The Concept of an Inflatable Reusable Launch Vehicle

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Abstract

This article discusses the concept of a launch vehicle with an inflatable reusable first stage. The first stage in the form of an inflatable streamlined thin-walled tank with a low bulk density (which may be less than the standard sea-level density of air) is proposed. Compressed light gases (hydrogen, helium) ensure the rigidity of the inflatable thin-walled tank. For return to Earth, the first stage is slows down in the upper atmosphere. In the lower atmosphere, the aerostatic lifting force exceeds the Earth's gravity. Thus, the first stage will float in the air (this will prevents its destruction). Then the first stage returns to the spaceport for reuse. The inflatable first stage of a launch vehicle will be very large. Therefore, the main purpose of this article is to study the influence of aerodynamic losses on the payload. The mass of the inflatable first stage and its vertical and horizontal stiffness will also be calculate (as a first approximation). These are the main questions that need to be answer for the opportunity further developed the concept. Therefore, this article should be consider as a preliminary study, not claiming to be complete.

Keywords

Inflatable; Reusable; Launch Vehicle; Propellant

1 Introduction

The main reason for the high cost of space flights is the single use of flying machines[1]. After the acceleration of the spacecraft, the launch vehicle (which is a technical masterpiece worth millions or even tens of millions of US dollars) falls to Earth, repeating the fate of the legendary Icarus. Therefore, the need to create reusable space transport systems is obvious and such projects arouse of great interest. Currently, there are several basic methods for returning flying machines to Earth.

The first return method is the vertical landing of the launch vehicles using their own engines. Projects of such launch vehicles were develop in the past (for example, Delta Clipper) (SDIO, 1993). The New Shepard project (Spaceflight101, 2017) is develop currently. The Falcon 9 project (SpaceX, 2021) is on the stage of commercial use. This method of returning launch vehicles has good prospects for commercial application. The disadvantages include the high consumption of propellant during landing. It leads to decrease in payload, and the additional consumption of the resource of rocket engines.

The second return method of the spacecraft's (or parts thereof) is to use parachutes. Parachutes are widely used in aviation and space technology. They repeatedly was use for landing probes on Venus, Mars, Jupiter, Titan (the moon of Jupiter) (Burke, 2014). Parachutes was use for soft landing (splashdown) Space Shuttle Solid Rocket Boosters (NASA, 1988, NASA, 2008). One of the most recent examples the use of this method in astronautics is return of the reusable Dragon spacecraft (Garcia, 2018). In most cases, the disadvantages of this method include the need to equip the spacecraft's simultaneously with multiple systems: thermal protection, parachute system and soft landing system.

The third return method is the horizontal (airplane) landing of flying machines (or parts thereof). The most famous are Space Shuttle (Launius and Jenkins, 2002, Fernholz, 2018) and Buran (Kogut, 2018, Afanasyev, 2018). Also known projects: MAKS (Plokhikh at al., 2012), XCOR Lynx (Messier, 2018), NASP (Rockwell X-30) (Heppenheimer, 2009, Pattillo, 2001), HOTOL (Postlethwaite, 1989), RLV/AVATAR (Martin at al., 2009), RLV-TD (Mohan at al., 2017), Skylon (Hyslop at al., 2019, Arefyev at al., 2019), Boeing X-43 (Hanlon, 2004), Cold (Rudakov at al., 1999), Dream Chaser (Federal Aviation Administration, 2018), SHEFEX (Longo at al., 2006), SpaceLiner (Lariviere and Kezirian, 2019) and others. The disadvantages include the need to equip spacecraft's with wings, empennage and landing gear (which increases aerodynamic losses, complexity and mass of the construction).

All of these methods can be use individually or together (in various combinations). However, their application did not lead to a multiple decrease in the

[1] If buy a new Red Tesla car for every trip to the store for groceries, then transportation costs can become comparable to the costs of delivery to low Earth orbit.

cost of spaceflights. Therefore, there is a need to develop other methods of return (for example, using aerostatic force). For this purpose, we can use an inflatable balloon folded inside a special container. If this balloon is made large enough (and filled with light gas), it can keep the spacecraft in the air. The main disadvantage is the long deployment time of the inflatable balloon. For example, during the VEGA Venus balloon experiment, a balloon with a diameter ≈ 3.4 m deployed in ≈ 100 s (Sagdeev et al., 1986). As the size increases, the filling time of the balloon will increase, which complicates their use.

