Aerodynamic Shape Optimization of Membrane Aeroshell Aerocapture Vehicle

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In this paper, flexible aeroshell shape optimization method for aerocapture mission is formulated using particle based membrane model and GA with response surface modeling. And the calculation is conducted considering various objective function. As a result, it is shown that proposed shape optimization method for flexible aeroshell vehicle works properly, and thus gives reasonable solutions. In this analysis, three clusters of solution sets are obtained; 1) solutions which minimized fuel mass fraction, 2) solutions which minimized both TPS mass fraction and outer frame mass fraction, 3) solutions which minimized ballistic coefficient. Cluster 1) could reduce the total mass fraction most in three kinds of clusters. It shows that fuel mass is much more sensitive to total mass than any other required mass for assumed entry condition. And also, tradeoff relationship between minimization of fuel mass fraction and minimization of outer frame mass fraction.

Key Words: Aerocapture, Flexible Aeroshell, Genetic Algorithm, Response Surface Model

Nomenclature

\( C_{ae} \) : ratio of aerodynamic force to stiffness
\( C_p \) : ballistic coefficient
\( C_D \) : drag coefficient
\( C_m \) : pitching moment coefficient
\( C_f \) : pressure coefficient
\( C(X) \) : equality constraints
\( d \) : design vector
\( D \) : Drag
\( E_{frame} \) : Young’s modulus of outer frame
\( E_{mem} \) : Young’s modulus of membrane
\( F_a \) : aerodynamic force on membrane
\( F_s \) : stretching force on membrane
\( STRCT\% \) : outer frame mass fraction
\( FUEL\% \) : fuel mass fraction
\( h_{mem} \) : thickness of membrane
\( I_{sp} \) : specific impulse
\( J(X) \) : objective function (vector)
\( L_{ref} \) : reference length of membrane
\( S_{frame} \) : cross sectional area of outer frame
\( S(X) \) : inequality constraint
\( t \) : time
\( T_{eq} \) : radiation equilibrium temperature
\( T_{frame} \) : surface temperature on outer frame
\( T_{mem} \) : surface temperature on membrane
\( TPS\% \) : thermal protection mass fraction
\( u \) : control vector
\( X \) : optimization design variables
\( x \) : state vector
\( \alpha \) : angle of attack
\( e_{frame} \) : strain in outer frame
\( e_{mem} \) : strain in membrane
\( \phi \) : objective function (scalar)

Subscripts

0 : initial
f : final

1. Introduction

1.1. Backgrounds

Recent years, planetary exploration including the Mars and Moon becomes an international concern. Particularly, the Mars attracts people’s attention because it has scientifically interesting topics as a terrestrial planet. The establishment of low cost, highly reliable and efficient space transportation system and orbit insertion technique for Mars or other planets seems to be an urgent task. In the planetary explorations, the most important thing is the on-board capability. Orbit insertion technique that uses OME (Orbital Maneuvering Engine) of Isp of more than 310sec has been established so far21.

However, such a conventional orbit insertion technique that uses the chemical propulsion system imposes a constraint that a large amount of chemical propellant for deceleration should be equipped through the trip from launch to arrival. This would put a limitation to the payload mass for scientific instruments essentially needed for observations. For example, if a spacecraft leaves Low Earth Orbit (LEO) of 400km and it is inserted into circular orbit of 500km around the Mars, the required fuel mass fraction for deceleration becomes 48% assuming that Isp is 315sec. Compared with such a conventional technique, the aerocapture technique23 has the
potential for providing substantial weight savings, because aerodynamic drag is used to remove enough kinetic energy. It also enables us to complete orbital maneuvering in smaller time than aerobraking, which takes several weeks or several months to insert the vehicle into the target orbit. Therefore, it is expected to reduce the cost of the interplanetary exploration in the future.

However, an aerocapture mission involves a lot of complicated technical difficulties. For example, the corridor width for entry path angle is quite narrow and it strongly depends on the vehicle’s aerodynamic characteristic. During the atmospheric flight, the aerodynamic lift must be adequately controlled to obtain an optimized descending and ascending rate without losing necessary deceleration so that the vehicle can avoid the crash onto the ground even when the uncertainties in the atmospheric properties exist. Moreover, the aerodynamic heating must be taken into account as one of the most critical problems in aerocapture mission. However, installing the flexible membrane as its aeroshell has the possibility to solve this problem. Because flexible aeroshell is lightweight and foldable, so a vehicle can decelerate at high altitude where atmospheric density is very low thanks to a large but low-mass aeroshell. A deployable and flexible aeroshell made by thermally-resistant textile has the potential for realizing a low ballistic coefficient aerocapture vehicle. Recent years, reentry system using deployable and flexible aeroshell has been researched and developed in various organizations.

