsky Rocket Equation* [14], the following relation holds:

$$v_r = -\overline{v}_e \, ln\big(1 - f_p\big), \qquad (6)$$

where $\overline{v}_e$ is the average exhaust velocity. In our case, (of the reusable rocket) $\overline{v}_e \approx 2,300$ m/s. For non-reusable rockets with flame temperatures in excess of 2,800 ^oC, $\overline{v}_e \approx 2,600$ m/s.

The drag loss is

$$v_d = \int_0^{t_0} \frac{F_d(t)}{M(t)} \, dt, \qquad (7)$$

where M is the projectile mass, t_0 is the time it takes the rocket to reach the culmination altitude, and $F_d(t)$ is the drag force at the time moment t.

At this point we define the effective loss of rocket velocity due to gravity. It is called *gravitational loss* and denoted v_g. We define this loss in terms of the rocket's culmination altitude. First, assume that the rocket is given it's impulse v_r instantaneously, and the drag loss is null. Such assumption is an abstract limit for a rocket, but it may be reality for a projectile fired at high altitude. Further assume that the rocket is launched from rest and zero altitude – which is the case for RLAs. Then the total kinetic and potential energy per unit rocket (projectile) mass at the beginning of trajectory is

$$\mathcal{E} = \frac{v_r^2}{2}, \qquad (8)$$

where v_r is the velocity gain due to the action of the rocket engine.

Second, we incorporate the aerodynamic drag loss into (8) to obtain

$$\mathcal{E} = \frac{\big(v_r - v_d\big)^2}{2}, \qquad (9)$$

where v_d is the drag loss. Third, we incorporate the gravitational loss into (9) to obtain

$$\mathcal{E} = \frac{\big(v_r - v_d - v_g\big)^2}{2}, \qquad (10)$$

where v_g is the gravitational loss. In the above expressions, v_r is known and v_d has been calculated in (7), while both $\mathcal{E}$ and v_g are unknown. In order to calculate v_g, we calculate $\mathcal{E}$ using the rocket's culmination altitude h_A, which is calculated for every case of simulation we present. The potential energy per unit rocket mass at the culmination altitude is

$$\mathcal{E} = gh_A, \qquad (11)$$

which is the same as the rocket energy earlier in the path given in (10). Equating (10) and (11), we obtain

$$v_g = v_r - v_d - \sqrt{2gh_A}. \qquad (12)$$

Below We Present a List of Programs Used in This Work
VFirstStage.m

The program *VFirstStage.m* calculates the motion of RLA along the vertical line. The user inputs RLA's mass, diameter, drag coefficient, propellant mass fraction, sea level exhaust velocity, vacuum exhaust velocity, and propellant burning time. Effective frontal area of the RLA is

$$A = \frac{\pi}{4} \, d^2. \qquad (13)$$

While the RLA engine is burning the thrust is obtained by combining (2) and (3):

$$F_t = \frac{M f_p}{t_b} \, v_e, \qquad (14)$$

where M is the RLA mass, f_p is the propellant mass fraction, t_b is the engine burning time, and v_e is the exhaust velocity. After the RLA's engine burns out, $F_t = 0$. Using this data, *VFirstStage.m* starts to perform iterative calculations on the RLA.

At time $t = 0$, *VFirstStage.m* has the following: RLA's mass M; RLA's altitude y; RLA's upward velocity **v**. *VFirstStage.m* uses the RLA's altitude to determine air density ρ and sonic velocity v_s at time $t = 0$. For air density as a function of altitude, we use US Standard Atmosphere 1976. *VFirstStage.m* calculates the Mach number as $\mathcal{M} = v/v_s$. For a given Mach number and for a given type of rocket, one can calculate the corresponding drag coefficient $C_d(\mathcal{M})$. The formulas are quite complicated. The drag force is given by (1):

$$F_d = C_d(\mathcal{M}) \ \frac{\rho v^2 A}{2}. \tag{15}$$

The drag force acts in the direction opposite of RLA's motion. RLA's upward acceleration is derived from all forces acting on the RLA as shown in Figure 1:

$$a = \frac{F_t - S_v \, F_d}{M(t)} - g, \tag{16}$$

where $S_v = 1$ for an ascending RLA and $S_v = -1$ for a descending RLA. At the level of accuracy used in this feasibility study, the Earth's rotation can be disregarded.

VFirstStage.m uses the information at time t to calculate similar information at a time $t + dt$. If the RLA's engines are still burning, then the its mass has decreased by the mass of propellant consumed:

$$dM = \frac{M \, f_p}{t_b} \, dt, \tag{17}$$

otherwise the RLA mass stays constant. The RLA's upward velocity has changed by

$$v(t + dt) = v(t) + a(t) \, dt. \tag{18}$$

Negative upward velocity means that RLA is descending. The RLA's position has changed by

$$y(t + dt) = y(t) + v(t) \, dt. \tag{19}$$

Based on the new mass, altitude, and velocity, *VFirstStage.m* calculates the RLA's drag force and acceleration at time $t + dt$. *VFirstStage.m* produces a time series of RLA's coordinates altitude y, and upward velocity v with time interval dt.

VRetroFire.m

The program *VRetroFire.m* is similar to the program *VFirstStage.m* with the following difference. The program *VRetroFire.m* incorporates a *retroburn*. The *retroburn* maneuver consists of the following. A short time (≈ 10 s) after RLA fires DLSRRs, it turns it's engines up and fires them for a given amount of time. As a result of retroburn, RLA's ascent velocity decreases and return is softened.

The user inputs the following:
    i) Time when retroburn starts;
    ii) Time when retroburn ends;
    iii) Exhaust velocity;
    iv) Fuelled rocket mass at the beginning of retroburn;
    v) Fuelled rocket mass at the end of retroburn.

VRetroFire.m also has the rocket's altitude and upward velocity at the beginning of the retroburn. *VFirstStage.m* produces a time series of RLA's coordinates altitude y, and vertical velocity dy/dt with time interval dt.

4. RLA Engines

4.1. Solid Propellant Rocket Engines

In the present subsection, we consider solid propellant engines used by light RLA. To keep the paper self-contained, we provide schematic picture of a solid propellant rocket on Figure 2 below.
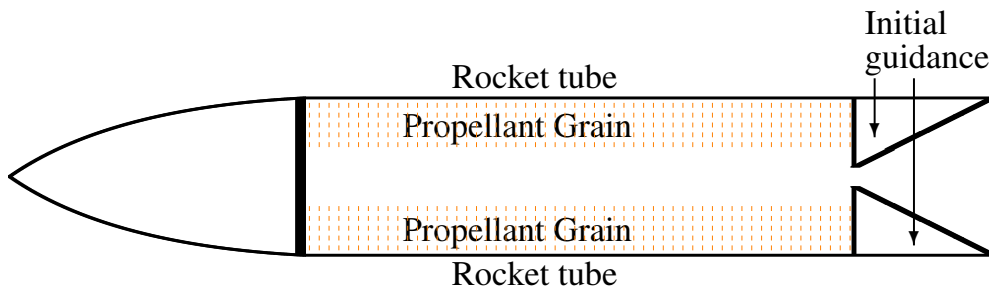

Figure 2. Solid propellant rocket.

4.1.1. Propellant Composition

Most modern artillery rockets as well as space rocket boosters use a solid propellant containing 70% ammonium perchlorate (AP), 15% aluminum powder, and 15% HTPB binder [15]. Even though artillery rockets are easily detectable, they are relatively inexpensive, while rocket-defence systems are expensive. Other propellants contain high explosives and/or highly nitrated nitrocellulose.