To solve the problem of balloon deployment (at limited time), the concept of an inflatable launch vehicle will be consider. It is assume that the hydrogen balloon will be permanently deploy and is part of the launch vehicle. There are various design for multistage launch vehicles. As an example, the tandem staging design (with a reusable first stage) will be consider. First stage as a streamlined thin-walled cylindrical tank divided into three sections. These sections store separately: liquid methane, liquid oxygen and hydrogen gas (Fig. 1). Inside the sections is create an increased pressure (several atmospheres) to maintain a stable shape. The size of the first stage[2] is calculate so that when it returns to Earth, its bulk density is less than that of atmospheric air. Due to the positive buoyancy, the first stage (without propellant) will be able to float in the Earth's atmosphere[3]. Then the first stage will move to the spaceport (using its own rocket engine or special tugboats).

Figure 1. Schematic diagram of an inflatable launch vehicle (in the diagram, the exact shape and proportions are not respected): 1 – nasal section of an inflatable launch vehicle (it includes the second stage, third stage and payload); 2 – upper part of the first stage (for hydrogen gas); 3 – middle section (for liquid methane); 4 – aft section (for liquid oxygen); 5 – rocket engines

For the made of the inflatable first stage, it is propose to use a super-strong fabric (for example, from graphene filaments). After cutting out need to measure, the elasticity coefficient of each piece of fabric. Based on these measurements, the optimum stitching (or gluing) line of these pieces will be calculated. A thin gas-tight film will be glue to the inner surface of the first stage. A special impregnation can use to seal the seams. After stitching the outer shell, into the first stage can pumped gas (air, etc.). After that, it is necessary to sew on internal partitions, install systems and mechanisms, etc. For made of inflatable first stage can used existing airship hangars.

2 Materials and Methods

This section considers methods for calculating indicators by which will be determined the main technical characteristics and parameters of the launch vehicle with an inflatable reusable first stage. The methods of calculating the aerodynamic losses, the mass of the propellant, the strength and mass of the body is considered.

2. 1 Comparative Calculation of Aerodynamic Losses

Suppose a launch vehicle moves upward at a speed v (at time t). The main factors determining the flight dynamics of the launch vehicle are its mass m, the thrust of rocket engines F_t, the force of gravity F_g, and the aerodynamic drag force F_d. Consider the changes in speed dv over the time interval dt. The thrust force F_t could provide a speed increase dv_h (in the absence of other forces). However, the aerodynamic drag force F_d will decrease the speed by dv_d, and the gravitational force F_g will reduce the speed by dv_g. With this in mind, the actual acceleration a will be:

$$a = \frac{dv}{dt} = \frac{dv_h - dv_d - dv_g}{dt} \tag{1}$$

The physical meaning of dv_h is the change in characteristic speed, dv_d is the aerodynamic loss, dv_g is the gravitational loss. Aerodynamic and gravitational losses depend from the many interrelated factors, and their determination is a difficult task. At the preliminary stage, it can assumed that the gravitational losses of an inflatable launch vehicle are approximately equal to the gravitational losses of classic launch vehicles. For aerodynamic losses, it is necessary to make an additional calculation (because of the large size of the inflatable launch vehicle). Suppose that a launch vehicle has mass m, and because of aerodynamic drag loses infinitesimal speed dv_d for infinitesimal period of time dt. With this in mind, the aerodynamic drag force F_d will be:

$$F_d = \frac{m dv_d}{dt} \tag{2}$$

[2] By means of hydrogen section.

[3] Upon return, the first stage will drift in the Earth's atmosphere in a vertical position, since its stern will be heavier (due to the weight of the rocket engines).

