Conceptual Study of a Rocket Dual Combustion Ramjet Combined-Cycle Engine for a Near Space Plane

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Abstract

In the present study, a Near Space Plane, powered by Multi-Module Rocket Dual Combustion Ramjet (RDCR) Combined-Cycle engine is presented. The operating modes of the engine include an ejector-jet mode, an ejector-ramjet mode, a DCR ramjet mode and a DCR scramjet mode. The plane will be dropped at about Mach 0.8 and 10km above sea level. Under the combination thrust of the gravity and the engine, it accelerates to about Mach 2 at 5-8km above sea level. Then it climbs up and accelerates the near space under the Ejector-ramjet mode or DCR ramjet mode. Ejector-ramjet mode provides higher acceleration with less efficiency, while DCR ramjet mode provides more efficiency and lower acceleration. Finally, the plane climbs up to about 26km, accelerates to Mach 6, and starts to cruise in DCR scramjet mode. To examine whether the concept would be capable of achieving, characteristics of the engine under different operating modes were studied analytically, ejector operation constraints are taken into account. A trajectory optimization was carried out to decide the engine and trajectory design parameters. The parameters effects analysis was carried out, the results indicates that there are no evidently significant effects for overall system performance, and the double parameters have relatively more significant effects.

1. Introduction

Near space is the region of Earth's atmosphere that lies between 20km and 100km above sea level. The area is of interest for military surveillance purposes and commercial interests for communications. Hypersonic vehicle is very suitable for flying in near space, and attracts great attention in recent years[1, 2].

In hypersonic flight condition, scramjet shows great performance and is prosperously investigated all around the world. However, scramjet cannot be self-started. Therefore, the combined-cycle engine has been studied. Several kinds of combined-cycle engines have been proposed[3, 4] and studied for a long time, one of which is a well-known rocket based combined-cycle engine (RBCC). The RBCC is composed of an ejector-jet mode, a ramjet mode, a scramjet mode, and a rocket mode. However, in the scramjet mode, it's very difficult for ignition and combustion stabilization[5-7], because it is either difficult for liquid kerosene to combust in supersonic flow when rocket being closed, or less efficient when remaining the rocket being as a torch[8].

Billig et al. of John Hopkins University (JHU) firstly proposed dual Combustion Ramjet[9] (DCR) in 1980s. DCR combines the best features of a ramjet and scramjet, and has several merits such as wider range of operating Mach number (3.5~6.5), easier ignition and more stable combustion, higher performance at low Mach number, and more convenient cooling of the wall. Unfortunately, there is also a disadvantage lies in the deficiency at a higher flight Mach number that exceeds 6.5.

Therefore, it is very attractive to combine ejector rocket with DCR for a near space plane flying at a Mach number no more than 6.5. In the present study, a conceptual Near Space Plane (Figure 1), powered by a Multi-Module Rocket Dual Combustion Ramjet (RDCR) Combined-Cycle engine, is presented. The engine is mounted under the venter of the near space vehicle airframe. The forebody of the Near Space Vehicle acts as a precompressor of the inlets, and the aftbody serves as part of the nozzle. To examine whether it would be capable of achieving, and to determine what influence planning of the flight path, characteristics of the engine under different operating modes will be studied analytically, and then, the trajectory optimizations will be implemented.

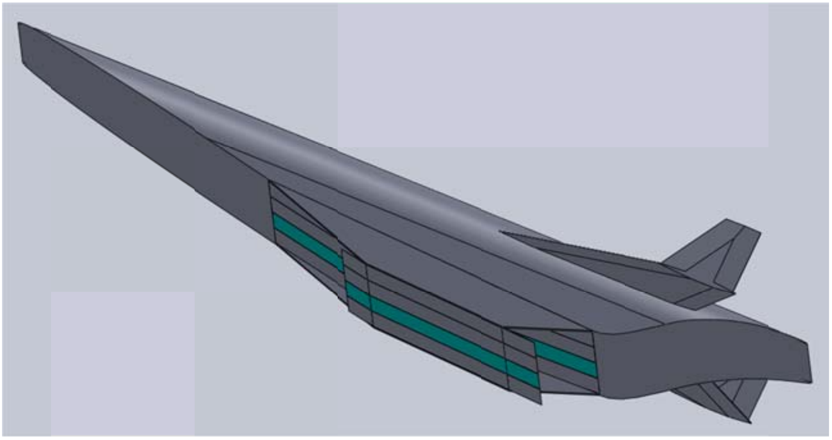


Figure 1: Schematic of a Near Space Vehicle with Multi-Module RDCR Combined-Cycle Engine

2. RDCRCC Engine Concepts

2.1 The operating modes

The RDCR Combined-Cycle engine module consists of the inlets, rockets, preburners, a combustor and a nozzle (Figure 2). The inlets include a supersonic inlet and a subsonic inlet. There are two rockets, four preburners and one combustor in the engine module.