1.2. Aerocapture mission

Figure 1 shows an aerocapture flight profile schematic. The vehicle approaches the Mars from a hyperbolic approach trajectory, shown at point 1, and then enters the atmospheric interface, shown at point 2. From point 3 to point 6, the drag on the vehicle provides the deceleration, which is required to capture the vehicle into the desired orbit. At point 7 and 8, small delta-V burn is performed to raise the periapsis and to circularize the vehicle around the planet, respectively.

This paper considered the following Mars aerocapture mission. The vehicle approaches the Martian atmosphere at an entry speed of 5.7km/s at an altitude of 120km, which is calculated using patched conic approximation, and the gross weight is 800kg. The target orbit is set to be 500km, which is a circular science observation orbit. Because the orbit after escaping from atmosphere is changed according to the ballistic coefficient and flight trajectory, the spacecraft must perform a small delta-V burning at the periapsis and to circularize the vehicle around the planet.

This paper considered the following Mars aerocapture mission. The vehicle approaches the Martian atmosphere at an entry speed of 5.7km/s at an altitude of 120km, which is calculated using patched conic approximation, and the gross weight is 800kg. The target orbit is set to be 500km, which is a circular science observation orbit. Because the orbit after escaping from atmosphere is changed according to the ballistic coefficient and flight trajectory, the spacecraft must perform a small delta-V burning at the periapsis and to circularize the vehicle around the planet.

1.3. Necessity of optimization

Firstly, numerical optimization makes the design process more efficient. When evaluating a certain design candidate, numerical analysis such as CFD and FEM will be executed. These are very useful design tools. However, after a certain design candidate has been evaluated, the next re-designed candidate is proposed and these analyses are made once again. Such a process needs quite a lot of calculation cycle. And also, re-designing is often made by experience or intuition of designers and by trial and error process. On the other hand, introducing the calculation which integrates the numerical analysis and optimization can automate the time-consuming re-design process. And also, such a calculation makes it possible for designers to guess the detail setting conditions for further analysis and experiments.

Secondary, design guidelines for optimum flexible aerocapture vehicle can be obtained. Compared with a rigid aeroshell vehicle, a flexible aeroshell vehicle has various advantages and characteristics such as lightweight, high packaging efficiency, low cost, heating avoidance. It is necessary to extract design guidelines to maximize such advantages and to meet requirements in aerocapture mission. Moreover, compared with a rigid aeroshell vehicle, for which the available variation of the aerodynamic characteristics is limited, flexible vehicle is possible to change its aerodynamic coefficient widely along its flight trajectory. The aeroshell shape is passively deformed by dynamic pressure. As a result, the aerodynamic coefficient of flexible aeroshell vehicle is varies along atmospheric flight trajectory. Therefore, process of obtaining solutions for flexible aeroshell vehicle might become much more complicated than that of rigid aeroshell vehicle.

1.4. Objective

The purpose of this work is to execute shape optimization of aerocapture vehicle, which installs deployable and flexible membrane as its aeroshell in order to construct an efficient design tool and to obtain design guidelines to fulfill requirements and maximize advantages of flexible aeroshell. In this paper, axisymmetric shape, that is ballistic entry vehicle, is considered as a preliminary step of the research.