The aerodynamic drag force F_d depends from drag coefficient C_d, drag area S, air density ρ_a, speed v. Taking this into account, it is written:

$$F_d = \frac{C_d S \rho_a v^2}{2} \tag{3}$$

Using equations (2; 3), it is obtained:

$$dv_d = \frac{C_d S \rho_a v^2}{2} \frac{dt}{m} \tag{4}$$

For evaluate the aerodynamic losses of an inflatable launch vehicle, will be use a comparison method with classic launch vehicle (for example, Saturn V)[4]. Suppose two launch vehicles move along the same trajectory (and have the same speed v at each point of the trajectory). Suppose these launch vehicles have the same drag coefficient C_d (at the same time, they can differ in appearance and shape). Classic launch vehicle has a mass m_l, drag area S_l, aerodynamic loss $d(v_d)_l$ (for period of time dt). Inflatable launch vehicle has a mass m_g, drag area S_g, aerodynamic loss $d(v_d)_g$ (for period of time dt). With this in mind, using equation (4) it is obtained:

$$d(v_d)_g = d(v_d)_l \frac{S_g}{S_l} \frac{m_l}{m_g} \tag{5}$$

Total aerodynamic losses of a classic launch vehicle is $(v_d)_l$ and total aerodynamic losses of inflatable launch vehicle is $(v_d)_g$. Summing the infinitesimal increments in equation (5) on condition S_g/S_l = Const, $m_l/m_g = 1$ it is obtained:

$$\frac{(v_d)_g}{(v_d)_l} = \frac{S_g}{S_l} \tag{6}$$

2. 2 Mass and Volume of the Inflatable First Stage

Consider the first stage body in the form of a cylinder of length l, radius r, and wall thickness dl. Under the condition $l >> r$, the mass of the ends of the cylinder can be neglected (to a first approximation). The density of the first stage material is ρ_t. With this in mind, the mass of the first stage body m_t will be:

$$m_t = 2\pi r l dl \rho_t \tag{7}$$

The pressure inside the first stage is P (N/m^2). Suppose the longitudinal section of the first stage has the shape of a rectangle, on which (perpendicular to plane) the force F_t acts:

$$F_t = 2rlP \tag{8}$$

This force is distribute on two sides of the cylinder with a total length of 2l. Assume that tensile strength is Q. Using equation (8), the following equation it is obtained:

$$Q = \frac{2rlP}{2ldl} = \frac{r}{dl} P \tag{9}$$

Using equations (7; 9), it is obtained:

$$m_t = 2\pi r l \frac{r}{Q} P \rho_t \tag{10}$$

After returning to Earth, the first stage contains inside light gases (hydrogen, helium). The density of atmospheric air is ρ_a, dry mass of the first stage is m_d (without propellant). Can be record the volume V necessary for the first stage to be lighter than the displaced air volume:

$$V = \frac{m_d}{\rho_a} \tag{11}$$

Using equation (11), the mass m_p of the propellant (at the start) will be:

$$m_p = V \rho_p = m_d \frac{\rho_p}{\rho_a} \tag{12}$$

2. 3 Bending Moment Calculation

For a beam, the bending moment M depends on length l and force F. Taking this into account, it is written:

$$M = \frac{Fl}{4} \tag{13}$$

For inflatable cylinder, the collapse bending moment M_c depends of internal pressure P and radius r (Wielsgosz and Thomas, 2002). With this in mind, it is obtained:

$$M_c = \frac{\pi}{4} \pi P r^3 \tag{14}$$

3 Results

This section shows the results of calculations of the technical characteristics of the launch vehicle with an inflatable reusable first stage. The overall parameters calculated so that, upon returning to Earth, the first stage has positive buoyancy in the Earth's atmosphere. Since this leads to a significant increase in the size, aerodynamic losses will be calculate. The gas pressure inside the first stage is compare with the maximum dynamic pressure (in order to show that aerodynamic drag does not disrupt the shape of the first stage body). It is show that the inflatable first stage has a sufficient stability in the longitudinal and transverse directions.

[4] The inflatable launch vehicle will be large size, this and determines the choice of the Saturn V (as the largest launch vehicle).

3. 1 Key Specifications of Inflatable Launch Vehicles

As a sample for comparison will be, use the Saturn V launch vehicle (as the largest of all existing launch vehicles). Saturn V has the following characteristics: starting mass 2728.5 t and thrust of first-stage engines ≈ $3.3 \cdot 10^7$ N; diameter 10.1 m and drag area S_l ≈ 80 m^2 [24; 25; 26; 27]. Suppose the inflatable launch vehicle has a similar starting mass, engine thrust). With this in mind the starting mass of the inflatable launch vehicle is 2728.5 t, engines thrust is ≈ $3.3 \cdot 10^7$ N. At the first stage of the inflatable launch vehicle, can be use SpaceX Raptor engines powered by a methane-oxygen propellant (Spaceflight101, 2021).

