32nd Congress
of the International Council
of the Aeronautical Sciences
September 6-10, 2021
Pudong Shangri-La, Shanghai, China

AN ENERGY BASED WEIGHT ESTIMATION TECHNIQUE OF TWO-STAGE-TO-ORBIT REUSABLE LAUNCH VEHICLES

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Abstract

An energy based weight estimation method of Two-Stage-To-Orbit (TSTO) reusable launch vehicles (RLVs) based on the transportation capacity was proposed. A horizontal take-off / horizontal landing (HTHL) TSTO reusable launch vehicle concept was designed. With a payload of 2 tons and entering low earth orbit (LEO) as mission requirements, the weight of the concept vehicle were analyzed. The results show that increasing engine thrust, increasing engine thermal efficiency, and reducing aerodynamic drag can effectively reduce the structural weight of the vehicle. The use of liquid propellant is conducive to reducing the gross weight of the vehicle, while the use of solid propellant is conducive to reducing the length of the vehicle. In comparison, the choice of liquid propellant is more suitable for TSTO RLVs. The estimation technique proposed in this paper and the analysis results have laid an important base for the further development of the TSTO RLVs design.

Keywords: Two-stage-to-orbit (TSTO); Conceptual design; Energy Based method; Weight estimation

1. Introduction

In order to achieve the requirements for low-cost, high-reliability space access, the reusable launch vehicles (RLVs) has become a hot spot in countries around the world. The United States, Russia and Japan have carried out a lot of research work since the 1980s[1, 2], and proposed various concept schemes for Single-Stage-To-Orbit (SSTO)[3] and Two-Stage-To-Orbit (TSTO)[4, 5] respectively. The TSTO concept allows that have completed work to be separated from the flight system and returned to the ground, significantly reducing propellant demand and reducing launch costs. It is more suitable for space transportation than the SSTO concept under the state of the art. However, the TSTO RLVs need to consider aerodynamics, thermal protection, structure, propulsion, trajectory, control and many other systems, and cannot apply the mass estimation method of traditional rocket or aviation aircraft directly, getting a closed-loop solution at the concept design phase is a very difficult challenge[6, 7].

At present, a large number of TSTO concepts had been published, which are all limited to the discussion of part characteristics such as configuration, power, and trajectory, and have not yet formed a complete design scheme. And there is no TSTO RLVs that achieved real flight for reference. The researches of the TSTO concept scheme mainly focuses on the whole system mass estimation and aerodynamic layout design. The researches on mass estimation of TSTO RLVs is mainly divided into two categories. One is to obtain the relationship between the size and mass of the aircraft by using the statistical analysis of the overall parameters of the aircraft[8]. This type of research focuses on mass estimation and does not consider whether the aircraft has the capability of getting into orbit. The other is to analyze the propellant consumption rate based on the assumed structural mass coefficient and a certain payload demand through the energy conservation relationship, so as to analyze the weight of aircraft. This type of research does not know the size information of the aircraft, that is, it is impossible to know whether the effective volume of the aircraft can be loaded with the required propellant. There are also some researches[9, 10] that combined the above two types. Starting from the ground take-off state, the relationship between the orbital capability and the mass is obtained by the layout parameters such as the wet areas to verify the feasibility of the weight and size of TSTO RLVs.

In this paper, a reversal analysis method of aircraft structure weight based on the transportation capacity and potential energy of the TSTO RLVs is proposed in consideration of the condition and performance of different engine modal and the characteristics of engine thrust and aerodynamic drag. Then, based on a hypothetical TSTO aircraft configuration with taking 2 tons payload into the low earth orbit (LEO) as the mission requirements, the weight of the aircraft were analyzed. The influence of propellant parameters, engine performance and flight profile on the size of the aircraft is analyzed. This method provides a theoretical basis for the preliminary design of the TSTO RLVs, and also helps to further optimize the overall design of the vehicle.

2. Methodology

The weight estimation method is shown in Figure 1. Firstly, determine the payload mass and payload volume of the mission. Then, according to the requirements of the orbit and the flight profile, the relationship between the parameters of the whole TSTO RLVs configuration, propellant and power is established starting from the structural weight of upper stage. Finally, an energy based weight estimation technique of TSTO RLVs with practical reference significance is formed.