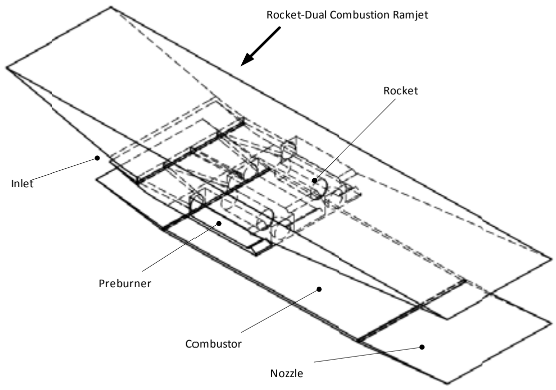


Figure 2: Schematics of RDCR Combined-Cycle engine module

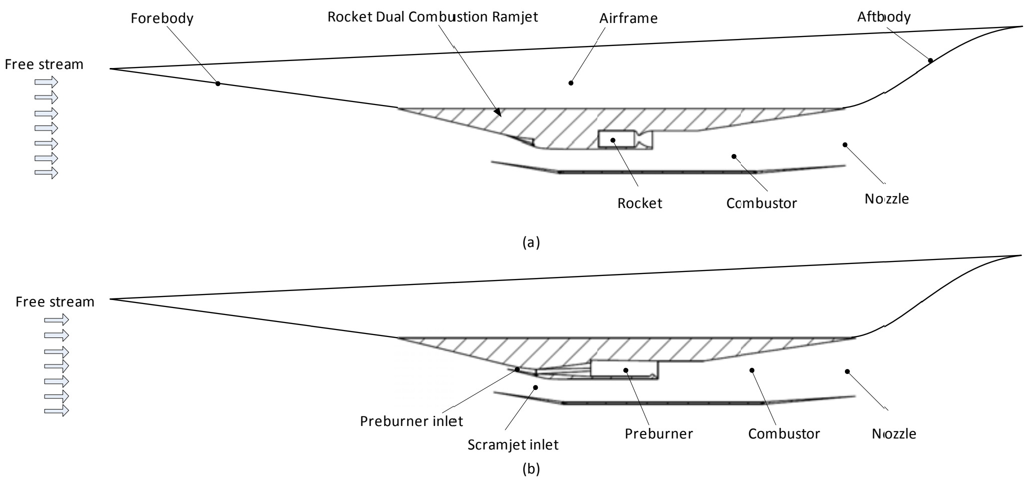


Figure 3: Schematics sections of RDCR Combined-Cycle engine module, (a) Section along the symmetry plane of the rocket, (b) Section along the symmetry plane of the preburner.

The operating modes of RDCR Combined-Cycle Engine include an ejector-jet mode, an ejector-ramjet mode, a DCR ramjet mode and a DCR scramjet mode.
In the ejector-jet mode, namely, the airbreathing rocket mode, air is breathed in by the ejector effect of the rocket. In this mode, the thrust is produced not only by the rocket engine itself, but also by the divergent section with the increased pressure of a mixture of air and rocket exhaust. Further subsonic combustion is attained by fuel injection from second fuel injectors.
In the ejector ramjet mode, the ejector rocket, which function as a gas generator to supply hot and fuel-rich gas, can operate at a higher mass flow rate to provide higher thrust for acceleration, or at a lower mass flow rate to gain higher specific impulse.
In the DCR ramjet and scramjet mode, the ejector rocket is closed and the preburner function as a torch to supply a continuous and stable igniter, which makes it possible to gain a much higher fuel specific impulse.
The RDCR Combined-Cycle Engine operates in the ejector-jet mode under the situation that the Mach number is less than three, in the ejector-ramjet mode or DCR ramjet mode from Mach 2-5.5, and in the DCR scramjet mode from Mach 5.5-6.0+. When the Mach number ranges from 2-6, whether to choose the ejector-ramjet mode or the DCR ramjet mode mainly depends on the multi-discipline optimization according to the tasks.
Since the operating Mach number ranges from 0.8-6.0+, geometry variable inlet and nozzle are reasonable selection. A geometry variable inlet made it more efficient within the whole operating range. A second throat at the exit of the engine is used to form a subsonic combustion in ejector -jet mode, ejector ramjet mode and DCR ramjet mode.

2.2 The trajectory

The plan trajectory of the Near Space Vehicle is sketched in Figure 4. Similar to X-43 developed by NASA, the Vehicle will be dropped at about Mach 0.8 and 10km above sea level. Under the combination thrust of the gravity and the engine, it accelerates to about Mach 2 at 5-8km above sea level, and then climbed to about 10-12km while the velocity being accelerated to about Mach 3-4. After that, Ejector-ramjet mode or DCR ramjet mode will be selected to accelerate the vehicle to about Mach 6. Ejector-ramjet mode provides more thrust with less efficiency, while DCR ramjet mode provides more efficiency and less thrust. A multi-discipline optimization should be carried out to decide which mode to choose. Then, the vehicle cruises under DCR scramjet mode, which provides stable combustion and high efficiency. Finally, after reaching the target point, the vehicle will glide back and land horizontally under the ejector-jet mode.

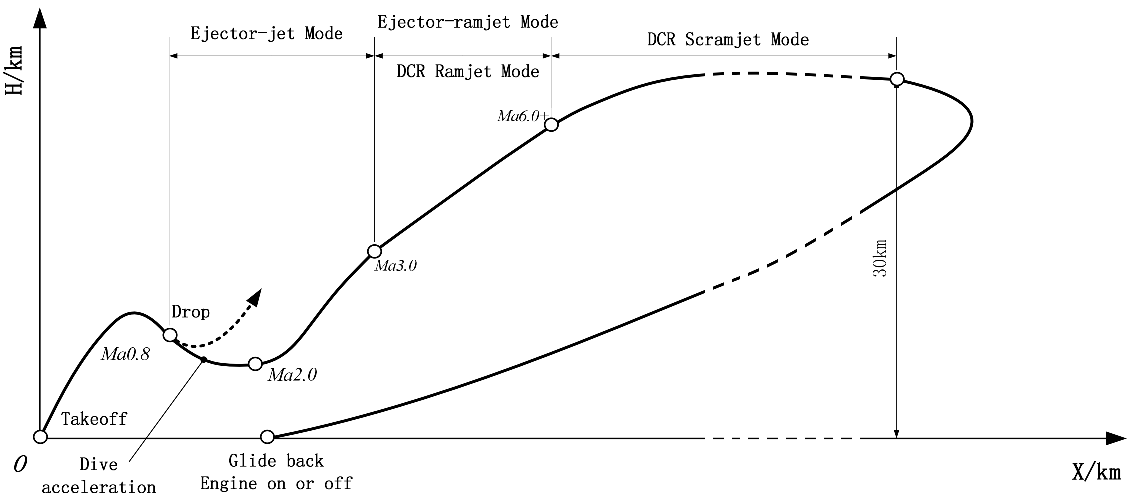


Figure 4: The sketch plan trajectory of the Near Space Vehicle.