State of the art propellant grains are very expensive. As we show in Subsection 3.1.3 below, the cost of propellant grain

for reusable rocket boosters is \$40 to \$76 per kg. The cost of propellant grain for non-reusable rocket engines is \$180 to \$1,450 per kg.

All the aforementioned propellants have flame temperatures of 2,500 ^oC to 3,500 ^oC. Such exhaust temperatures may be incompatible with multiply reusable rocket nozzles.

RLAs have a choice of fuels burning at lower temperature and rate. Some formulations contain ammonium nitrate (AN).

Ammonium nitrate undergoes phase transitions at -18 ^oC and +32 ^oC. These phase transitions are accompanied by volume change of about 4%, thus they must be avoided in order to avoid fuel grain destruction [16]. Ammonium nitrate can be phase-stabilized by addition of 10% potassium nitrate or 2% potassium fluoride [16].

Other formulations have mildly nitrated nitrocellulose. The chemical formula for nitrocellulose is $C_6H_{10-x}O_5(NO_2)_x$. The number $0 < x < 3$ determines the oxidizer content of nitrocellulose. Nitroglycerin (NG) is used in double base propellants with nitrocellulose. NG formula is $H_5C_3N_3O_9$ [17]. By varying nitration of nitrocellulose and content of nitroglycerine, we can produce propellant with desired flame temperature. The propellants we are interested in should have flame temperatures around 2,000 ^oC.

In Table 1 below, we list several propellants. In the first column, we list propellant composition. In the second column, we list the throat temperature. In the third column, we list exhaust velocity into vacuum given an initial pressure of 70 atm and expansion of 12. Both temperature and the exhaust velocity are calculated using the program called Rocket Propulsion Analsis [18], which is available online. The fourth column lists the burning rate at 70 atm. The fifth column lists *burn rate exponent*. For almost all propellants, the burning rate is approximated by

$$r_b(P) = r_b(P_0)\left(\frac{P}{P_0}\right)^n, \qquad (20)$$

where r_b is the burning rate, P is pressure, P_0 is the reference pressure, and n is the *burn rate exponent*.

In rows 1-4, expression xByANzMA denotes a propellant composed of x% HTPB binder, y% ammonium nitrate, z% powdered *magnalium* – an alloy composed of 50% magnesium and 50% aluminum. In row 7, 90NC10N10Mg denotes a propellant composed of 10% magnesium and 90% nitrocellulose. Notation "90NC10N" mean that the nitrocellulose contains 10% nitrogen. In rows 9-10, xNGyC denotes a propellant composed of x% nitroglycerin and y% cellulose.

Table 1. Performance of solid propellants (NA – not available).

Propellant Composition	Temperature at 70 atm	v_e at 70 atm, Exp 12 Sea Level	v_e at 70 atm, Exp 12 Vacuum	Burning rate at 70 atm	n coeff.
1. 20B64AN16MA [19]	1,701 ^oC	1,993 m/s	2,258 m/s	3.7 mm/s	0.5
2. 20B60AN20MA	1,928 ^oC	2,084 m/s	2,360 m/s	NA	NA
3. 20B56AN24MA [19]	2,076 ^oC	2,174 m/s	2,456 m/s	3.8 mm/s	0.4
4. 25B55AN20MA [19]	1,861 ^oC	2,038 m/s	2,306 m/s	2.3 mm/s	0.7
5. 18% HTPB, 63% AN, 15% Mg, 4% AC [20]	1,665 ^oC	1,936 m/s	2,192 m/s	1.9 mm/s	0.05
6. Nitrocellulose 12% N	2,033 ^oC	1,966 m/s	2,230 m/s	NA	NA
7. 90NC10N10Mg	2,020 ^oC	1,960 m/s	2,222 m/s	NA	NA
8. 70% AP, 15% Al 15% HTPB [21]	2,910 ^oC	2,320 m/s	2,621 m/s	7.9 mm/s	0.35
9. 66NG34C [22]	1,924 ^oC	1,962 m/s	2,224 m/s	5.8 mm/s	0.7
10. 77NG23C [22]	2,568 ^oC	2,168 m/s	2,455 m/s	9.3 mm/s	0.7

Table 1 above is similar to Table 3 in author's work on DLSRRs [1]. The main difference is the pressure. For RLA engines it is 70 atm and for DLSRR engines it is 40 atm. DLSRR engines are discardable, thus they use less expensive materials which can sustain lower pressure.

The presence of magnesium increases combustion temperature. This temperature increase catalyses the reaction and helps it run to completion. Magnesium droplets combust very quickly and thus almost immediately add large amount of energy to the process. Magnesium has a melting point of 650 ^oC. Based on the data extrapolated from [23], the boiling point of magnesium is 1,800 ^oC at 70 atm.

Finding a propellant optimal in terms of both cost and properties remains an open problem. For now we assume, that our propellant has flame temperature 2,000 ^oC, burning rate of 3.6 mm/s, and exhaust velocity of 2,100 m/s at sea level and 2,400 m/s in vacuum. These parameters are close to those presented on Row 2 of Table 1 above. Using the densities of

energetic materials in [24], the density of our propellant should be 1.6 g/cm^3.

4.1.2. Fuel Grain Shape

The grain shape is the shape of the fuel within the rocket case. Different rockets use a wide variety of grain shapes illustrated in Figure 3 below.
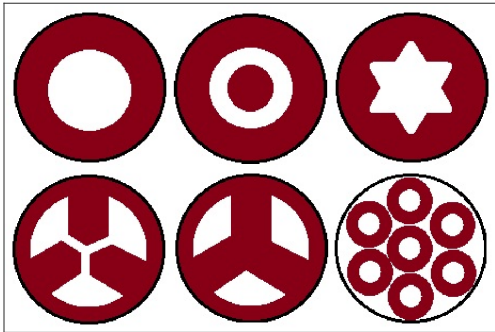

Figure 3. *Propellant grain shapes.*

DLSRRs and solid propellant RLAs may have the same propellant grain shape. The main difference is the size of the grain – DLSRRs should have a caliber of 30 cm, while solid propellant RLAs should have a caliber of 80 cm to 150 cm.

The grain shape may change over the length of the rocket tube.

Different types of grain shapes provide different thrust curves. In the present paper, we focus on grain shapes providing constant thrust. Propellant loading is the fraction of rocket tube volume occupied by the propellant. The best propellant grain shape for the RLA satisfies three requirements. First, the burning propellant surface area should experience minimal change as the propellant is burning. This would ensure steady thrust. Second, the propellant loading should be as high as possible. This would maximise fuel mass ratio within the rocket. Third, the design should be as simple as possible. This would minimize cost.

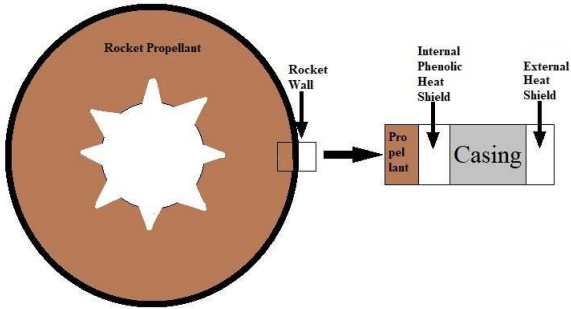A diagram of a rocket motor cross-section is given below.