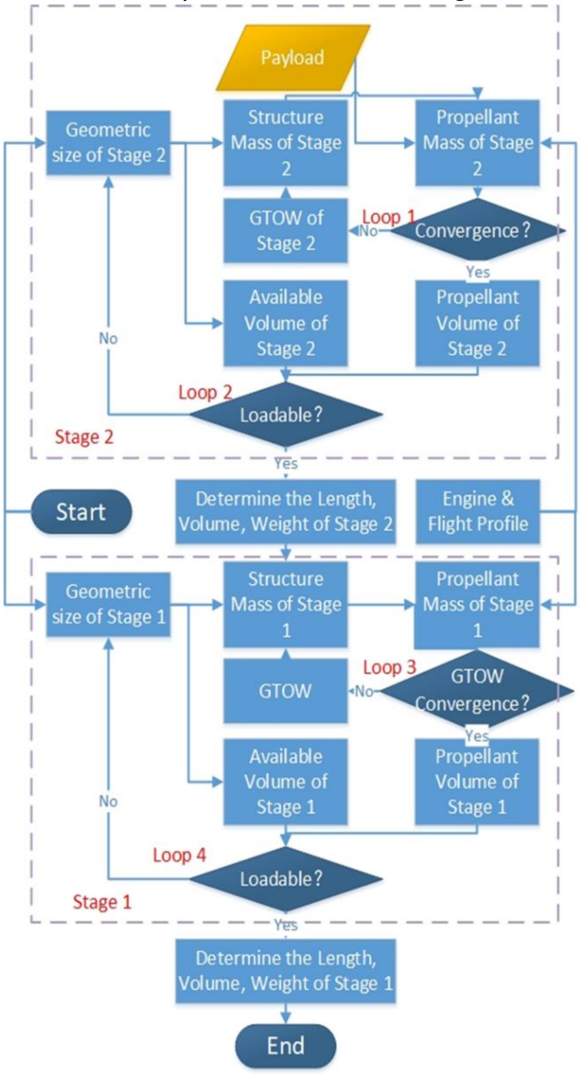

Figure 1 – Flow chart of aircraft weight estimation method.

The implementation steps are as follows:
(1) Based on the size-based structural weight estimation method, the second stage structural weight of the aircraft is obtained, and the weight of the propellant required for the second stage is analyzed including the orbital payload.
(2) Based on the structural effective volume known from the structural size, analyze whether the volume is sufficient to carry the second stage propellant and the orbit payload. Then, a loop iteration is performed such that the effective volume of the second stage is equal to the sum of the propellant volume and the payload volume. Thereby, the second stage aircraft size and weight are obtained.
(3) Similar to step (1), the structural weight of the first stage of the aircraft is obtained using a size-based structural weight estimation method. The weight of the propellant required for the first stage is analyzed on the basis of considering the gross weight of the second stage as the payload of the first stage.
(4) Similar to step (2), based on the structural effective volume known from the structural size, analyze whether the volume is sufficient to carry the first stage propellant. Then, a loop iteration is performed such that the effective volume of the first stage is equal to the propellant volume. Thereby, the first stage aircraft size and weight are obtained.
Typically, the first stage structural weight estimation process includes the take-off gross weight (GTOW) parameter, and the second stage structural weight estimation process includes the landing weight parameter. Therefore, the iterative analysis is also required in the weight estimation of the propellant in step (1) and (3) to obtain accurate results.
Next, the structural weight estimation method and the propellant weight estimation method will be described in detail.

2.1 Structural Weight Estimation Method

Structural weight estimation is a very important but relatively weak link in the concept design stage[7]. At present, almost all aircraft design departments have derived their own weight evaluation formulas based on the overall parameter data of their developed aircraft using mature regression analysis methods. According to the structural characteristics of TSTO two-stage aircraft, combined with engineering practice, this paper forms the following calculation method of structural weight estimation.