![Fig. 1. Aerocapture flight schematic.](image)

2. Models

2.1. Formulation of optimization problem

In this section, optimization problem for aerocapture mission is formulated. In a certain dynamic system, defining the state vector \( x(t) \in \mathbb{R}^n \) and control vector \( u(t) \in \mathbb{R}^m \), which are the time dependent variables, and design vector \( d \in \mathbb{R}^d \) which is independent of time, state equation of this system is given as Eq.(1). In this paper, three degree of freedom equation of motion is considered. Following formulation is the general expression which considers the simultaneous optimization of aerodynamic shape and trajectory control, but in this paper, as a preliminary step, attitude control is not considered, therefore \( u(t) = 0 \).
\[ x(t) = f(x(t), u(t), d) \]  

(1)

Initial \((t = t_0)\) and terminal \((t = t_f)\) conditions are given as constraints as follows:

\[
\begin{align*}
C_{ue}(x(t_0), u(t_0), d) &= 0 \\
S_{ue}(x(t_0), u(t_0), d) &\leq 0 \\
C_{ue}(x(t_f), u(t_f), d) &= 0 \\
S_{ue}(x(t_f), u(t_f), d) &\leq 0
\end{align*}
\]  

(2)

Equality and inequality constraints when flying in the atmosphere at an arbitrary time \(t \in [t_0, t_f]\) are,

\[
C(x(t), u(t), d) = 0 \\
S(x(t), u(t), d) \leq 0
\]  

(4)

Dividing the time \(t \in [t_0, t_f]\) into \(N\) nodes, control vector and state vector for each time node are given as follows:

\[
\begin{align*}
u_i &= (i = 0, 1, \ldots, N) \\
x_i &= (i = 0, 1, \ldots, N)
\end{align*}
\]  

(5)

Defining the variable vector and objective function vector as \(X\) and \(J(X)\), respectively, optimization problem can be formulated as mathematical programming problem shown as follows:

variables: \(X = [u_0, u_1, \ldots, u_N, d]\)  

minimize: \(J(X) = \{\phi_1, \phi_2, \ldots, \phi_k\}\)  

subject to:

\[
\begin{align*}
C(X) &= C(x(t), u(t), d) = 0 \\
S(X) &= S(x(t), u(t), d) \leq 0
\end{align*}
\]  

(7)

(8)

2.2. Variables

As a shape model for flexible aeroshell, thin flare-type aeroshell, which is composed of a conical membrane and a rigid outer frame supporting structure, is selected. Design variable vector \(d\) is defined as the size in each part of the shape model. These are listed in Fig.2 and table 1. Variables can be roughly divided into two categories: 1) Variables related with vehicle shape 2) Variables related with aeroshell and frame materials. In this paper, Zylon® and titanium are assumed to be used as aeroshell and frame material. Material properties for Zylon® and titanium are listed in table 2. Considering not increase the weight of rigid outer frame, it is assumed that hollow cross sectional type of 1mm of thickness is used.

Table 1. Design variables and their domain

<table>
<thead>
<tr>
<th>Property</th>
<th>Zylon®</th>
<th>Titanium</th>
</tr>
</thead>
<tbody>
<tr>
<td>Curvature of nose radius</td>
<td>0.5 ≤ R&lt;sub&gt;nr&lt;/sub&gt;[m] ≤ 2.5</td>
<td></td>
</tr>
<tr>
<td>Radius of flare</td>
<td>2.0 ≤ R&lt;sub&gt;flare&lt;/sub&gt;[m] ≤ 8.0</td>
<td></td>
</tr>
<tr>
<td>Flare angle</td>
<td>10 ≤ θ&lt;sub&gt;flare&lt;/sub&gt;[deg] ≤ 70</td>
<td></td>
</tr>
<tr>
<td>Radius of outer frame</td>
<td>0.01 ≤ R&lt;sub&gt;nose&lt;/sub&gt;[m] ≤ 0.2</td>
<td></td>
</tr>
<tr>
<td>Material of aeroshell</td>
<td>Zylon®</td>
<td></td>
</tr>
<tr>
<td>Material of outer frame</td>
<td>Titanium</td>
<td></td>
</tr>
</tbody>
</table>

Table 2. Property of materials

<table>
<thead>
<tr>
<th>Property</th>
<th>Zylon®</th>
<th>Titanium</th>
</tr>
</thead>
<tbody>
<tr>
<td>Young’s modulus</td>
<td>18 [GPa]</td>
<td>120 [GPa]</td>
</tr>
<tr>
<td>Thickness</td>
<td>0.125 [mm]</td>
<td>1 [mm]</td>
</tr>
<tr>
<td>Density</td>
<td>0.142 [kg/m³]</td>
<td>4.5 [g/cm³]</td>
</tr>
<tr>
<td>Allowable strain</td>
<td>3.5 [%]</td>
<td>1.0 [%]</td>
</tr>
<tr>
<td>Allowable temper.</td>
<td>650 [degC]</td>
<td>800 [degC]</td>
</tr>
</tbody>
</table>