3. Trajectory design

3.1 Thrust calculation model

The engine reference station for designation scheme employed hereinafter for engine analyses is given in Figure 5, a lifting body with the vehicle's forebody performing a large part of the inlet compression; the aftbody constitutes part of the nozzle. The engine therefor occupies the entire lower surface of the near space vehicle.

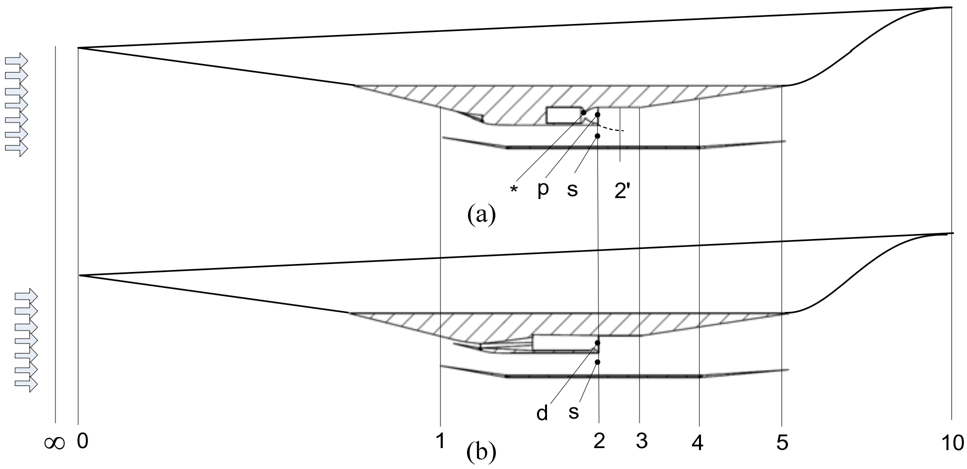


Figure 5: Simplified engine reference stations, (a) Section along the symmetry plane of the rocket, (b) Section along the symmetry plane of the preburner. Where ∞ - free-stream condition, 0 – external compression entrance, 1 - internal compression entrance, * - rocket throat, p-rocket nozzle exit, d - preburner exit, s - inlet exit, 2 - constant area entrance, 2' a virtual station where $p_{2's} = p_{2'p}$, 3 - constant area exit, 4 - combustion exit, 5 – internal nozzle exit, 10 – external nozzle exit.

A. Inlet

Given the flight Mach number M_∞ and the altitude Y_0, the secondary-flow properties, the total temperature T_{ts} and the specific heat ratio γ_s, are known. Then, the total pressure p_{ts} can be obtained for the approximate range $1<M_\infty<5$ by $p_{ts}=1.0\text{-}0.075(M_\infty\text{-}1)^{1.35}\,p_{t\infty}$, which is taken from American Military Standard MIL-E-5008B, and it is also consistent quite well until $M_\infty=6$. The corresponding inlet exit Mach number is approximate $M_{s0}=M_\infty^{0.5}$. For a given M_s, p_{ts} can be obtained by a modified correlation

$$p_{ts} = \begin{cases} \left[1.0-0.075\left(M_\infty-1\right)^{1.35}\right]\theta(\gamma, M_\infty^{0.5})\,p_{t\infty} & M_s \le 1 \\ \left[1.0-0.075\left(M_\infty-1\right)^{1.35}\right]\theta(\gamma, M_\infty^{0.5})/\theta(\gamma, M_s)\,p_{t\infty} & 1 < M_s < M_\infty^{0.5} \end{cases}, \tag{1}$$

where,

$$\theta(\gamma, \mathrm{M}) = \left[\frac{(\gamma+1)M^2}{2+(\gamma-1)M^2}\right]^{1/(\gamma-1)} \Bigg/ \left[\frac{2\gamma}{\gamma+1}M^2 - \frac{\gamma-1}{\gamma+1}\right]^{1/(\gamma-1)} \tag{2}$$

Since the Mach number of the preburner inlet $M_d<1$, the total pressure p_{td} can also be obtained by $p_{td}=[1.0\text{-}0.075(M_\infty\text{-}1)^{1.35}]\theta(\gamma, M_\infty^{0.5})\,p_{t\infty}$.

B. Ejector Modeling

In the ejector-jet and ejector-ramjet mode, the inlet of the preburner is closed, the rocket thrusters function as an ejector primary flow, and the flow from scramjet inlet function as an ejector secondary flow. The momentum was transferred from the rocket exhaust to the breathed air or from the breathed air to the rocket exhaust through a dividing streamline. After the interaction, the breathed air and the rocket exhaust flowed in parallel one-dimensionally at the same pressure with conservation of mass, impulse function, and energy. When inflow air was in subsonic speed, a flow rate of the breathed air was determined through this interaction.

Since it is generally considered that simultaneously mixing and combustion (SMC) will be less efficiency than diffuse and burning (DAB), the DAB mode will be adopted in the calculation. Therefore, an equivalence ratio of the ejector rocket is assigned to 1.0, and assumes that there is no combustion during the mixing.