The inflatable first stage of the launch vehicle consists of three sections. The lower section is file with liquid oxygen. The middle section is file with liquid methane. The upper section is file with hydrogen gas. Suppose the pressure inside the sections is 3 atm. Helium is use to pressurize the lower section, and hydrogen is used for the middle and upper sections. As will be shown below, the «dry» mass of first stage is 30 t (without propellant). The air density is ρ_a = 1.2 kg/m^3. Using the equation (11) it is obtained the volume of first stage V_g = $2.5 \cdot 10^5$ m^3. Supposing, then its length is ≈ 157 m, diameter is ≈ 14.25 m, drag area S_g ≈ 159.5 m^2.

Suppose that the inflatable launch vehicle and the Saturn V have the same drag coefficients (in the first approximation). In addition, two launch vehicles have the same masses and the same flight dynamics (in the first approximation). This allows comparisons the aerodynamic losses of the two launch vehicles (by comparing their drag areas). Since $S_g/S_l \approx 2$, using equation (6) it is obtained $(v_d)_g \approx 2\ (v_d)_l$. Aerodynamic losses of the Saturn V $(v_d)_l$ ≈ 46 m/s (Shuneiko, 1973, Bilstein, 1996, NASA, 1968, Apollo facts, 2021). Aerodynamic losses of the inflatable launch vehicle $(v_d)_g$ ≈ 92 m/s.

For the first stage of the Saturn V the characteristic speed $(v_h)_l$ ≈ 3660 m/s (Shuneiko, 1973, Bilstein, 1996, NASA, 1968, Apollo facts, 2021), and for the first stage of the inflatable launch vehicle the characteristic speed $(v_h)_g$ = 3706 m/s (3660 m/s – 46 m/s + 92 m/s). However, the actual maximum speed is 2300 m/s at an altitude of 67 km. After that, the first stage is separate (and the rest of the launch vehicle continues to fly). The first stage continues to move up, then falls down (braking occurs due to aerodynamic resistance). In the proposed concept, the bulk density of the first stage (without propellant) is less than the density of air near the Earth's surface. Therefore, the aerostatic lift exceeds the gravitational force of attraction, and the first stage will be able to float in the air over the Earth's surface.

Thus for the first stage of the inflatable launch vehicle the characteristic speed $(v_h)_g$ = 3706 m/s and the specific impulse of engines I_{sp} = 334 s (SpaceX Raptor). Using the Tsiolkovsky equation it is obtained the propellant mass m_p ≈ 1848 t. The mass of liquid methane is 385 t, the mass of liquid oxygen is 1463 t (ratio 1:3.8) (Spaceflight101, 2021). The volume of the aft section is ≈ 1282 m^3, the volume of the middle section is ≈ 928 m^3, and the volume of the upper section is ≈ 22790 m^3. At the start, the mass of gaseous hydrogen inside the middle and upper sections is ≈ 6 t. During the flight, part of the hydrogen is pump into the middle section to maintain pressure (and the other part ≈ 4 t will be release into the atmosphere)[5]. At the finish, the pressure will decrease to ≈ 1 atmosphere (≈ 2 t of hydrogen will remain in the middle and upper sections). Helium is not emitted into the atmosphere and the pressure in the aft section will remain 3 atmospheres (the mass of helium is 0.7 t).

For Saturn V, the total mass of the second and third stages (including the payload) is 438.5 t and payload is 140 t. Therefore, the payload ratio is 0.32 (140 t / 438.5 t). This coefficient can use to determine the payload of an inflatable launch vehicle (as a first approximation). Suppose that two launch vehicles use the same propellant and engines (for second and third stages). For an inflatable launch vehicle, the total mass of the second and third stages (including the payload) will be 846.5 t (2728.5 t - 30 t - 4 t - 1848 t). Considering this, the payload of inflatable launch vehicle will be 271 t (846.5 t · 0.32).