2.3. Objective function

Here, following objective functions can be considered: 1) minimization of fuel mass fraction (FUEL%), 2) minimization of thermal protection system mass fraction (TPS%), 3) minimization of outer frame mass fraction (STRCT%), 4) minimization of ballistic coefficient (C<sub>b</sub>), 5) minimization of radiation equilibrium temperature (T<sub>eq</sub>). If aeroshell shape is asymmetric, maximization of L/D should be considered for keeping robustness against corridor width and dispersion of atmospheric density.

\[
\begin{align*}
\phi_1(X) &= FUEL\% \\
\phi_2(X) &= TPS\% \\
\phi_3(X) &= STRCT\% \\
\phi_4(X) &= C_b \\
\phi_5(X) &= T_{eq}
\end{align*}
\]  

(9)  

(10)  

(11)  

(12)  

(13)

Fuel mass fraction is evaluated using Eq.(14). The state of the art bi-propellant orbital maneuver engine (OME) achieves an I<sub>sp</sub> larger than 315sec<sup>11</sup>. Practically, an OME which realizes a much larger I<sub>sp</sub> may be able to be considered, but in this paper, I<sub>sp</sub> of 315sec is assumed as a conservative value. TPS mass per unit area on vehicle’s nose is evaluated using empirical relations shown in Eq.(15) and Eq.(16). It is known that the application limit of metal TPS is up to around 1200degC. Therefore, in this paper, it is assumed that ceramic tile is used when surface temperature is more than 1000degC, and on the other hand, metal TPS is used for less than 1000degC. Here, surface temperature around nose is assumed to be the same
temperature as stagnation point. Heating rate at stagnation point is estimated by Tauber’s relation\(^4\), and surface temperature is assumed to be the radiation equilibrium temperature.

\[
FUEL\% = 1 - \exp \left[ -\Delta V / \left( g J_0 \right) \right]
\]

(14)
ceramic tile\([kg/m^2]\) \(= 0.0118 \times T_{\text{eq}}[^{0}\text{C}] - 0.8833\)

(15)
metal TPS\([kg/m^2]\) \(= 0.01 \times T_{\text{eq}}[^{0}\text{C}]\)

(16)

2.4. Constraints

Constraints are, 1) structural strength 2) surface temperature and 3) static stability. As shown in Eq.(17) and Eq.(18), a structural strength is evaluated by considering allowable strain of both rigid outer frame and membrane. Assuming that rigid outer frame should sustain the aerodynamic drag as a compressive force acting on outer frame, strain in outer frame can be estimated considering the balance of compressive force and tensile force acting on membrane as shown in Eq.(19).

\[
\varepsilon_{\text{frame}}^{\text{max}} < \varepsilon_{\text{frame}}^{\text{allowable}}
\]

(17)

\[
\varepsilon_{\text{mem}}^{\text{max}} < \varepsilon_{\text{mem}}^{\text{allowable}}
\]

(18)

\[
\varepsilon_{\text{frame}}^{\text{max}} = \frac{D_{\text{frame}}}{S_{\text{frame}}} E_{\text{frame}}
\]

(19)

Constraints on surface temperature on both membrane and outer frame, and static stability are shown in Eq.(20),(21), and (22).

\[
T_{\text{frame}}^{\text{max}} < T_{\text{frame}}^{\text{allowable}}
\]

(20)

\[
T_{\text{mem}}^{\text{max}} < T_{\text{mem}}^{\text{allowable}}
\]

(21)

\[
C_{\text{ae}} < 0
\]

(22)

2.5. Aerodynamics and structural analysis model

The dynamics of membrane aeroshell and flow field are closely related with each other. The aerodynamic force deforms the shape of aeroshell, and deformation of aeroshell changes the flow field in turn. Here, as a fluid-structure interaction analysis model, a particle-based membrane model\(^9\) is used. In this model, membrane is described by a finite number of virtual particles as shown in Fig.3. Deformation of aeroshell can be analyzed by solving the equation of motion for each particle shown in Eq.(23). In Eq.(23), tensile force, aerodynamic force are considered. Surface pressure on aeroshell is estimated by modified Newtonian theory\(^10\). Strain occurred on membrane can be calculated by evaluating the distance of neighboring particles. In this model, the magnitude of deformation of membrane aeroshell is dominated by the non-dimensional parameter called \(C_{\text{ae}}\) as shown in Eq.(24). \(C_{\text{ae}}\) stands for the ratio of aerodynamic force to stiffness of membrane.