It is assumed that the throat area $A*$ of the primary rocket nozzle is choked and that a supersonic primary flow entrains a subsonic secondary flow into a constant-area ejector mixing duct. The conservation equations (continuity, momentum, and energy) are applied to the overall control volume consisting of the constant-area ejector-mixing

duct. A frictionless, steady, adiabatic flow is assumed, neglecting pressure losses due to air inlet compression, primary and secondary flow interaction, heat release, and flame holders. Uniform velocity and pressure distribution are assumed for the primary and secondary flows at the control-volume entrance (station 2), and a uniform, fully mixed flow is assumed at the exit of the control volume (station 3). No reaction is assumed to occur in the mixer, whereas a secondary combustion is considered in the ramjet burner if the primary flow is fuel rich or extra fuel is added in the ramjet burner.
Thus, for a given flight condition and propellant combination, assuming $p_{2's} = p_{2'p}$ at a virtual station 2', the characteristics of the mixed flow in station 3 are determined by five parameters:

a) The primary rocket mixture ratio ψ, which determines the primary-flow characteristics, that is, γ_p, R_p, $\Theta = T_{ts}/T_{tp}$, and $c = C_{ps}/C_{pp}$.
b) The 2 ejector rocket geometries, that are, $\varphi^* = A^*_p/A_3$ and $\varphi_s = A_s/A_3$.
c) The secondary flow velocity coefficient λ_s, which determines the Mach number M_s, the total pressure p_{ts}, and the static pressure p_s.
d) The primary total pressure p_{tp}, which determines $\beta = p_{ts}/p_{tp}$, $\lambda_{2's}$, $\lambda_{2'p}$ and the primary rocket mass flow $\dot{m}_s$.

Using minimized Gibbs free energy method, the primary flow properties (T_{tp}, γ_p, and λ_p) can be computed as a function of rocket chamber mixture ratio ψ for a given propellant combination.
Thus, the secondary-to-primary mass flow ratio n can be evaluated as

$$n = \frac{\dot{m}_s}{\dot{m}_p} = \frac{K(\gamma_s, R_s)}{K(\gamma_p, R_p)} \frac{\beta}{\sqrt{\Theta}} \frac{\varphi_s}{\varphi^*} q(\gamma_s, \lambda_s) \ , \tag{3}$$

where,

$$K(\gamma, R) = (\gamma/R)^{1/2} \left[(\gamma+1)/2\right]^{-(\gamma+1)/2(\gamma-1)} \text{ and } q(\gamma,\lambda) = \left[(\gamma+1)/2\right]^{1/(\gamma-1)} \lambda \left[1-\lambda^2(\gamma-1)/(\gamma+1)\right]^{1/(\gamma-1)} . \tag{4}$$

The energy and continuity equations give the properties of the mixed flow

$$\gamma_3 = \frac{1+nc}{\gamma_s/\gamma_p + nc}\gamma_s \ , \ R_3 = \frac{R_p + nR_s}{1+n} \text{ and } \frac{T_{t3}}{T_{tp}} = \frac{1+nc\Theta}{1+nc} . \tag{5}$$

Whereas, having combined momentum and continuity equations,

$$\lambda_3 = \frac{Z \pm \sqrt{Z^2-4}}{2} , \tag{6}$$

where,

$$Z = \frac{\sqrt{\frac{2\gamma_p}{\gamma_p+1}R_p}\lambda_p + n\sqrt{\Theta}\sqrt{\frac{2\gamma_s}{\gamma_s+1}R_s}\lambda_s + \left(1+\frac{1-\varphi_{2's}}{\varphi_{2's}}\right)\sqrt{\frac{\gamma_p+1}{2\gamma_p}R_p}\left(\frac{1}{\lambda_p} - \frac{\gamma_p-1}{\gamma_p+1}\lambda_p\right)}{(1+n)\sqrt{\frac{1+nc\Theta}{1+nc}}\sqrt{\frac{\gamma_3+1}{2\gamma_3}R_3}} . \tag{7}$$

Real roots are obtained for λ_3, if | Z | > 2. The negative sign root gives $\lambda_3 \leq 1$, while the positive sign gives $\lambda_3 \geq 1$. The subsonic solution is obtained also by allowing the possible supersonic solution to diffuse through a normal shock wave. When | Z | ≤ 2, the ejector exit is chocked. In the present study, only subsonic solution is taken into account. Moreover, it is supposed that the mixer duct has a length, which will allow a fully mixed flow at its end. Then the momentum equation allows one to compute the total pressure in station 3 and to determine the engine total pressure p_{t3}

$$p_{t3} = p_{tp} \frac{K(\gamma_p, R_p)}{K(\gamma_3, R_3)} \frac{\varphi^*}{q(\gamma_3, \lambda_3)} \sqrt{\frac{1+nc\Theta}{1+nc}} (1+n) \,. \quad \textbf{(8)}$$

The ejector must face constraints that limit design parameters and performance. Three constraints concern the secondary-flow entrance into the ejector, one related to the aforementioned mixer exit choking ($\lambda_3 \le 1$), and another to the rocket and one to the virtual station 2'.
The following constraints must be satisfied at engine section 2:
1) Blockage: the primary flow should not occupy all of the mixer area, that is, $\varphi^* \le q(\gamma, \lambda_p)$; then, for a given geometry φ^*, an absolute maximum value of λ_p, $\lambda_{p,\,\mathrm{lim}}$ exists.
2) Backflow: the static pressure p_p should not exceed p_{ts}, that is, $\beta \ge \pi(\gamma_p, \lambda_p)$, where, $\pi(\gamma, \lambda)=[1-(\gamma-1)\lambda^2/(\gamma+1)]^{\gamma/(\gamma-1)}$; then, a maximum allowed value of β can be found: $\beta_{\min}(\varphi^*, \gamma_p) = \pi(\gamma_p, \lambda_{p,\mathrm{lim}})$.
3) Secondary-flow choking: assuming $p_{2's} = p_{2'p}$,

a) When the inlet is unstart, $\lambda_s \le 1$, then, $\beta \le \pi(\gamma_p, \lambda_{2'p})/\pi(\gamma_s,1)$; then the limit for the secondary-flow's choking is $\beta_{ck}(\varphi^*,\gamma_p) = \beta_{\min}/\pi(\gamma_s,1)$.
b) And, while the inlet is start, $\lambda_s \ge 1$, then, $\beta \ge \pi(\gamma_p, \lambda_{2'p})/\pi(\gamma_s,1)$; then the limit for the secondary-flow's choking is $\beta_{ck}(\varphi^*,\gamma_p) = \beta_{\min}/\pi(\gamma_s,1)$.