Taking into account the drag area S_g ≈ 159.5 m^2 and internal pressure of 3 atmospheres, a tensile force is ≈ $4.9 \cdot 10^7$ N (along the longitudinal axis of first stage). This force is ≈ 6 times greater than the weight of the upper part of the launch vehicle (and almost 2 times the total starting weight of the launch vehicle). It is will ensure rigidity and stability of the inflatable launch vehicle. It will show below that there is a multiple reserve for increasing the internal pressure due to the increase in the mass and strength of the first stage body. However, an internal pressure of 3 atmospheres is sufficient (and the need to increase it seems unlikely).

It is supposed the first stage to be made of graphene, which has a tensile strength of Q = 130 GPa (Lee, 2008). With a thickness of one layer of graphene $0.34 \cdot 10^{-9}$ m, and a surface density of $0.77 \cdot 10^{-6}$ kg/m^2, his bulk density is ≈ 2265 kg/m^3. Using equation (10), it is body mass m_t ≈ 1.3 t (with a 5-fold safety factor). With a body surface area of ≈ $7 \cdot 10^3$ m^2, the average wall thickness of the body dl ≈ 0.08 mm. If divide the longitudinal tensile force ≈ $4.9 \cdot 10^7$ H into the cross section of the body wall ≈ 0.0037 m^2, it is obtained the tension stress ≈ 13 GPa. In the longitudinal direction, the first stage will have 10-fold margin of safety.

[5] Because of aerodynamic heating, the pressure rises and part of the hydrogen gas must be release into the atmosphere. If necessary, excess hydrogen can use for cooling of rocket engines. This will solve the problem of aerodynamic heating of rocket engines (since the first stage will move stern first aft when falling).

At the first stage, can be use SpaceX Raptor engines with a thrust-to-weight ratio ≈ 200 (Spaceflight101, 2021). With a thrust of ≈ 3.3 10^7 N, the total mass of the engines will be ≈ 17 t. When returned to Earth, about ≈ 2 t of hydrogen remain inside the first stage. The total mass of the main components of the first stage is ≈ 30 t (body 1.3 t + engines 17 t + hydrogen 2 t + helium 0.7 t + mass reserve 9 t). The mass reserve is intended for solve various problems (which may arise at creating an inflatable first stage). For example, if necessary, can be increase the strength and inside pressure of the inflatable first stage in several times.

Using graphene gives the best results. However for the manufacture of the inflatable first stage, not only graphene can be use, but also less strong materials (for example, Kevlar with a tensile strength of ≈ 3.62 GPa, density ≈ 1440 kg/m^3) (Quintanilla, 1999). Let us say the calculated tensile strength of the first stage is ≈ 3 GPa. Then according to the equation (10), the mass of the first stage body is 7.8 t (or increase by 6.5 t). Since the mass reserve is 9 t, the replacing graphene with Kevlar will not significantly affect the technical characteristics of the launch vehicle with an inflatable first stage.

3. 2 Bending Moment, Dynamic Pressure and Aerodynamic Heating

The Saturn V launch vehicle flight evaluation reports show that the maximum bending moment achieved during the Apollo 14 mission (January 31, 1971). At 76 seconds of flight (at 21:04:23) its value was 116,000,000 lbf·in (NASA, 1971) or 13.1 million N·m. The main factor affecting the magnitude of the bending moment is wind (Johnson and Vaughan, 2020). Let us assume that a launch vehicle with an inflatable first stage will be use in similar conditions (taking into account its larger size). Then using the equation (13) the maximum bending moment can be estimated at M ≈ 50 million N·m (in the first approximation). Using equation (14) the collapse bending moment it is obtained M_c = 271 million N·m. Thus, the inflatable first stage has a 5-fold reserve of stiffness. In addition, the inflatable first stage can restore its original shape (when the external load decreasing). This is one of the advantages of an inflatable structure (for a classic launch vehicle, a significant change in shape is irreversible).

The dynamic pressure on the Saturn V (the Apollo 14 mission) reached its maximum value 655.80 lb/ft^2 (NASA, 1971) or 31.4 kPa. Thus, the internal pressure (≈ 303 kPa) of the inflatable first stage almost an order of magnitude exceeds external dynamic pressure. This will ensure the maintenance of the shape during the flight through the Earth’s atmosphere.