2.1.1 Structural Weight of the First Stage

Considering that the first stage of the TSTO Vehicle needs to reach a certain flying height and speed while carrying the second stage. Referring to the research results in reference [7], the main structural weight estimation formulas including the fuselage, wing, tail and landing gear:

$$W_{Body} = 0.383612 M_{TO}^{0.35} N_z^{0.25} L^{0.5} D_{body}^{0.849} W^{0.685} \tag{1}$$

$$W_{Wing} = 0.007145 \left(M_{TO} N_z\right)^{0.5} S_w^{0.622} A^{0.785} \left(t/c\right)_{root}^{-0.4} \left(1+\lambda\right)^{0.05} \left(\cos\Lambda\right)^{-1} S_{scw}^{0.04} \tag{2}$$

$$W_{Tail} = 0.452 \left(M_{TO} N_z\right)^{0.5} S_{vert}^{0.718} Ma^{0.34} b_{vert}^{-1} \left(1+S_{rud}/S_{vert}\right)^{0.348} AR_{vert} \left(1+R_{vert}\right)^{0.25} \left(\cos\Lambda_{vert}\right)^{-0.323} \tag{3}$$

$$M_{\lg} = 62.21 \left(M_{TO} \times 10^{-3}\right)^{0.84} \tag{4}$$

Where, M_{body} is the mass of body; M_{wing} is the mass of wing; M_{tail} is the mass of vertical tail; M_{tps} is the mass of thermal protect structure; M_{lg} is the Mass of landing gear; M_{TO} is the takeoff weight; N_z is the ultimate load factor; L is the Body length; D_{body} is the Body height; W is the Width of body; S_w is the planform area of trapezoidal wing; A is the aspect ratio; $(t/c)_{root}$ is the thickness to chord ratio at the wing root; λ is the tip ratio of wing; Λ is the wing sweep angle at 25% MAC; S_{scw} is the planform of wing mounted control surfaces; S_{vert} is the planform area of vertical tail; Ma is the Mach number; b_{vert} is the span of tail or tip fins; S_{rud} is the planform area of rudder; AR_{vert} is the aspect ratio of vertical tail; R_{vert} is the taper ratio of vertical tail; Λ_{vert} is the Tail wing sweep angle at 25% MAC.

2.1.2 Structural Weight of the Second Stage

At present, a variety of reusable space vehicles have been developed in the world, and the method of structural weight estimation of each part of the vehicle has been formed based on the statistical data of the overall parameters. In this paper, referring to the research results in reference [11] and [12], the main structural weight estimation formulas including the fuselage, wing, tail, and thermal protection system (TPS):

$$M_{body} = 2.167 A_{body}^{1.075} \tag{5}$$

$$M_{wing} = N_z M_{land} \left[\frac{1}{1+\delta\left(\frac{S_{body}}{S_{\exp}}\right)} \right]^{0.386} \left(\frac{S_{\exp}}{t_{root}}\right)^{0.572} \left[K_{wing} b_{str}^{0.572} + K_{ct} b_{body}^{0.572} \right] \tag{6}$$

$$M_{tail} = 26.06 \left(S_{vert} \left(t/c \right)_{vert}^{0.244} b_{vert}^{0.0364} \right)^{0.8674} \tag{7}$$

$$M_{tps} = 1.366 A_{body} + 2.845\left(2S_{\exp}\right) + 1.572\left(2S_{vert}\right) + 3.468\left(2S_{bf}\right) + 0.508 A_{body} \tag{8}$$

Where, M_{tps} is the mass of thermal protect structure; A_{body} is the surface area of the body; M_{land} is the Landed mass of vehicle; δ is the wing efficiency factor; S_{body} is the planform area of body; S_{exp} is the exposed wing planform area; t_{root} is the Wing thickness at root; b_{str} is the wing structural span along the half chord line; b_{body} is the maximum width of the body; K_{wing} is the exposed wing configuration constants; K_{ct} is the wing carry-thru constants; b_{vert} is the span of tail or tip fins; S_{bf} is the Planform area of body flap.

2.2 Propellant Weight Estimation Method

No matter what type of power is used, the relationship between the acceleration process and the mass of propellant consumed can be established by Newton's second law. That is, the equation of acceleration of the aircraft can be established by using the engine specific impulse to represent the thrust:

$$m\frac{dV}{dt} + mg\sin\gamma = -gI_{sp}\left(1-\frac{D}{T}\right)\frac{dm}{dt} \tag{9}$$