C. Second Fuel Injector and Choking Condition

The combustion energy release is modeled as heat addition with mass addition, and the total energy was the sum of fuel reaction heat and those of the rocket exhaust, fuel, and the breathed air, therefore, the total temperature

$$T_{t4} = \frac{C_{p3}T_{t3} + fC_{pf}T_{tf} + \eta_b f h_{PR}}{(1+f)C_{p4}}, \quad \textbf{(9)}$$

where, h_{PR} is the fuel reaction heat, and η_b is heat addition efficiency , here, η_b=0.90.
1) Subsonic combustion
In the ejector-jet, ejector ramjet, and DCR ramjet modes, λ_3<1, subsonic combustion was attained with fuel injecting from the second injector, and the choking condition was assigned at the exit of the engine combustion(station 4), that is λ_4=1, and, $v_4=[2\gamma_4 R_4 T_{04}/(\gamma_4+1)]^{0.5}$. Neglecting the combustor drag and fuel injection axial velocity, assuming that $p_{t4}= p_{t3}$, the area A_4 can also be acquired with mass conservation.
2) Supersonic combustion
In DCR scramjet mode, λ_3>1, choking was not assigned at the exit of the engine combustion, constant pressure combustion is assumed. Assume that fuel is injected perpendicular to the stream direction. Neglecting the combustor drag and fuel injection axial velocity, the momentum conservation equation can be written as $v_4=v_3/(1+f)$, where, $v_3=\lambda_3[2\gamma_3 R_3 T_{03}/(\gamma_3+1)]^{0.5}$, the mass ratio of fuel to air $f = m_f/m_3$. Then, $\lambda_4=v_4/[2\gamma_4 R_4 T_{04}/(\gamma_4+1)]^{0.5}$, and, the area A_4 can also be acquired with mass conservation.

D. Divergent Section

Assuming that the expanding of the combustion gas in the divergent section is isentropic, therefore, $p_{t10}=\eta_e p_{t4}$, and $T_{t10}=T_{t4}$. In addition, the velocity coefficient λ_{10} can be obtained by solving the mass conservation equation

$$A_4 q(\gamma_4, \lambda_4) = A_{10} q(\gamma_4, \lambda_{10}) \quad \textbf{(10)}$$

E. Uninstalled engine thrust

The optimization objectives are the specific impulse I_{sp} and thrust T. An optimum expansion is assumed ($p_{10} = p_\infty$), then,

$$I_s = \frac{\dot{m}_{10}v_{10} - \dot{m}_0 v_\infty + (p_{10} - p_\infty)A_{10}}{\dot{m}_{10} - \dot{m}_0}, \quad \textbf{(11)}$$

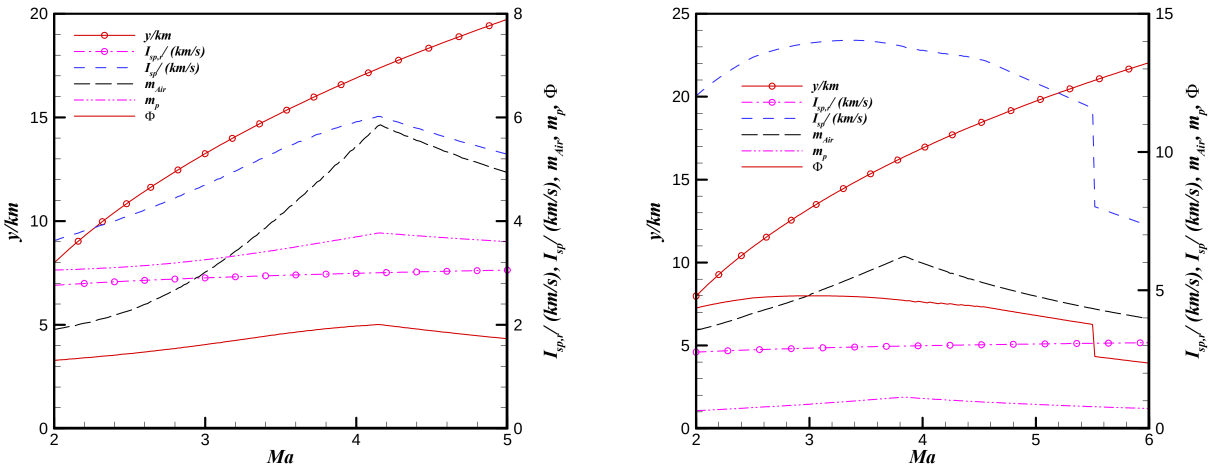
and,

$$T = \left(\dot{m}_{10} - \dot{m}_0\right) I_{sp} \;, \tag{12}$$

where, the velocities of v_{10} and v_∞ can be obtained with $v = [2\gamma RT/(\gamma+1)]^{0.5}$.

F. Engine performance

The reference rocket operates with a propellant combination of liquid oxygen and kerosene, at a total pressure of 5Mpa and a fuel to air equivalence ratio of 1.0. The fuel reaction heat of the kerosene is about 4.3MJ/kg. The engine performance was calculated under a dynamic pressure of 100kPa with the ejector rocket in operation or being closed, and Figure 6 shows the results, respectively. Where, the mass flow rate of the breathed air and the propellant are normalized with that of the reference rocket.



(a) The engine performance with ejector rocket on (b) The engine performance with ejector rocket off

Figure 6: The engine performance against flight Mach number at a dynamic pressure of 100kPa.