Figure 4. *Rocket motor cross section.*

The casing has to withstand very high pressures during firing. The mass of casing is inversely proportional to *specific strength* of the material it is made of. Specific strength of a material is its yield strength divided by its density. Specific strength has units of

$$\frac{N/m^2}{kg/m^3} = \frac{N \cdot m}{kg} = \frac{J}{kg}. \tag{21}$$

Generally, composite materials have higher specific strength than metals. Duralumin has specific strength of $1.65 \cdot 10^5$ J/kg, maraging steel has specific strength of $2.05 \cdot 10^5$ J/kg, and hardest titanium specific strength of $2.07 \cdot 10^5$ J/kg. Best carbon-fiber reinforced plastic called Graphite 1M has specific strength of $1.4 \cdot 10^6$ J/kg [15]. Composite materials also have great fatigue resistance – they almost never fail within 3,000 cycles of pressurization and depressurization [25]. Thus, casing of the Light RLA motor would be made from Graphite 1M carbon-fiber reinforced plastic.

4.1.3. Propellant Grain Cost

Manufacture of solid rocket engines involves countless difficulties and expenditures. Least expensive units are large rocket boosters. Data for these boosters can be found in [26]. It is tabulated below. The costs in the Table 2 below are in 2018 dollars.

Table 2. *Cost of large rocket boosters.*

Rocket	Cost per 1 kg propellant	Cost per 1 N thrust	Cost per 1000 N·s
Titan IV strap-on	$76	$3.7	$30
Titan IV upgrade strap-on	$69	$3.0	$24
Shuttle booster	$40	$1.7	$17
Castor 4A strap-on	$63	$1.7	$34
Graphite motor	$59	$1.4	$24

Even though Shuttle boosters need to be refurbished between the flights, their costs are relatively low due to reusability.

Military rockets generally carry higher cost. In Table 3 below we present the cost information on some of the rockets currently employed by US Military and Air Force.

Table 3. *Cost of military rocket motors.*

Rocket	Propellant Mass	Motor Cost Fraction [27]	Rocket cost	Motor cost	Motor cost per 1 kg propellant
GMLRS	98 kg	13%	$136,000	$17,700	$180
SM-6	829 kg	20%	$4,100,000 [28]	$820,000	$990
SM-3	923 kg	8%	$12,000,000 [29]	$960,000	$1,040
PAC-3	159 kg	3%	$7,700,00 [29]	$230,000	$1,450
Trident II	30,255 kg	21%	$37,300,000	$7,800,000	$258

SM-3, SM-6, and PAC-3 rocket engines have exorbitant cost. GMLRS and Trident II have lower cost due to economics of scale. Trident II has large scale of itself, while GMLRS is produced by thousands. The only reason solid rocket motors have been used for large rockets is their superiority to non-reusable liquid propellant motors. Solid rocket motors are thrice less expensive and are more reliable [30].

The cost for solid fuel grain of RLA must be lower. First, RLA motor uses Ammonium Nitrate propellant or low-grade nitrocellulose. This brings very significant savings. Second, RLA engine is reusable. Only the propellant grain has to be reloaded between firings. Hopefully, we can reduce the cost of RLA propellant grain to $10 per kg.

4.2. Liquid Propellant Rocket Engines

In a liquid-rocket engine, both fuel and oxidizer are fed into the combustion chamber. Both liquids come into the chamber through hundreds of spray nozzles in order to ensure rapid mixing. Inside combustion chamber, fuel reacts with oxidizer producing combustion with temperature between 1,500 ^{o}C and 3,500 ^{o}C. Engines with higher combustion temperature generally produce higher specific impulse, yet high combustion temperature also causes rapid erosion of combustion chamber. Combustion products are expanded through a nozzle producing jet stream in one direction and thrust in the opposite direction.

There are two types of systems which pressurise fuel and oxidizer for combustion chamber. In a pump-feed rocket engine, the fuel and oxidizer come into the pump at a relatively low pressure. The pump pressurizes fuel and oxidizer and feeds them into the combustion chamber. The pump is powered by a small fraction of fuel and oxidizer. In the pressure-fed rocket engine, the fuel and oxidizer are contained in corresponding tanks at a high pressure.

RLA engine is pump-fed. Both oxidizer and fuel have to be fed into combustion chamber at a pressure of at least 100 *bar*. The chamber pressure itself is 70 *bar*. However, an increase of chamber pressure increases instability.

4.2.1. Possible Liquid Propellants

Rockets use several oxidizers – liquid oxygen, concentrated solutions of hydrogen peroxide $\left(H_2O_2\right)$ in water, nitric acid $\left(HNO_3\right)$, nitrogen tetroxide $\left(N_2O_4\right)$ [15]. A 95% solution of hydrogen peroxide can undergo catalytic decomposition producing steam and oxygen at 870 ^{o}C [18], which is above the autoignition temperature of

most fuels. That is an important advantage for rocket engines. In the past hydrogen peroxide had problems of instability, but with modern containers and stabilizers, concentrated hydrogen peroxide is very stable. In drum quantities, it loses up to 0.4% H_2O_2 per year [31]. Dissociation rate of hydrogen peroxide is inversely proportional to water content [32] – thus high grade peroxide is more stable than low grade one.

State of the art rockets use many different fuels and oxidizers [15]. Performance of several liquid fuel-oxidizer combinations is tabulated in Table 4 below. In the first column, ethylene oxide is written as EthOx. The second column lists the oxidizer. HP95 denotes 95% solution of hydrogen peroxide. LO2 denotes liquid oxygen. The third column is the oxidizer to fuel mass ratio. The fourth column is the temperature at rocket throat. Throat temperature is lower than adiabatic flame temperature. The fifth column is exhaust velocity at sea level. The sixth column is exhaust velocity in vacuum. The fifth and sixth columns, "Expansion 12" means throat-to-nozzle area ratio is 12.

The values in the last three columns are deduced from the values calculated by Rocket Propulsion Analysis (RPA) program [18]. These values are actual rather than ideal. The ideal velocities are calculated by RPA. Actual sea level exhaust velocity is the ideal sea level exhaust velocity multiplied by 0.92. Actual vacuum exhaust velocity is the ideal vacuum exhaust velocity multiplied by 0.94.

For RLA engine, we would like a non-cryogenic oxidizer and exhaust temperature to be at most 1,800 ^{o}C. At that temperature methanol fuel with 95% H_2O_2 oxidizer produces the lowest exhaust velocity, propane produces the highest, while ethylene oxide is the second best. In our opinion, the advantage of ethylene oxide such as easy flammability outweigh any disadvantages.

Hypergolic propellants ignite as soon as propellant spray and fuel spray are combined. Red fuming nitric acid (RFNA) consists of 79% HNO_3 and 19% N_2O_4 [33]. This oxidizer is hypergolic with fuel consisting of carene and norbornadiene. Concentrated hydrogen peroxide is hypergolic with the following fuel mixtures. First, it is hypergolic with ETA – the mixture of ethanolamine and 10% $CuCl_2$ [34]. Second, it is hypergolic with ETAFA – the mixture of 47.5% Ethanolamine, 47.5% Furfuryl Alcohol, and 5.0% $CuCl_2$ [35]. Third, it is hypergolic with pyrrole [36]. Fourth, it is hypergolic with EEC – the mixture of 61% monoethanolamine, 30% ethanol, and 9% hydrated copper nitrate. On contact, EEC ignites with a delay of only $1.6 \cdot 10^{-2}$ s [37].