\[
m_j \frac{d^2 \vec{r}}{dt^2} = (\vec{F}_t) + (\vec{F}_a)
\]

(23)

\[
C_{\text{ae}} = \frac{L_{\text{mem}} \left( 1/2 \rho V_i^2 \right)}{E_{\text{mem}} h_{\text{mem}}}
\]

(24)

3. Optimization Method

3.1. Multi-objective genetic algorithm

In order to evaluate the performance of flexible aeroshell aerocapture vehicle, it is important to consider its performance from various perspective of view because there are various items to be evaluated such as FUEL\%, TPS\%, \(C_{\text{B}}\), \(L/D\), and structural strength. In this paper, multi-objective genetic algorithm\(^11\) is used as optimization method for following reasons; 1) it is able to evaluate a number of objective functions at once without tuning the weighting factors. 2) it enables designers to obtain diverse solutions and pareto frontier. 3) GA does not require derivative or continuity of functions. Therefore, it is expected to obtain the global solutions, and solutions are not sensitive to initial guess unlike derivative-dependent method. GA is imitating the evolution process of living thing, and it consist of generating initial data set, evaluating the performance for each individual, selection, crossover, and mutation.

3.2. Response surface method\(^12\)

When evaluating the performance of each individual in GA process, it is required to analyze its aerodynamic characteristics using Newtonian or CFD calculation. Especially in the case of flexible aeroshell vehicle, aerodynamics and structural analysis take so much time, because behavior of deformation, thus aerodynamic characteristics, is different according to the dynamic pressure and membrane materials. Therefore, in this paper, aerodynamics and structural analysis is approximated using response surface model. Response surface modeling technique has been originally developed to analyze the results of physical experiments to create empirically based models of the observed response values. According to this modeling, relationship between sampled data \(\mathbf{X}\) and approximated response \(\mathbf{F}\) can be expressed as shown in Eq.(25).

\[
\mathbf{F} = g(\mathbf{X}) + \varepsilon
\]

(25)

where \(\varepsilon\) is random error that is assumed to be normally distributed with mean zero and variance \(\sigma^2\).

In this paper, Kriging interpolation model\(^13\) is applied for following reasons; 1) assumption of order for interpolation curve is not necessary unlike spline or polynomial interpolation. 2) approximation error \(\sigma^2\) of obtained response surface can be calculated. 3) this technique is suitable for the...
case where there is no measurement error in data $X$ like the computer experiment, because it makes a surface that always passes through the sampled point. In this calculation, it is desired to arrange the sample point to cover all the areas of solution space as much as possible. For this reason, Latin Hypercube Sampling method$^{14}$ that is one of the design of experimental method is applied for generating initial sample data set.

### 3.3. Calculation process

Fig.4 shows the calculation process. At first, initial sample data set is generated using Latin Hypercube Sampling method. Secondary, aerodynamics and structural analysis is executed using particle-based membrane model and Newtonian approximation for all sampled initial data set. In this phase, aerodynamic characteristics such as drag coefficient, and structural strength such as strain are obtained for some case of aerodynamic characteristics such as drag coefficient, and approximation for all sampled initial data set. In this phase, using particle-based membrane model and Newtonian Secondary, aerodynamics and structural analysis is executed data set is generated using Latin Hypercube Sampling method. Fig.4 shows the calculation process. At first, initial sample 3.3. Calculation process

![Fig. 4. Calculation process](image)

**Fig. 4. Calculation process**

### 4. Results

Objective functions are, 1) minimize FUEL%, 2) minimize TPS%, 3) minimize STRCT%, 4) minimize $C_r$ and 5) minimize $T_{eq}$. Constraint is posed on only allowable strain of outer frame (Eq.17) in this work. Entry angle is assumed to be -8.0 deg. In Fig.5, obtained solutions can be roughly divided into 3 clusters; 1) solutions that minimize fuel mass and ballistic coefficient. 2) solutions that minimize TPS mass and outer frame mass. 3) solutions that minimize radiation equilibrium temperature and ballistic coefficient.