Under the situation that the dynamic pressure is fixed at 100kPa, the flight altitude increase proportionally from about 8-22km. When the ejector rocket is on, the engine operates at ejector-jet or ejector ramjet mode. With the Mach number increasing from 2 to 5, the normalized mass flow rate of the breathed air increases from 1.9 to a peak of 5.8, and then decreases to 5.0. The non-proportionally growth of the breathed air is caused by mixer exit choking, that is, the mixing flow of the rocket exhaust and the breathed air choke at the mixer exit. The specific impulse ranges from 3.6-6.0km/s and the corresponding thrust augmentation ranges from 1.3-2.0, a higher thrust can be provided at the cost of more propellant consumed, the efficiency should be evaluated in the trajectory optimization.

When the ejector rocket is off, the engine experiences a DCR ramjet or scramjet mode. With the Mach number increasing from 2 to 6, the normalized mass flow rate of the breathed air has a peak during the process, increasing from 3.6 to 6.2, and then decreasing to 4.6. At Mach 5.5, the subsonic combustion turns to supersonic combustion, which made the specific impulse sharply dropped from about 11-14km/s to 7-8km/s.

3.2 Basic relations of motion

For a vehicle that flies within the atmosphere, the gravitational attraction of all other heavenly bodies may be neglected. Let it be assumed that the vehicle is moving in rectilinear equilibrium flight and that all control forces, lateral forces, and moments that tend to turn the vehicle are zero. Then, the trajectory is two-dimensional and is contained in a fixed plane. The wings are inclined to the flight path at an angle of attack α, which give a lift in the direction normal to the flight path. Figure 7 shows two-dimensional free-body force diagram for flying vehicle schematically.

Let θ be the angle of the flight path with the horizontal and ψ the angle of the direction of thrust with the horizontal. Therefore, the acceleration equation in the direction of the flight path and the direction normal to the flight path can be written as

$$m\left(du/dt\right) = F\cos\left(\psi - \theta\right) - D - mg\sin\theta \quad \textbf{(13)}$$

$$mu\left(d\theta/dt\right) = F\sin\left(\psi - \theta\right) + L - mg\cos\theta \quad \textbf{(14)}$$

where, D and L are the aerodynamic drag force and lift force, which are in a direction opposite and normal to the flight path, respectively. They can be expressed as functions of the flight speed u, the mass density of the air ρ, and reference surface of area A_{ref}:

$$D = C_D \tfrac{1}{2}\rho u^2 A_{ref} \quad \textbf{(15)}$$

$$L = C_L \tfrac{1}{2}\rho u^2 A_{ref} \quad \textbf{(16)}$$

where, C_D and C_L are drag and lift coefficients, respectively. The drag and lift coefficients are primarily functions of the vehicle configuration, flight Mach number, and angle of attack.

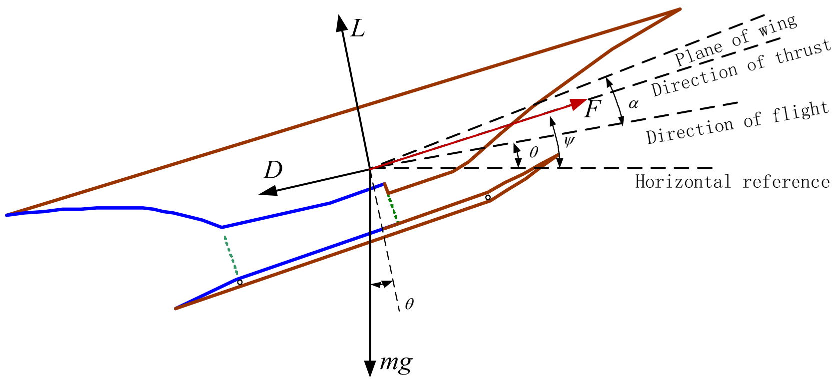


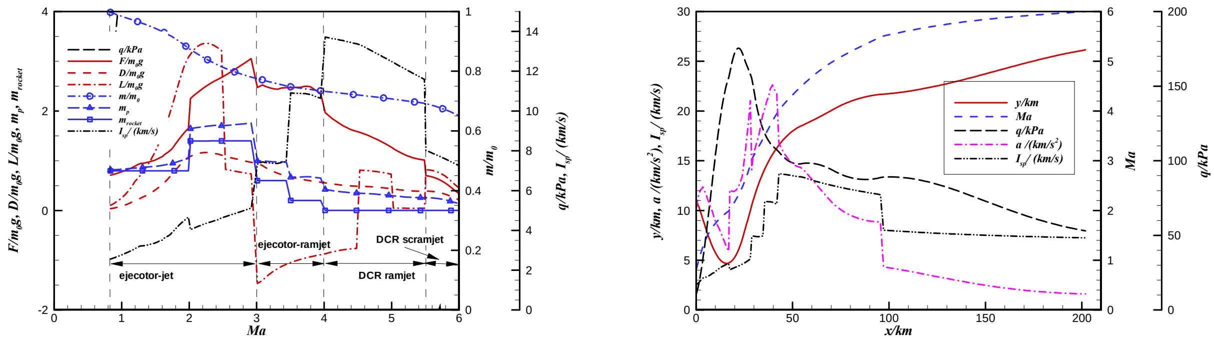
Figure 7: Two-dimensional free-body force diagram for flying vehicle.

4. Numerical results

4.1 An example trajectory

For a vehicle propelled by a RBCC engine, the primary difficulty lies in the insufficiency thrust to accelerate the vehicle at lower initial Mach number about 0-1.5. At that beginning, the dynamic pressure $\frac{1}{2}\rho u^2$ is quite small, therefore the lift will be insufficient and the flight path tend to declined downward at an angle of θ. Then, partial of the gravity, $mg\sin\theta$, has a positive acceleration on the vehicle, which helps the engine propel the vehicle at a higher acceleration. Here, an example was calculated with an initial flight path angle of $\theta = -30^\circ$, and 11km above sea level, the results were given in figure 9.