Table 4. *Performance of liquid bipropellants.*

Fuel	Oxidizer	Oxidizer to fuel ratio	Temperature	v_e at 70 atm, Expansion 12 Sea Level	v_e at 70 atm, Expansion 12 Vacuum
EthOx	HP95	1.31	1,801 ^{o}C	2,180 m/s	2,470 m/s
EthOx	HP95	3.8	2,654 ^{o}C	2,473 m/s	2,793 m/s
Methanol	HP95	1.74	1,801 ^{o}C	2,150 m/s	2,435 m/s
Methanol	HP95	3.3	2,407 ^{o}C	2,375 m/s	2,684 m/s
Propane	HP95	3.67	1,800 ^{o}C	2,218 m/s	2,513 m/s
Propane	HP95	7.8	2,561 ^{o}C	2,465 m/s	2,726 m/s
EthOx	LO2	0.52	1,800 ^{o}C	2,200 m/s	2,495 m/s
EthOx	LO2	1.8	3,300 ^{o}C	2,616 m/s	2,950 m/s
Methanol	LO2	0.68	1,790 ^{o}C	2,159 m/s	2,447 m/s
Methanol	LO2	1.5	2,960 ^{o}C	2,549 m/s	2,874 m/s
Propane	LO2	1.45	1,800 ^{o}C	2,292 m/s	2,599 m/s
Propane	LO2	3.6	3,290 ^{o}C	2,675 m/s	3,017 m/s

Performance of several hypergolic combinations is tabulated in Table 5 below. The first column lists the fuels. The second column lists the oxidizers. HP95 denotes 95% solution of hydrogen peroxide. The third column is the oxidizer to fuel mass ratio. Throat temperature is lower than adiabatic flame temperature. The temperature of the flame where oxidizer and fuel contact is higher. The fifth column is exhaust velocity at sea level. The sixth column is exhaust velocity in vacuum. The fifth and sixth columns, "Expansion 12" means throat-to-nozzle area ratio is 12. We assume a shifting chemical equilibrium in the rocket due to high temperature.

Table 5. *Performance of hypergolic propellants.*

Fuel	Oxidizer	Oxidizer to fuel ratio	Temperature	v_e at 70 atm, Expansion 12 Sea Level	v_e at 70 atm, Expansion 12 Vacuum
ETA	HP95	2.36	1,799 ^{o}C	2,042 m/s	2,313 m/s
ETA	HP95	3.4	2,220 ^{o}C	2,226 m/s	2,517 m/s
ETAFA	HP95	2.0	1,798 ^{o}C	2,055 m/s	2,327 m/s
ETAFA	HP95	3.8	2,408 ^{o}C	2,307 m/s	2,607 m/s
EEC	HP95	2.59	1,802 ^{o}C	2,099 m/s	2,378 m/s
EEC	HP95	4.6	2,367 ^{o}C	2,323 m/s	2,626 m/s

A *monopropellant* is a substance which can burn by itself without an oxidizer. It contains both fuel and oxidizer in a metastable mixture or component. In Table 6 we tabulate the performance of several monopropellants. The second column is the throat temperature.

Table 6. *Performance of liquid monopropellants.*

Monopropellant	Temperature	v_e at 70 atm, Expansion 12 Sea Level	v_e at 70 atm, Expansion 12 Vacuum
27% Ethylene 73% N_2O	1,802 ^{o}C	2,141 m/s	2,344 m/s
11% Ethylene 89% N_2O	3,090 ^{o}C	2,382 m/s	2,693 m/s
Nitromethane	2,185 ^{o}C	2,200 m/s	2,495 m/s
92% Nitromethane 8% Methanol	1,805 ^{o}C	2,085 m/s	2,314 m/s
76% Nitromethane 24% Nitroethane	1,803 ^{o}C	2,089 m/s	2,369 m/s

Monopropellants burning at about 1,800 ^{o}C have slightly lower exhaust velocity than bipropellants. The choice of monopropellant or bipropellant would also depend on combustion process.

The first two rows of Table 6 represent Nitrous Oxide Fuel Blend. The work on stabilizing the formula containing 11% Ethylene 89% N_2O is currently in progress. In such proportions, the mixture is extremely reactive and very unstable. Mixtures with much lower concentrations of nitrous oxide are much more stable, have much lower combustion temperature, and thus have lower exhaust velocity. One advantage of nitrous oxide fuel mixture is that it has a low boiling point, thus it self-pressurises the fuel tank if it is kept at about 0 ^{o}C to 20 ^{o}C. Vapor pressure of nitrous oxide is tabulated below [38]:

Table 7. Nitrous oxide properties.

Temperature	Vapour pressure	Liquid density
–23 ^{o}C	16.5 atm	1.04 kg/L
–12 ^{o}C	23.1 atm	0.98 kg/L
0 ^{o}C	31.3 atm	0.90 kg/L
10 ^{o}C	40.7 atm	0.84 kg/L
15 ^{o}C	45.1 atm	0.82 kg/L
21 ^{o}C	52.4 atm	0.75 kg/L

4.2.2. Liquid Propellant Cost

The prices listed below are from 2010s, most commonly 2019. Furfuryl alcohol costs $1.00 to $2.00 per kg [39]. Ethanolamine costs about $1.80 per kg [40]. Ethylene oxide costs $1.50 per kg [41]. Ethylene costs $0.90 per kg [42]. The 50% solution hydrogen peroxide in water costs $0.50 per kg [43]. The price of 50% solution hydrogen peroxide in water on IndiaMart in late 2020 is $0.40 per kg. Given that hydrogen peroxide purification also requires processing work, 98% pure hydrogen peroxide should be more expensive than a similar weight of hydrogen peroxide in 50% solution. Hydrogen Peroxide Handbook provides the latest data from 1967 [44]. At that time, hydrogen peroxide cost seven times as much as now if we adjust for inflation. An important point is that concentrating hydrogen peroxide from 70% to 98% increased it's price on peroxide basis only by 26%. Thus we can be certain that with prices for 50% solution and modern technology it is possible to produce 95% pure hydrogen peroxide at $2.00 per kg. The prices of both the fuels and oxidizers are low enough to fuel the RLA.

As of 2019, Nitromethane in large quantities costs $1,428 per 42 gallon drum or $8 per kg [45]. One may worry that nitromethane prices will rise when the product is in greater demand, but it is unlikely. Nitromethane can be produced by nitration of methane with nitric acid and oxygen at 435 ^{o}C. Nitration of ethane yields 10%-20% nitromethane and 80%-90% nitroethane [46]. Methane can be nitrated by nitric acid at 410 $^o C$ and 18:1 molar ratio of methane to nitric acid. The process is 33% efficient [47]. Nitrous Oxide costs $8 – $11 per kg [48].

4.3. Rocketprop Landing Engines

Aside from the main rocket engine, RLA may have two or more auxiliary engines. The auxiliary engines are used for landing. Several concepts for auxiliary engines of RLA exist. In our opinion, the rocketprop engine is the best choice for an engine used only for landing.

A rocketprop consists of a rotary engine and a propeller. The rotary engine is similar to that of an air turborocket described in Fundamentals of Aircraft and Rocket Propulsion [49]:

> (The propeller) is driven by a multi-stage turbine; the power to drive the turbine is derived from combustion of (rocket fuel) in a rocket-type combustion chamber. Since the gas temperature will be in the order of 3000 ^{o}C, additional fuel is sprayed into the combustion chamber for cooling purposes before the gas enters the turbine.

After the gas leaves the turbine it further expands and comes out of the nozzle. This provides additional thrust. Sometimes, before coming out of the nozzle, the gas is heated by addition of extra oxidizer.