Where, m is the mass of the aircraft; V is the speed of the aircraft; t is time; g is the acceleration of gravity; I_{sp} is the specific impulse of engines; D is the drag of the aircraft; T is the thrust of the engine. For rocket-powered aircraft, the push-to-drag ratio of this type of aircraft is extremely large, and the I_{sp} of the rocket engine is approximately constant during flight, so that the ratio of the mass during flight to the initial mass is

$$\mu = \exp\left(\frac{\Delta V + \Delta V_g}{gI_{sp}}\right) \tag{10}$$

Where

$$\Delta V = V_2 - V_1, \quad \Delta V_g = 2g\left(H_2 - H_1\right)/\left(V_1 + V_2\right) \tag{11}$$

ΔV is the speed increment; ΔV_g is the potential energy increment; V_1 is the speed at the beginning of the flight; V_2 is the speed at the end of the flight; H_1 is the altitude at the beginning of the flight; H_2 is the altitude at the end of the flight. This equation will be changed to the Tsiolkovsky rocket equation without considering the potential energy increment.

For airbreathing-powered aircraft, the thrust-to-drag ratio of the aircraft is relatively small, the drag loss is not negligible, and the I_{sp} is no longer constant. The equation (9) is no longer suitable. On this basis, this paper introduces the concept of engine power TV.

$$TV = -\eta q \frac{dm}{dt} \tag{12}$$

Where, η is the thermal efficiency of the engine; q is the heating value of the fuel. Substituting equation (12) into equation (9) and writing it as equation (10) has the form:

$$\mu = \exp\left(\frac{\Delta V + \Delta V_g}{g I_{eff}}\right) \tag{13}$$

Where

$$I_{eff} = \frac{2\eta q}{g\left(V_1 + V_2\right)}\left(1 - \frac{D}{T}\right) \tag{14}$$

I_{eff} can be understood as the equivalent specific impulse of the propellant in the airbreathing engine. The above-mentioned method for calculating the mass of the propellant can be used in different flight profiles, and only needs to be substituted into the speed and altitude parameters at different stages.

3. Hypothetical TSTO RLVs Configuration

The first stage of a TSTO RLVs is to fly in the atmosphere. Therefore, the air breathing combined engine, including turbine-based combined cycle (TBCC) and rocket-based combined cycle (RBCC) engines, or reusable rocket (RR) can be used. At the same time, the second stage can only use RR engine due to the requirement to complete the orbit flight. Considering the economy and volume of the propellant, the first and second stage engines are suitable for hydrocarbon propellants[13].

Drawing on the aerodynamic configuration of the space shuttle and X-37B of the United States, and considering the required fuel volume of the two stages, this paper proposes a TSTO aerodynamic configuration scheme, as shown in Figure 2. The second stage adopts the scheme of small aspect ratio swept wing with side strip and single vertical tail. The first stage uses a fuselage structure with the same slenderness ratio as the second stage to achieve high volume utilization. The wing of the first stage increases the wingspan and airfoil area based on aerodynamic performance requirements such as ground take-off weight and climb rate, and uses winglet wings to increase lift and improve lateral stability. As a preliminary conceptual concept design, the first stage engine is placed on the abdomen of the fuselage, and detailed design research has not been carried out.

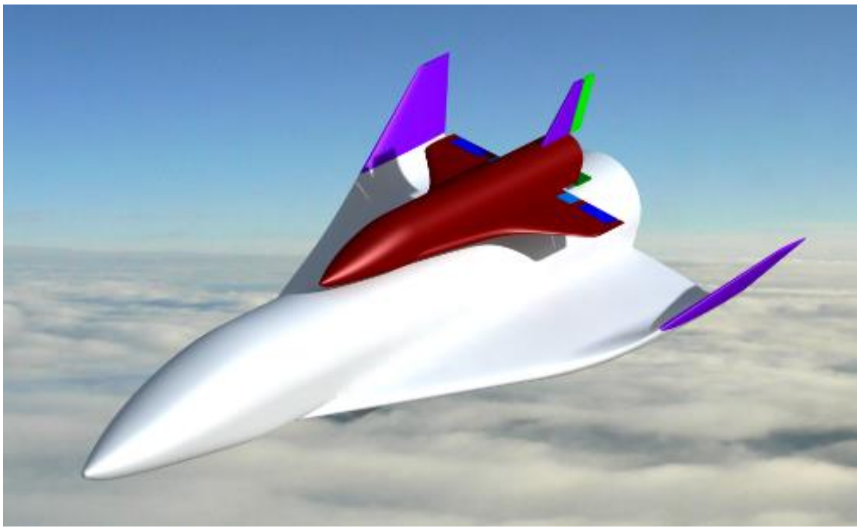

Figure 2 – Schematic diagram of two-stage-to-orbit vehicles.