Due to the low dynamic pressure, through operates in ejector-jet mode, the thrust to weight is less than one at the very beginning, which makes it hardly to accelerate the vehicle upward. While declined downward, with the help of gravity, the acceleration increases to above $10m/s^2$. The vehicle flight declines from 11km to about 4.5km above sea level, where the velocity accelerates to about Mach 2.0 with a dynamic pressure of about 150kPa. The lift turns the flight path direction to horizontal, thereafter, continuously turns the direction upward. At that turning point, not only does the ejector augment the thrust to weight to above 2.3, but also an enough lift can be obtained at a moderate attack angle to bear the vehicle gravity, which add up to an acceleration of above $20m/s^2$. Following that, the total pressure of the ejector rocket decreases to operate in ejector ramjet or DCR ramjet mode, this reduces the propellant consuming and elevates the propellant specific impulse. The propellant specific impulse ranges from 7km/s to 10km/s with different ejector rocket total pressure in ejector ramjet mode, and 11km/s to 14km/s in DCR ramjet mode. Then, when the Mach number exceeds 5.5, the engine operates in DCR scramjet mode, supersonic combustion is introduced to decrease the heat load, the fuel specific impulse ranges from 7km/s to 8km/s. Finally, the cruise point is reached at Ma 6.0 and about 26km above sea level. At the cruise point, the remain mass to initial mass is about 0.65.



(a) History of forces, residual vehicle mass ratio, specific impluse and normalized propellant mass vs. Mach number

(b) History of trajecotry altitude, specific impluse, Mach number, acceleration velocity and dyanmic pressure vs. Horizontal range

Figure 9: An example trajectory of a vehicle dropped at Ma 0.8 and 11km above sea level with an initial flight path angle of θ = -30˚.

4.2 Trajectory optimization

In the present analysis, the near space plane is handled as a point mass following a two dimensional path in the equatorial plane. It will be drop at about Mach 0.8, 10km above sea level and in a direction of -30˚≤θ≤10˚, starting its engine in ejector-jet mode. After leaving the runway, it accelerates to Mach 2.5 and switches to ejector ramjet or DCR ramjet mode. Finally, at Mach 6 and 25~30km above sea level, it switches to DCR scramjet mode, and cruises to the destination. This analysis does not account for cruising and landing. The constraints are a maximum dynamic pressure of 200kPa and a maximum loading of 30m/s^2.

The actual space plane will use ailerons to control its flight. For simplicity, however, a time history of the attack angle was used to determine the flight trajectory. Specifically, the attack angle was assumed to remain within the range of -5º ~ 8º. In order to use the optimization procedure, the function must be discretized. For this process, the Mach histories of the attack angle α and ejector rocket total pressure p_{tp} were divided into 9 parts.

The 22 design parameters include:

1) The vehicle inlet area A_0;
2) 2 ejector rocket geometries: $\varphi^* = A^*_p/A_3$ and $\varphi_s = A_s/A_3$;
3) Initial trajectory angle θ_0 and altitude H_0;
4) 9 segments of attack angle α: α_1, α_2, …, α_9;
5) 8 segments of ejector rocket total pressure p_{tp}: p_{0R1}, p_{0R2}, …, p_{0R8}, and, p_{0R9}=0, since the engine will work at DCR scramjet mode;

The propose of trajectory optimization is to find a set of design parameters to propel the vehicle to cruise point, where the Mach number M_c=6 at Y_c=26km altitude in horizontal direction (θ_c=0). The error function δ is defined as

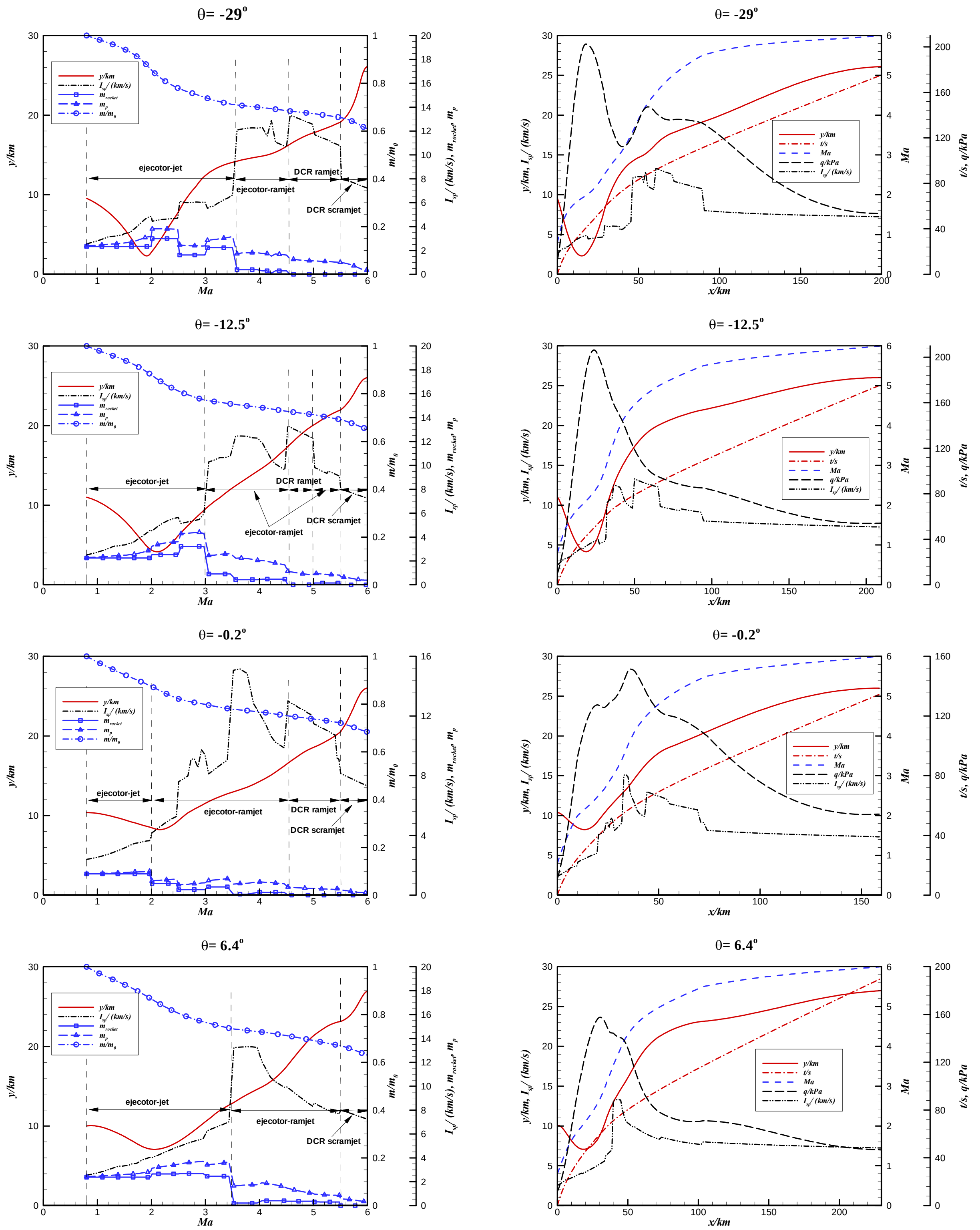
$$\delta = |Y - Y_c| + |\theta| \tag{17}$$