Fig.5 shows that cluster 1 becomes total optimum, and its total required mass is about 31~37%. This is because, as shown in Fig.6, minimization of fuel mass and the ballistic coefficient is most effectively contributing to the decrease of total required mass. Representative example of cluster 1, which is fuel mass optimum shape, is shown in fig.9.1 (red line) and at the left of fig.9.2. Colored contour shows the pressure coefficient on the membrane. Its drag coefficient is 1.3, and flare radius is 7.2m. As shown in Fig.7, fuel optimum was obtained around $C_B=3.8kg/m^2$. Optimum drag coefficient and optimum flare radius to the minimization of fuel mass were chosen so that the ballistic coefficient becomes around 3.8kg/m².

Fig.5 and Fig.6 show that cluster 2 tends to minimize TPS mass and outer frame mass, but it requires much fuel mass. Representative example of cluster 2, which is outer frame mass optimum shape, is shown in fig.9.1 (blue line) and at the right of fig.9.2. Its drag coefficient is 0.91. Both flare angle and flare radius of “frame mass optimum” are smaller than those of “fuel mass optimum”. This is because “frame mass optimum” solutions tried to decrease the size of the frame by reducing both flare angle and the flare radius. As a result, reducing the flare angle decreases drag coefficient, and reducing the flare radius increases ballistic coefficient, thus fuel mass is increased in this case.

Fig.8 shows that relationship between frame mass and ballistic coefficient forms pareto frontier. Required frame mass for cluster 1 and cluster 2 is about 40kg and 10kg, respectively. Difference of required frame mass is about 30kg, but this value is not so significant when compared with the difference of fuel mass between cluster 1 and cluster 2. Required fuel mass for cluster 1 and cluster 2 is about 30kg and 300kg, respectively. Therefore, it is understood that minimization of fuel mass is much more important than minimization of frame mass for assumed entry condition.

Fig.10 shows the results of trajectory analysis for “fuel mass optimum” solution and “frame mass optimum” solution. Maximum radiation equilibrium temperature for “fuel mass optimum” solution and “frame mass optimum” solution are, 670 degC and 820 degC. Due to the objective function that minimizes the ballistic coefficient and radiation equilibrium temperature, maximum value can be reduced below 1000 degC. As a result, using a lightweight metal TPS becomes possible, and it brings advantages on total mass.
Fig. 5. Ballistic coefficient vs. required total mass.

Fig. 6. Ballistic coefficient vs. required mass.

Fig. 7. Ballistic coefficient vs. fuel mass.

Fig. 8. Ballistic coefficient vs. outer frame mass.

Fig. 9. Representative example of solution for cluster 1 (red) and cluster 2 (blue).

Fuel mass optimum ($C_D=1.3$)  Frame mass optimum ($C_D=0.91$)

Fig. 9.2. Representative deformed example of solution for cluster 1 (left) and cluster 2 (right).
5. Conclusions

In this paper, in order to break through the restriction of aerodynamic heating, which is the biggest problem in aerocapture mission, it is assumed that deployable and flexible aeroshell is installed to the vehicle. Aeroshell shape optimization method using particle based membrane model and GA with response surface modeling is formulated and the calculation is conducted, considering various objective functions such as minimization of fuel mass fraction, minimization of outer frame mass fraction, minimization of ballistic coefficient, minimization of TPS mass fraction. As a preliminary step of the work, axisymmetric aeroshell shape is assumed.

As a result, it is shown that proposed shape optimization method works properly without significant approximation errors in response surface modeling, and thus gives reasonable solutions. Total optimum solution is obtained in the cluster that minimized fuel mass. It is because fuel mass is much more sensitive to ballistic coefficient than any other required mass, for assumed entry condition. And also, tradeoff relationship can be seen between outer frame mass and ballistic coefficient. In this analysis, drag coefficient, flare radius, ballistic coefficient, and radiation equilibrium temperature for total optimum solution was, 1.3, 7.2m, 3.8kg/m² and 670degC, respectively. Comparison of mass saving between “optimum flexible vehicle” and “optimum rigid aerocapture vehicle” should be analyzed as a future work.

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References