According to the characteristics of the different propulsion system and the efficiency of the propellant, the analysis of this paper is based on the stage separation state of 30 km and Mach 6, and the injection state of 100 km and 7450 m/s. For an aircraft with a maximum flight speed of Mach 6, the TBCC propulsion system is relatively economical comparing with RBCC or RR. Considering the working speed range and working performance of the three working modes of TBCC, the working conditions and performance parameters of different engine modes are preliminarily determined as shown in Table 2. This paper mainly considers liquid hydrocarbon fuel as the propellant. But for comparison and analysis, the parameters of typical solid and liquid propellants are given here, as shown in Table 3.

Table 2 Working conditions and performance parameters of different engine modes

Modes	Velocity /Ma	Altitude /Km	Thermal efficiency /%	T/D
Turbojet	0~2.5	0~15.0	40	1.2
Ramjet	2.5~4.0	15.0~22.5	45	1.4
Scramjet	4.0~6.0	22.5~30.0	50	1.5

Table 3 Propellant parameters list

	I_{sp} /s	Density /Kg·m^{-3}	Calorific MJ/Kg
Liquid	350	810	42.8
Solid	280	1500	42.8

4. Results and Discussion

4.1 Relationship between Structural Mass and Size

As shown in Figure 3, it is the mass of stage 2 varies with the length of the fuselage. It can be seen from the figure that the mass tends to rise rapidly with the length of the fuselage. Comparing the mass data of the space shuttle and X-37B, the stage 2 aerodynamic configuration scheme designed in this paper is slightly smaller in mass over the same length, which is directly related to the more slender fuselage structure, smaller wingspan and smaller airfoil. This indicates that a comprehensive consideration needs to be made between the mass of the aircraft and the interior volume of the aircraft and the aerodynamic characteristics of the aircraft.

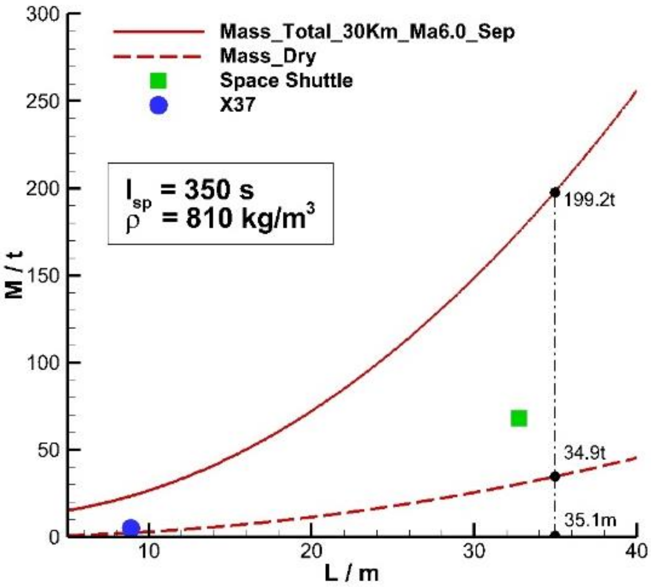

Figure 3 – Weight curve over fuselage length of stage two.

According to the calculation method established in this paper, the weight and size parameters for the second stage are determined. Including: fuselage length 35.1 meters, gross weight 199.2 tons, dry weight 34.9 tons, and propellant weight 162.4 tons. Based on this, the weight and size parameters for the first stage are also determined. As shown in Figure. 4, it is the weight curve over fuselage length of the first stage. It can be seen from the figure that, with the increase of the fuselage length, the structural mass of the aircraft is growing faster than the propellant mass. According to this calculation process, the final parameters of the first stage are: the fuselage length is 74.4 meters, and the total take-off weight is 759.6 tons.