The turbine inlet temperature for a small uncooled turbine can be at most 800 ^{o}C [49]. The choice of fuel and propellant has to be such that propellant excess would not produce carbon residue particles.

The turbine efficiency is at least 90% [49]. Turbine efficiency is the fraction of the gas energy released under ideal expansion converted into motive power. The gas energy released under ideal expansion depends on initial gas temperature, initial gas pressure, final gas pressure, and chemical equilibrium composition of the gas. This energy can be calculated by the Rocket Propulsion Analysis program [18].

Performance for several liquid propellants for

aforementioned rotary engines is tabulated in Table 8 below. The cooling of the gases inside the turbine has been taken into account within the calculations programmed in RPA [18]. The first column is fuel. EthOx40 denotes 40% ethylene oxide and 60% water. EthOx50 denotes 50% ethylene oxide and 50% water. Ethanol50 denotes 50% ethanol in 50% water.

The second column is oxidizer. LO2 denotes liquid oxygen.

The fifth column is energy transmitted into engine rotary power per gram propellant, when the gas expands from 40 atm pressure to 3 atm pressure over a 90% efficient turbine.

The sixth column is the exhaust velocity after the stream is further expanded from 3 atmospheres to 1 atmosphere.

Table 8. *Performance of propellants for turbine rotary engines.*

Fuel	Oxidizer	Oxidizer to fuel ratio	Temperature	Propellant Energy	Exhaust Velocity
EthOx40	85% H_2O_2	0.6	810 $^\circ$C	950 J/g	1,250 m/s
Methanol	85% H_2O_2	0.7	820 $^\circ$C	1,070 J/g	1,170 m/s
Propane	65% H_2O_2	3.0	810 $^\circ$C	1,070 J/g	1,180 m/s
Kerosine	65% H_2O_2	2.7	815 $^\circ$C	1,020 J/g	1,150 m/s
EthOx50	LO2	0.2	810 $^\circ$C	960 J/g	1,100 m/s
Methanol	LO2	0.28	810 $^\circ$C	1,020 J/g	1,280 m/s
Ethanol50	LO2	0.38	810 $^\circ$C	970 J/g	1,080 m/s

The propeller generates thrust by using the mechanical power provided by the engine to create backward airstream. Propeller efficiency is the fraction of rotary engine energy used to create the backward airstream and to propel a vehicle. The rest of the energy goes into the air stream wake behind the propellers and into the turbulence. During the rocket powered ascent, the propellers are turned parallel to the air stream (see Figure. 5). The total efficiency of gearbox-propeller system is at least 70% [49].

As a landing engine, a rocketprop is much more suitable than a rocket. At low velocity, the fuel produces much greater specific impulse in a rocketprop than in a rocket.

Conventional turbojet, turboprop, and piston driven propeller engines provide much greater specific impulse than rocketprop engines. As we see from the Table 8 above, one gram of rocketprop propellant provides 950 J to 1,070 J to the rotary engine. This value is very low relative to other engines. Diesel engines obtain about 19,000 J of rotary energy per gram fuel [50].

The main factor making rocketprop engine very useful as an RLA landing engine is it's high specific power. The landing engine does not need a lot of energy, since the landing can take less than a minute. Conventional turbojet, turboprop, and piston driven propeller engines have much higher mass per unit power and mass per unit thrust than rocketprop. These engines are much more suitable for long range aircraft, and useless as RLA landing engines.

5. Rocket Launcher Aircraft

5.1. Variety of RLAs and Their Trajectories

5.1.1. RLAs

RLAs would have much greater variability than fighters, bombers, and most other types of military aircraft. First, RLAs can have a wide range of liftoff masses – from under 1 ton to over 1,000 tons. The lightest RLAs should be reusable solid propellant rockets. DLSRRs would be discardable second stages. After any first stage burns out and fires the second stage, the RLA lands and the mode of landing depends of the type of RLA. Namely, a lighter RLA lands on a parachute, a medium RLA lands using a rocketprop engine, and a heavier RLA lands using a rocket engine.

RLAs with liftoff masses ranging from 5 tons to 500 tons can be modified versions of space boosters and ballistic missiles currently in use. Spanish rocket Arion 2 has liftoff mass 7 tons. American rocket Electron has liftoff mass 10.5 tons. American rocket Intrepid 1 has liftoff mass 24.2 tons. Russian rocket Rockot has liftoff mass 107 tons. Russian rocket Angara 1.2 has liftoff mass 171 tons [51].

Jeffery Becker has a concept of heavy RLA, which he calls Theater Guided Missile Carrier (T-CVG). In Becker's concept, a modified version of reusable first stage of Falcon I rocket is used as RLA. This stage rises vertically, firing second stage rockets which have a range of 1,600 km to 8,000 km [52]. Falcon 9 has liftoff mass of about 550 tons – thus this RLA is on the heavier end of the spectrum. New Glenn rocket built by Blue Origin Company has a reusable first stage. New Glenn's liftoff mass is undisclosed, but based on thrust it should be at least 1,300 tons. It is powered by seven BE-4 engines which use liquid methane fuel and liquid oxygen oxidizer [51]. The main engines of the first stage burn 198 s and accelerate the second stage to 2.7 km/s [53]. In our opinion, a modified version of New Glenn rocket can make an excellent heavy RLA.

Second, RLAs can use different types of engines. Light RLAs can be powered by solid propellant rockets. RLAs with liftoff mass of 5 tons to 50 tons can be powered by liquid propellant, solid propellant, or hybrid propellant rockets. Liquid propellant rockets have lower fuel costs and higher

engine costs. Solid propellant rockets have higher fuel costs and lower engine costs [26]. Heavy RLAs can be powered by liquid propellant rockets. It is possible that some RLAs can be powered by air-breathing engines such as turbojet or air turborocket.

Third, RLAs can use different types of propellant. For liquid propellant rockets, there are many choices of fuel, oxidizer, and a wide range of possible fuel-oxidizer mass ratios. In Subsection 3.2, we discuss several possibilities for liquid propellants. For solid propellant rockets, there are many choices of propellant and grain shape. As we discuss in Subsection 3.1, some types of solid propellant are prohibitively expensive. For hybrid propellant rockets, there are many choices of solid fuel and liquid oxidizer. For all types of rockets, designing optimal combinations in terms of both cost and performance remains a challenging problem.

Fourth, RLAs can use different types of rocket engines producing given thrust. There is variability among different liquid propellant rocket engines, among different solid propellant rocket engines, and among different hybrid propellant rocket engines. Different engines have different chamber pressures and expansion ratios. High chamber pressure improves performance but increases the engine cost. Fifth, RLAs can use different types of airframes and different airframe materials. Generally, materials with superior properties are more expensive. Finding optimal balance is an engineering problem. Sixth, RLAs may be either completely or partially reusable. One concept of partially reusable RLA has reusable engines and discarding fuel and oxidizer tanks. We explore this concept as Medium RLA below.

5.1.2. RLA Trajectories

Aside from the vast variabilities of RLAs, each RLA can fly on several trajectories. An RLA can rise almost vertically, then bend its trajectory and release DLSRRs at an angle of 20° to 40° with respect to the vertical axis. This trajectory is called *forward trajectory*. An RLA can rise vertically and land at its liftoff point. This is called *vertical trajectory* RLA. (In our time frame, it is acceptable to disregard the correction due Earth's rotation.) DLSRRs fired by a vertical trajectory RLA will have to supply the horizontal component of their velocity by using their own engines.