Consider the aerodynamic heating of the first stage at returning to Earth. During braking, the first stage will move stern first (since the stern is heavier). The hot air flow will not be able to local overheating of the first stage body (because of its extremely high thermal conductivity). Suppose that the speed is ≈ 2300 m/s, the accommodation coefficient is ≈ 0.25 (for stern), the mass is ≈ 30 t. During aerodynamic braking, the first stage will receive a thermal energy of ≈ 2·10^{10} J. For cooling, can be use a tank with liquid hydrogen. Let us assume that the hydrogen will heated to temperature of 373 K. The specific heat of vaporization and heating of hydrogen will be ≈ 5.1 MJ/kg. Therefore, ≈ 4 t of liquid hydrogen will be needed (this is the upper limit, since the thermal radiation of the surface and the total heat capacity of the first stage are not taken into account). One can see that the mass reserve (9 t) makes it possible to store liquid hydrogen. The heated hydrogen will be release into the atmosphere.

4 Discussion

Let us compare three methods returning of the first stage: free fall (with complete destruction), aerostatic returning, and vertical landing (with the help of engines). For aerostatic returning of the first stage, the payload is 271 t (Section 3.1.). Let us say we abandon the use of aerostatic force and remove the hydrogen section. For a decrease in size, aerodynamic losses will decrease by about 2 times (to the level of Saturn V). The starting mass of the launch vehicle is 2728.5 t, the "dry" mass of the first stage is 30 t, and the characteristic speed of the first stage is 3660 m/s. With a specific impulse of 334 s the total mass of the second and third stages (with a payload) will be 863 t. Using the payload ratio (from Section 3. 1.) it is obtained the payload is 276 t (863 t · 0.32). However, the situation will change with a vertical landing. It is known that the payload mass (LEO) of Falcon 9 is 22.8 t (without first stage return) and 15.6 t (with first stage return) (Sesnic, 2021). With an approximate extrapolation of these data, the payload will decrease from 276 t to 189 t. The data obtained can be present in the form of a Table 1.

Table 1. The payload mass of the launch vehicles with different methods of returning the first stage

First stage return method	Payload mass, t
Free fall of the first stage	276
Aerostatic returning of the first stage	271
Vertical landing of the first stage	189

Table 1 shows that the maximum payload (276 t) is achieved in the case of the free fall of the first stage (however, the first stage will be destroyed). With aerostatic returning of the first stage, the payload will decrease by 5 t (or ≈ 2%). With a vertical landing, the payload will decrease by 87 t (or ≈ 32%). Thus, the aerostatic returning will keep the first stage for reuse, at the expense of an insignificant decrease in payload mass. Therefore, aerostatic returning looks more promising that vertical landing or free fall.

5 Conclusion

The article discusses the concept of a launch vehicle with an inflatable reusable first stage. The starting mass of launch vehicle is 2728.5 t, payload capacity 271 t (to LEO). The payload capacity of a launch vehicle with an inflatable reusable first stage (≈ 271 t) can significantly exceed the payload capacity of the Saturn V (≈ 140 t). This is the result of the simple and lightweight design of the inflatable first stage and high technical characteristics of the SpaceX Raptor engines.

The first stage (in the form of an inflatable streamlined cylindrical tank) may have the following characteristics: length 157 m, diameter 14.25 m, volume $2.5 \cdot 10^4$ m^3, propellant mass 1848 t, and «dry» mass 30 t. For its manufacture, it is supposed to use thin ultra-strong material (for example, graphene). The necessary rigidity of body supposed to obtain due to the high pressure of light gases inside the first stage. In the article shows, that such an inflatable structure can withstand the emerging during the flight vertical and horizontal loads (at an internal pressure of about 3 atmospheres). The first stage use a methane-oxygen propellant (the SpaceX Raptor engines). The second and third stages uses a hydrogen-oxygen propellant (the J-2 engines).

The main advantage of this concept is the ease of returning the first stage for reuse. During the flight, a first stage loses most of its mass, and its bulk density decreases to 1.2 kg/m^3 (i.e., it becomes less than the standard air density at sea level). Due to this, after returning the first stage will be able to floats in the Earth’s atmosphere (this will keep its in serviceable and working status). Thus, a first stage does not need in the sophisticated technical means of providing a soft landing. In addition, it has a number of other advantages: the lack of power frame, lightweight, high strength, the possibility of transportation by air etc. All this gives reason to consider the proposed concept as a promising direction for the development of space transportation technologies.

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