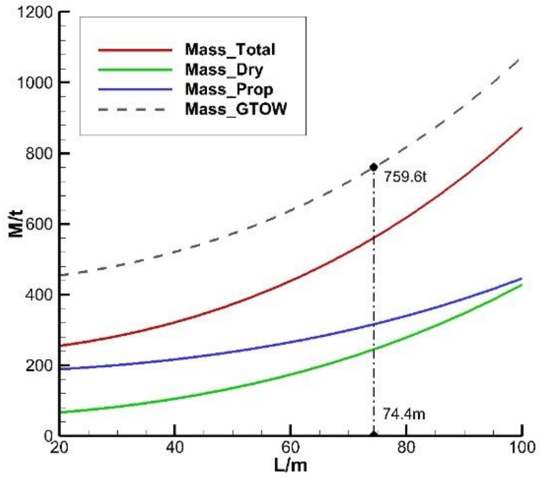

Figure 4 – Weight curve over body length of stage one.

4.2 Effect of Engine Efficiency on Structural Mass

Specific impulse (usually abbreviated I_{sp}) is a measure of the efficiency of rocket and jet engines. By definition, it is the impulse delivered per unit of propellant consumed, and is dimensionally equivalent to the thrust generated per unit propellant flow rate. For the rocket engine used in the second stage, this paper directly analyzes the influence of I_{sp} on the weight and size of the second stage of TSTO. Figure 5 and figure 6 show the influence of different I_{sp} on the weight and size of the second stage using the liquid and solid propellants with the parameters shown in table 3.

The results show that, Regardless of whether liquid fuel or solid fuel is used, the structural mass decreases rapidly with the increase of the I_{sp}.

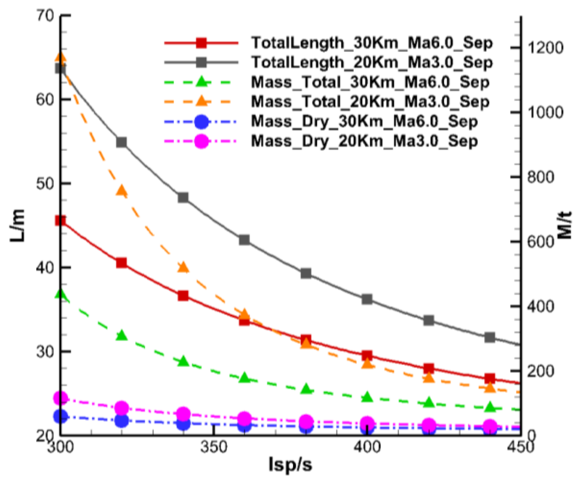

Figure 5 – Influence of different liquid fuel I_{sp} on stage two structure weight and size.

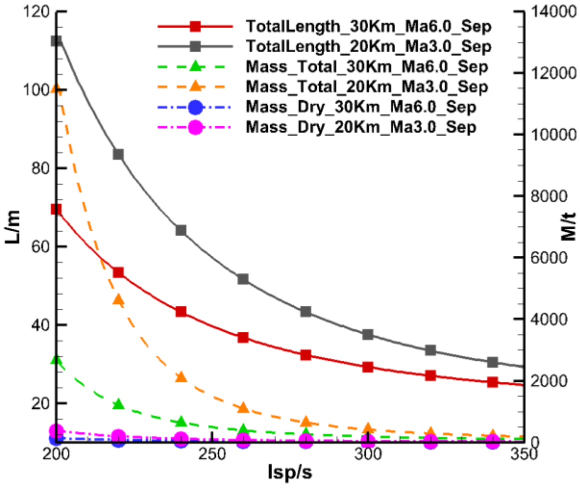

Figure 6 – Influence of different solid fuel Isp on stage two structure weight and size.

For the liquid propellant, the I_{sp} is increased by 10% from 350 seconds to 385 seconds, which can shorten the structure length by 12 %, from 35.1 meters to 30.9 meters, and reduce the dry weight of the structure by 22.6%, from 34.9 to 27.0 tons. For the solid propellant, the I_{sp} is increased by 10% from 280 seconds to 308 seconds, which can shorten the structure length by 12.4 %, from 32.3 meters to 28.3 meters, and reduce the dry weight of the structure by 23.6%, from 29.6 to 22.6 tons.

From this point of view, the reduction of the structure scale is faster than the increase of the specific thrust of the engine. On this view, the reduction of the structure weight and size is faster than the increase of the I_{sp}. Therefore, improving the I_{sp} of the engine is an effective means to reduce the weight and size of the second stage of TSTO.