Both vertical and forward trajectories have advantages. The advantage of forward trajectory is that DLSRRs are fired at an angle with respect to the vertical axis. DLSRR engines can exert thrust in the direction of DLSRR motion throughout DLSRR ascent. DLSRR engine provides the entire forward velocity component, while RLA provides most of upward velocity component. The advantage of vertical trajectory is the fact that an RLA on forward trajectory will land at a distance of tens to hundreds of kilometers away from the point of its liftoff. Returning RLA from such distance to the firing point may be time-consuming. Alternatively, an RLA may have wings and auxiliary engines to return to firing point. This will require

RLA to have a lot of extra mass and subsequently have lower propellant mass fraction f_p.

Performance of RLA is defined as the set of the following parameters. The first parameter is the velocity at which DLSRRs are launched. High launch velocity reduces work requirement for DLSRR engines. The three RLAs we are considering release DLSRRs at 1,200 m/s to 2,355 m/s. The second parameter is the altitude from which DLSRRs are launched. High launch altitude reduces DLSRR aerodynamic velocity loss. High launch altitude also reduces the aerodynamic heating of DLSRR. Even though DLSRR is easily detectable it is a relatively inexpensive rocket while rocket-defence systems are expensive. The three RLAs we are considering release DLSRRs at an altitude of 40 km to 100 km. The third parameter is the DLSRR mass fraction f_r out of liftoff mass. Payload delivered to target is directly proportional to f_r. The three RLAs we are considering have f_r between 0.038 and 0.1.

Other parameters, which are of limited interest is DLSRR flight time, and DLSRR flight ceiling. The three RLAs we are considering have flight times of 6.5 min to 8.0 min and flight ceilings of 114 km to 198 km.

5.2. RLAs Considered

In this subsection, we describe three RLAs envisioned by the author. The author suggests a set of realistic tentative parameters for the three RLAs. Designing an actual rocket with optimal parameters is far beyond the scope of this work. In fact, engineering a rocket with all details requires many expert-years of theoretical and experimental work [11].

Prior to describing the three RLAs considered in this work, we introduce several parameters. The *gross liftoff mass* M_{GL} is the total mass of loaded and fuelled RLA at liftoff. The *propellant mass fraction* f_p is the ratio of the mass of propellant used by the main engine during ascent to gross liftoff mass. The *RLA mass fraction* f_a is the ratio of empty RLA mass to gross liftoff mass. The *DLSRR mass fraction* f_r is the ratio of the mass of DLSRR with containers to gross liftoff mass. The *auxiliary fuel mass fraction* f_f is the ratio of the mass of fuel used after the firing of DLSRR to gross liftoff mass. The fuel counted in f_f is the fuel used for landing and possible retroburn. The total distribution of mass in a fully loaded RLA can be written as

$$f_p + f_a + f_f + f_r = 1. \qquad (22)$$

Below we present three RLA concepts. First one is Light RLA. Second one is Medium RLA. The third one is Heavy RLA.

5.2.1. Light RLA

Light RLA is a reusable solid propellant rocket with auxiliary landing engines. It is shown in Figure 5 below:
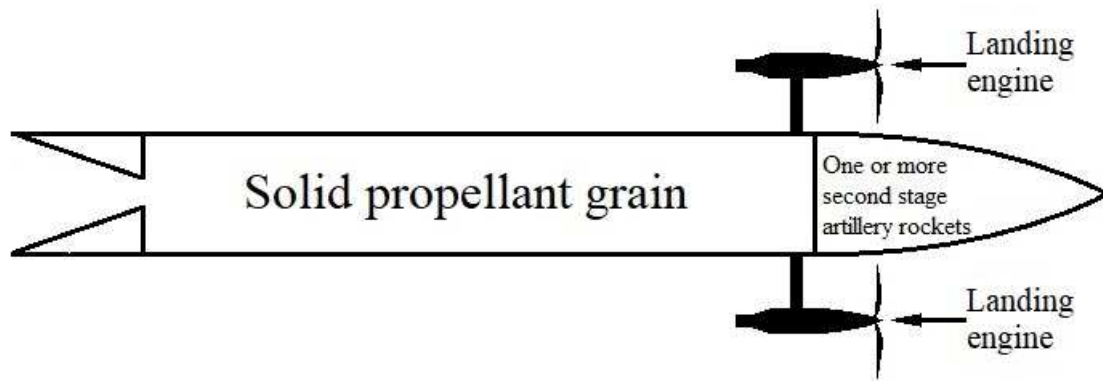
Figure 5. *Light RLA.*

As we show in Figure 4, the rocket propellant covered by a phenolic heat shield is encased into a high pressure vessel. A propellant grain, its phenolic inner casing and its igniter forms a single *solid charge*. In Light RLA, the solid charge is the only part expended in every sortie. During each sortie, the propellant grain burns out and the heat shield is scorched and becomes useless. Replaceable solid charge is inserted into casing. As discussed in Subsection 3.1, the casting is made out of Graphite 1M carbon-fiber reinforced plastic.

Below, we present dimensions and mass distribution of the Light RLA. The RLA without packaged DLSRRs has length of 7.5 m. RLA has diameter of 1.2 m. The solid charge has length of 6 m and diameter of 1.16 m. Gross liftoff mass of Light RLA is 10 tons. Solid fuel makes up 6 tons. Packaged DLSRRs make up 1 ton. Dry mass of Light RLA is 2.3 tons. Phenolic inner casing has a mass of 200 kg, and landing fuel has mass of 500 kg.

The engine has internal pressure of 70 atm and nozzle expansion ratio of 12. Exhaust velocity is 2,040 m/s at sea level, and 2,285 m/s in vacuum. Flight averaged exhaust velocity is 2,177 m/s.

5.2.2. Medium RLA

Medium RLA is a reusable rocket with discarding fuel and oxidizer tanks presented in Figure 6 below:
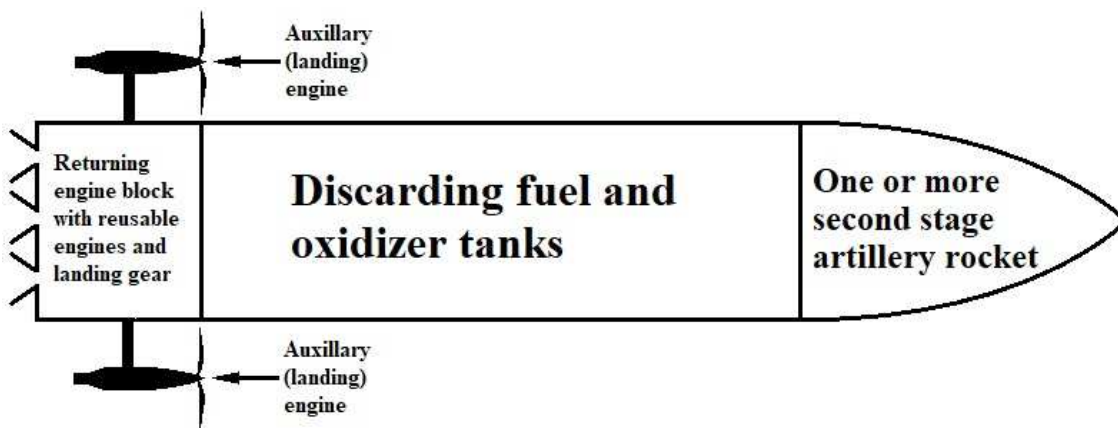

Figure 6. *Reusable rocket with discarding tanks.*

Rigorously speaking the RLA considered below is partially reusable; however, we will use the term "reusable" for the remaining part of the paper. The cost of discarding fuel tanks is orders of magnitude lower then the cost of RLA. The concept of aircraft using discarding fuel tanks also known as *drop tanks* is not new. Many aircraft have used drop tanks since 1940s [54, 55, 56]. Currently United Launch Alliance is working on Vulcan rocket – a space launch rocket in which the propellant tank is discarded, while the engines are recovered [57]. Airbus is developing Adeline – a rocket stage in which the fuel tank is discarded, while the engine returns to the spaceport using two propeller engines [58].