The minimization problem of equation (17) will be solved through the integrated optimization method.

A. Effect of initial trajectory angle θ_0

The effect of initial trajectory angle θ_0 on the vehicle performance and trajectory design parameters was investigated, and some of the results were shown in Figure 10. During the trajectory, the frontal three modes experienced, including the ejector-jet mode, the ejector ramjet mode and the DCR ramjet mode, can be evidently distinguished with the total pressure of ejector rocket. The specific impulse ranges from 3-6km/s in ejector-jet mode, 6-9km/s in ejector ramjet mode, 11-14km/s in DCR ramjet mode and 7-8km/s in DCR scramjet mode.

Generally, the lower the initial trajectory angle θ_0 is, the the higher the maximum dynamic pressure and the lower altitude of the trajectory bottom tend to be experienced.



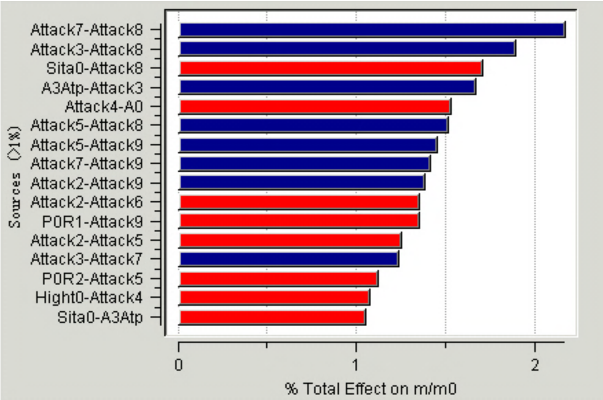
(a) History of trajecotry altitude, residual vehicle mass ratio, specific impluse and normalized propellant mass vs. Mach number

(b) History of trajecotry altitude, specific impluse, Mach number, time and dyanmic pressure vs. Horizontal range

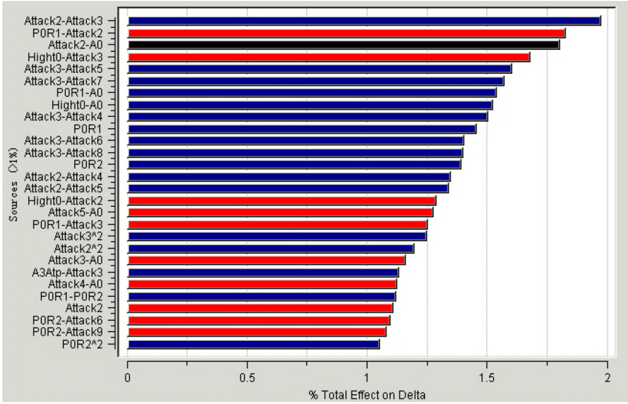
Figure 10: The effect of initial angle θ_0 on the vehicle performance and trajectory parameters.

B. DOE analysis

By using Design of Experiment (DOE) method, the effects of design parameters on performances are confirmed as shown in Figure 11. The parameters effects analysis result indicates that there are no evidently significant effects for overall system performance, and the double parameters (α_7-α_8, α_3-α_8, θ_0-α_8, etc.) have relatively more significant effects. That means that the trajectory optimization of the near space vehicle is of great complexity, the correlations among the design parameters play very important role.



(a) The Pareto plot for m/m_0



(b) The Pareto plot for δ

Figure 11: The effect of design parameters on performance

5. Conclusions

In the present study, a Near Space Plane, powered by Multi-Module Rocket Dual Combustion Ramjet (RDCR) Combined-Cycle Engine has been presented. The vehicle was planned to be dropped at about Mach 0.8 and 10km above sea level. Under the combination thrust of the gravity and the engine, it accelerates to about Mach 2 at 5~8km above sea level. Then it climbs up and accelerates under the Ejector-ramjet mode or DCR ramjet mode. Ejector-ramjet mode provides higher acceleration with less efficiency, while DCR ramjet mode provides more efficiency and lower acceleration. Finally, the plane climbs up to about 26km altitude, accelerates to Mach 6, and starts to cruise in DCR scramjet mode. To examine whether the concept would be capable of achieving, characteristics of the engine under different operating modes were studied analytically, ejector operation constraints are taken into account. A trajectory optimization was carried out to decide the engine and trajectory design parameters. The parameters effects analysis was carried out, the results indicates that there are no evidently significant effects for overall system performance, and the double parameters have relatively more significant effects.

Acknowledgment

The financial support of the National Natural Science Foundation of China (No. 11172324) is gratefully acknowledged.

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