At the same time, as shown in Table 4, the structural weight and size of the second stage under different fuel are calculated. The results shown that, structural size can be smaller, but the structural weight is significantly higher when using the solid propellant. Judging from the specific values in Table 4, when the length of the aircraft is not much different, the use of liquid propellants is more conducive to controlling the gross weight of the second stage, thereby effectively controlling the gross weight of the first stage. This is an important reason for choosing a liquid fuel propellant.

Table 4 Structural weight and size under different fuel

Propellant type	Length /m	Gross Weigh /t	Dry Weight /t	Propellant Weight /t
Liquid	35.1	199.2	34.9	162.4
Solid	32.3	260.4	29.6	228.8

In this paper the engine efficiency of TBCC is described by introducing the engine thermal efficiency parameters and fuel heating value parameters into the engine power. Based on this, the influence of the engine thermal efficiency on the first stage and overall size of the TSTO is analyzed when a TBCC engine is used in the first stage.

Table 5 shows the analysis results of the weight and size of TSTO when the thermal efficiency of the TBCC engine is increased or decreased by 5% based on the reference value. When the thermal efficiency of the engine increased by 5%, the fuselage length, total takeoff weight, dry weight, and propellant weight were reduced from 74.4 meters, 759.6 tons, 244.7 tons, and 315.6 tons to 67.9 meters, 639.5 tons, 200.1 tons, and 240.2 tons, respectively. The relative changes were -8.7%, -15.8%, -18.2%, and -23.9%. When the thermal efficiency of the engine reduced by 5%, the fuselage length, total takeoff weight, dry weight, and propellant weight were increased to 84.1 meters, 980.8

tons, 325.6 tons, and 456.0 tons, respectively, and the relative changes were 13.0%, 29.1%, 33.1% and 44.5%. It should be said that the thermal efficiency of the engine has a significant effect on the weight and size of the TSTO structure.

Table 5 Influence of engine thermal efficiency on structure weight and size

Thermal efficiency /%			Length /m	Gross Weight /t	Dry Weight /t	Propellant Weight /t
Turbojet	Ramjet	Scramjet				
35	40	45	84.1	980.8	325.6	456.0
40	45	50	74.4	759.6	244.7	315.6
45	50	55	67.9	639.5	200.1	240.2

Similarly, increasing the vehicle's thrust-to-drag ratio is also conducive to controlling the first stage and overall weight and size of the TSTO vehicle. Table 6 shows the effect of different push-to-drag ratios on the structure weight and size. It can be seen from the table that no matter whether the thrust of the engine is increased or the aerodynamic drag is reduced, as long as the thrust-to-drag ratio of the vehicle can be increased, the structure weight and size of the first stage can be greatly reduced.

In fact, the biggest constraint of the TSTO is the low matching degree between the propulsion capability of the engine and the aerodynamic drag and structural weight of the vehicle. In the case where it is difficult to greatly improve the propulsion capacity of the engine, it is of great significance to reduce the weight of the vehicle.

Table 6 Influence of engine thrust on structure weight and size

Thrust-to-drag Ratio T/D			Length /m	Gross Weight /t	Dry Weight /t	Propellant Weight /t
Turbojet	Ramjet	Scramjet				
1.1	1.3	1.4	142.4	3684.4	999.3	2485.9
1.2	1.4	1.5	80.6	730.2	211.8	292.2
1.3	1.5	1.6	65.0	489.9	137.4	153.3

5. Conclusion

In this paper, a reverse analysis method for the weight and size assessment of TSTO RLVs based on the transportation capacity was proposed. A horizontal take-off / horizontal landing (HTHL) TSTO reusable launch vehicle concept was designed. With a payload of 2 tons and entering low earth orbit (LEO) as mission requirements, the weight and size of the concept vehicle were evaluated. The analysis results show that engine efficiency has a significant influence on the overall size of the vehicle. Increasing engine thrust, increasing engine thermal efficiency, and reducing aerodynamic drag can effectively reduce the structural weight of the vehicle. The use of liquid fuel is conducive to reducing the gross weight of the vehicle, while the use of solid fuel is conducive to reducing the size of the vehicle. In comparison, the choice of liquid propellant is more suitable for TSTO RLVs. The estimation technique established in this paper and the analysis results have laid an important foundation for the further development of the TSTO RLVs design.

6. Acknowledgments

The authors would like to acknowledge the support provided for this research from the National Natural Science Foundation of China (numbers 11972359).

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8. Copyright Statement

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