The discarding propellant tank is a partitioned cistern 2 m in diameter and 10 m long. The cistern with these dimensions can hold 10 tons of EEC fuel in one compartment and 26 tons of 95% hydrogen peroxide oxidizer in the other compartment. The total mass of propellant used in liftoff is thus 36 tons.

In order to estimate the total liftoff mass, and masses of individual components, we estimate component mass fractions. The mass fraction of propellant used during ascent is $f_p = 0.65$ by design. Given the liftoff propellant mass and its mass fraction, we find the gross liftoff mass of $M_{GL} = 55$

tons. The masses of leading engines are included within the original RLA mass.

Rocket engines generally have thrust-to-weight ratios of 70 or more [15]. Engines designed to be reused hundreds of times should have lower thrust to weight ratios, which still exceed 30. The initial thrust of the rocket we are designing is about $1.5\ g\ M_{GL}$. Thus, the rocket engine proper has a mass fraction $f_{RE} \leq 0.05$. The mass fraction for the engine block with fuel used for return and landing should be at most 0.13, which implies the mass of 7.2 tons.

The mass fraction for discarding propellant cistern should be 0.12 [59], which implies the mass of 6.6 tons. Subtracting aforementioned mass fractions from 1, we obtain the mass fraction of DLSRR with containers as $f_r = 0.1$, which implies the mass of 5.5 tons.

The main rocket engine has firing time of 90 s. It uses EEC fuel and 95% hydrogen peroxide propellant. It has combustion chamber temperature 1,800 ^{o}C, combustion chamber pressure of 70 atm, and expansion area ratio of 12. Exhaust velocity is 2,100 m/s at sea level, and 2,380 m/s in vacuum. Flight averaged exhaust velocity is 2,277 m/s.

5.2.3. Heavy RLA

This RLA is a modified version of the first stage of Falcon 9 rocket. Before describing the modified version of the rocket, we describe the original. The first stage of unmodified Falcon 9 rocket has a dry mass of 25.6 tons. It holds 395.7 tons of propellant. The engine firing time is 162 s. The engines consume 2.44 tons of propellant per second. The stage has diameter of 3.7 m and length of 41.2 m [60]. Falcon 9 first stage uses 9 Merlin 1D engines. These engines use kerosene fuel and liquid oxygen oxidizer. The chamber pressure is 97 atm and the area expansion ratio is 16. The fuel to oxidizer ratio is 1:2.33 [61]. They have exhaust velocity of 2,766 m/s at sea level and 3,140 m/s in vacuum [62].

Unlike the first stage of Falcon 9, Heavy RLA would have to perform hundreds of sorties. Thus, it would have to be sturdier, and it's performance would be somewhat lower than that of first stage of Falcon 9. We intend the engines of Heavy RLA to fire for 120 s. Given propellant consumption rate, the total mass of propellant used during Heavy RLA ascent is 293 tons. In order to ensure sturdiness and multiple reusability, the dry mass of Heavy RLA should be at least 1.5 times higher than that of the first stage of Falcon 9. The dry mass of Heavy RLA should be 39 tons. The mass fraction of propellant used during ascent is $f_p = 0.75$ by design. Given the liftoff propellant mass and its mass fraction, we find the gross liftoff mass of $M_{GL} = 390$ tons. A mass of

$$M_{GL} - M_{Dry} - M_{Liftoff\ Fuel} = 58\ tons$$

is divided between DLSRR, DLSRR package, and fuel used after DLSRR release. Our tentative division of the mass above is 43 tons fuel and 15 tons packaged DLSRR. The use of fuel is explained in Subsection 4.3 below.

The engines of Heavy RLA are modified versions of Merlin 1D engines. Like their prototype, these engines use kerosene

fuel and liquid oxygen oxidizer, the chamber pressure is 97 atm and the area expansion ratio is 16. According to a calculation using Rocket Propulsion Analysis software [18], the oxidizer to fuel ratio of 2.33:1 used in Merlin 1D engines generates combustion chamber temperature of 3,357 ^{o}C. This temperature is optimal for Merlin 1D engines, but it is impossible for Heavy RLA engines which must be reusable several hundred times. Reducing oxidizer to fuel ratio to 1.425 reduces combustion chamber temperature to 2,050 ^{o}C, which should be suitable for Heavy RLA engines. It also reduces exhaust velocity to 85% of their original value. Heavy RLA engines have exhaust velocity of 2,350 m/s at sea level, and 2,670 m/s in vacuum. Flight averaged exhaust velocity is 2,592 m/s.

5.3. Vertical Sortie

5.3.1. Outline

The role of RLA is to fire artillery rockets from high altitude at a high initial velocity. RLA should be able to perform the firing maneuver thousands of times during its' service. RLA is unmanned.

Any RLA should be suitable for a vertical trajectory sortie as well as a forward trajectory sortie. In this work, we consider the vertical trajectory. Maneuvers performed by an RLA on vertical trajectory are described below. We also suggest possible parameters for such maneuvers.

1. RLA uses its main rocket engine for vertical ascent.
2. RLA raises the DLSRR to a high *firing altitude* h_f, where the air resistance would have low effect on it. A reasonable altitude is $h_f \geq 25$ km.
3. RLA fires the DLSRR with *initial velocity* $v_0 \geq 1,200$ m/s at release. Initially DLSRR flies in the vertical direction. DLSRR must generate horizontal component of velocity using its own engines.
4. Given that RLA releases the artillery rocket at an altitude h_f and vertical velocity v_0, the ceiling of RLA should be

$$h_c = h_f + \frac{v_0^2}{2g} \geq 110\ km. \qquad (23)$$

During the sortie, RLA must avoid being hit by antiaircraft fire. As we discuss below, a RLA flying over the firing site is much less vulnerable than a fighter or a bomber aircraft flying close to a target.

5. The RLA should complete each sortie in 6 to 11 minutes. It should be able to be refuelled and reloaded with another set of artillery rockets as soon as possible. Under ideal conditions, RLA should be able to fly 2 to 4 sorties per hour and up to 30 to 50 sorties per day.

5.3.2. Schedules

We calculate the flight schedule of the rockets along with time series of their altitudes and velocities using the programs *VFirstStage.m* and *VRetroFire.m*. As described in Subsection 2.1, the mathematics used by the programs is straight forward. Since we have all information about parameters of rocket

engines, the only thing we need to know is the drag coefficient C_D used in (15). It is calculated by RASAero software [63]. We use a typical 1.2 m diameter rocket as a model for RLA. For larger rockets, the drag coefficients are slightly lower. For RLAs, the base area must include the base area of landing engines and connections to them. The drag coefficients are presented in Table **??** below.

Table 9. Drag coefficient of RLA.

$\mathcal{M}$	.25	.50	.75	.90	1.05	1.25	1.50	1.75	2.00
C_d	.21	.21	.21	.21	.41	.41	.41	.37	.34
$\mathcal{M}$	2.25	2.50	3.00	3.50	4.00	4.50	5.00	5.50	6.00
C_d	.31	.29	.26	.23	.21	.19	.17	.16	.15

Below we present vertical flight schedules for the three RLAs described above. The times are given in minutes:seconds. First, we consider the Light RLA.

-00:05	Main rocket engine starts.
00:00	Vertical liftoff using the main rocket engine.
01:00	Main rocket engine turns off. Altitude 29 km. Velocity 1,299 m/s up.
01:09	DLSRRs fired. Altitude 40 km. Velocity 1,202 m/s up.
01:09 – 5:25	RLA on ballistic trajectory. Vulnerable to long-range rockets.
03:11	Flight culmination. Altitude 113.5 km.
05:25	RLA reaches the altitude of 25 km. Velocity 1,310 m/s down.
05:55	Landing engines turned on. Altitude 3 km. Velocity 279 m/s down.
06:25	RLA landing.

Second, we consider the Medium RLA.

-00:05	Main rocket engine starts.
00:00	Vertical liftoff using the main rocket engine.
01:30	Main rocket engine turns off. Altitude 44 km. Velocity 1,426 m/s up.
01:35	DLSRRs fired. Altitude 50 km. Velocity 1,380 m/s up.
01:30 – 6:30	RLA on ballistic trajectory. Vulnerable to long-range rockets.
03:55	Flight culmination. Altitude 147.1 km.
06:33	RLA reaches the altitude of 25 km. Velocity 1,540 m/s down.
07:32	Landing engines turned on. Altitude 3 km. Velocity 126 m/s down.
08:02	RLA landing.

Third, we consider the heavy RLA. After releasing DLSRR, Heavy RLA takes a different course of action than the light and medium RLAs. Heavy RLA points the engines up and uses them to decelerate. This maneuver is called *retroburn*. State of art Falcon 9 first stage rockets use engines to decelerate during descent. This maneuver softens rocket entry into dense atmospheric layers [64]. Without retroburn, a rocket would reenter the dense layers of atmosphere at a very high velocity. In that case, the rocket would experience extremely high deceleration, which would break the rocket. Performing the retroburn maneuver while the rocket is still ascending softens rocket entry into dense atmospheric layers as well, and it also decreases the time the rocket spends on ballistic trajectory. This decreases the rocket's vulnerability to air defense. A sortie schedule inclusive of retroburn is below.

-00:05	Main rocket engine starts.
00:00	Vertical liftoff using the main rocket engine.
02:00	Main rocket engine turns off. Altitude 90 km. Velocity 2,395 m/s up.
02:04	DLSRRs fired. Altitude 100 km. Velocity 2,355 m/s up.
02:04 – 02:14	RLA turns by 180°.
02:14	Retroburn begins. Altitude 123 km. Velocity 2,260 m/s up.
02:14 – 2:44	At 2:15, RLA has empty mass 39 tons and 43 tons propellant. During retroburn, 30 tons propellant is consumed.
02:44	Retroburn ends. Altitude 169.5 km. Velocity 747 m/s up.
02:44 – 7:08	RLA on ballistic trajectory. Vulnerable to long-range rockets.
04:00	Flight culmination. Altitude 197.9 km.
07:08	RLA reaches the altitude of 25 km. Velocity 1,797 m/s down.
07:23	Landing engines turned on. Altitude 5 km. Velocity 718 m/s down.
07:23 – 7:38	During landing, 12 tons of propellant is consumed. One ton propellant remains in tanks and tubing.
07:38	RLA landing.

5.4. Summary of RLA Performance

In Table 10 below, we summarize parameters for the three RLAs and their sorties.

Table 10. *Performance of liquid monopropellants.*

	Light RLA	Medium RLA	Heavy RLA
Mass distribution			
Gross liftoff mass, ton	10.0	55.0	390
Liftoff propellant mass, ton	6.0	35.7	293
Landing propellant mass, ton	0.5	1.2	43
RLA dry mass, ton	2.5	6.0	39
Discarded mass, ton	0	6.6	0
Packaged DLSRR mass, ton	1.0	5.5	15
RLA dimensions			
RLA diameter, m	1.2	2	3.7
RLA length without DLSRR, m	7.5	12	42
Main engine parameters			
Rocket burning time, s	60	90	120
Combustion chamber pressure, atm	100	70	97
Nozzle expansion ratio	16	12	16
Fuel	solid	EEC	kerosene
Oxidizer	solid	HP95	LO2
Combustion chamber temperature, ^{o}C	1,800	1,800	2,050
Sea level exhaust velocity, m/s	2,040	2,100	2,350
Vacuum exhaust velocity, m/s	2,285	2,380	2,670
Average exhaust velocity, m/s	2,177	2,277	2,592
DLSRR release parameters			
DLSRR release altitude, km	40	50	100
DLSRR release velocity, m/s	1,202	1,380	2,355
Sortie parameters			
Sortie length, min:s	6:25	8:02	7:38
Flight ceiling, km	114	147	198
Ballistic trajectory time, min:s	4:16	5:00	4:24

6. Conclusions

In this work we demonstrate the feasibility of Rocket Launcher Aircraft – a new type of military aircraft which carries Drone Launches Short Range Rockets to a high altitude, supplies them with high initial velocity, and fires them. The system consisting of RLA and DLSRR delivers payload to the target at a fraction of cost of the state of the art short range rockets. Unlike the state of art fighters and bombers, the RLA itself never approaches the target. RLAs can perform 2-4 sorties per hour and 30-50 sorties per day.

RLAs would have much greater variety than fighters, bombers, or any other class of aircraft. RLAs would have liftoff masses from 1 ton to over 1,000 tons. RLA would be powered by all kinds of solid, liquid, and hybrid rockets. Each particular RLA would be capable of both vertical and forward

inclined sorties. In this work we have considered vertical sorties only.

Light RLA is fuelled by solid propellant. It would have liftoff mass of 10 tons. It would release DLSRRs at an altitude of 40 km with initial velocity of 1,200 m/s up. The packaged DLSRRs would have a mass of 1 ton. Medium RLA is fuelled by liquid propellant. It would have liftoff mass of 55 tons. This RLA would have a discarding propellant tank. Medium RLA would release DLSRRs at an altitude of 50 km with initial velocity of 1,380 m/s up. The packaged DLSRRs would have a mass of 5.5 tons. Heavy RLA is a liquid propellant rocket modeled after Falcon 9 first stage. It would have liftoff mass of 390 tons. It would release DLSRRs at an altitude of 100 km with initial velocity of 2,355 m/s up. The packaged DLSRRs would have a mass of 15 tons. We are looking forward to significant role of RLAs within future military.

Appendix

Appendix I: List of Abbreviations – General

ATACMS	army tactical missile system (USA)
GMLRS	guided multiple launch rocket system (USA)
RASAero	aerodynamic analysis and flight simulation software
RLA	rocket launcher aircraft
RPA	rocket propulsion analysis
DLSRR	drone launched short range rocket

Appendix II: List of Abbreviations – Chemicals

AN	ammonium nitrate
EEC	the mixture of 61% monoethanolamine, 30% ethanol, and 9% hydrated copper nitrate
ETA	the mixture of ethanolamine and 10% $CuCl_2$
ETAFA	the mixture of 47.5% Ethanolamine, 47.5% Furfuryl Alcohol, and 5.0% $CuCl_2$
HP95	95% hydrogen peroxide
LO2	liquid oxygen
NC	nitrocellulose
NG	nitroglycerine
PSAN	phase stabilised ammonium nitrate
RFNA	red fuming nitric acid – 79% HNO_3 and 19% N_2O_4

Conflicts of Interest

The author declares that he has no competing interests.

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