Multiobjective Design Exploration of Airplane for Mars Exploration

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Designs of an airplane for Mars exploration are presented. The designs are found with the multiobjective design exploration framework. The result shows that requirements such as flight distance of roughly 100(km), science payload of 200(g), and diameter of entry capsule of roughly 1(m) are reasonable.

火星探査飛行機の多目的設計探査

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多目的進化計算を用いて火星探査飛行機の多目的複合領域設計探査を行った．その結果，航続距離約 **100km**，サイエンスペイロード重量 **200g**，飛行機を収納するのに必要なエントリカプセル内径約 **1m** の機体の設計が可能であることが示された．

Nomenclature

AR = Aspect ratio (=b/c)
b = Main wing span length (m)
c = Main wing average chord length (m)
C_D = Aerodynamic drag coefficient
C_f = Skin friction coefficient
C_L = Aerodynamic lift coefficient
D = Total Drag (N)
D_t = Drag of tail (N)
E = Battery Capacity (J)
FF_t = Form factor of tail wings
FF_f = Form factor of body
$l_{fuselage}$ = Gravity on Mars (m/s^2)
L = Aerodynamic lift (N)
m = Total weight (kg)
$m_{battery}$ = Battery weight (kg)
m_{fix} = Fixed weight (kg)
$m_{fuselage}$ = Fuselage weight (kg)
m_{margin} = Margin weight (kg)
m_{motor} = Motor weight (kg)
m_{prop} = Propeller weight (kg)
m_{rib} = Main wing rib weight (kg)
m_{skin} = Main wing skin weight (kg)
m_{spar} = Main wing spar weight (kg)
m_{tail} = Tail wing weight (kg)
m_{wing} = Main wing weight (kg)
M = Cruise Mach number
n = Load factor
N = Number of Motors for propulsion
P = Power (W)
r = Propeller blade radius (m)
$r_{fuselage}$ = Fuselage radius (m)

R	=	Cruise distance (m)
Q	=	Torque (Nm)
S	=	Main wing area (m^2)
S_f	=	Fuselage area (m^2)
S_t	=	Tail wing area (m^2)
t	=	Main in thickness (m)
t_f	=	Cruise time (s)
T	=	Trust (N)
U	=	Cruise velocity (m/s)
$(x/c)_{max}$	=	Maximum thickness position
α	=	angle of attack (deg)
η_{motor}	=	Motor efficiency
σ	=	Yield stress
ρ	=	Mars air density (kg/m^3)
ρ_s	=	Material density (kg/m^3)
Ω	=	Rotation speed (rps)

I. Introduction

Mars is one of the most interesting planets in the solar system. It's interesting research topics include 1) how was plate tectonics on Mars, 2) how are dust storms on Mars generated, and 3) How the water-path-like geometries are generated. To answer these questions, sampling and measurement of dust in the air, high-resolution data of geological features and residual magnetic field of large area are required.

Current and previous missions to Mars base on ground-based rovers and orbiters. Rovers provide detailed data on the surface but their reach is limited to a small area. On the other hand, orbiting sensors provide large spatial coverage but resolution of the measured data is very low. Thus, idea of Mars exploration by an aircraft has been proposed[1-4]. Among many options for Mars exploration by an aircraft such as balloon, helicopter, and flapping wing, airplane with fixed wing has some advantages over other options such as larger spatial coverage, controllability, reliability, and cost.

In this paper, an airplane for Mars exploration is proposed and its multidisciplinary design optimization result is presented.

II. Formulation of Design Optimization Problem

Here, we propose an airplane for Mars exploration, which fits into a Mars atmosphere entry capsule of 1m diameter. The main wing and the tail are folded to fit into the capsule. We assume the airplane is released from the capsule and is deployed its wing and tail in the air as proposed in the reference 4. The objective of the design optimization problem are

(1) Minimization of the required capsule diameter
(2) Maximization of cruise distance
(3) Minimization of Cruise speed
(4) Maximization of difference between cruise speed and minimum speed.

We assume a rectangular wing planform and Ishii airfoil. Vertical tail volume ratio and horizontal tail volume ratio are 0.035and 0.400. Aspect ratio of the vertical tail wing is 1.50 and NACA0006 is assumed for the tail wings. The fuselage diameter is 0.15 (m). According to the above assumptions, the aircraft is parameterized with

(1) Main wing Aspect ratio
(2) Main wing span length
(3) Cruise speed
(4) Total weight.

To maintain margin for stall angle, cruise angle of attack is constrained to be less than eight degrees. Another constraint is applied to the cruise distance to be less than 100 (km).

Flowchart of the performance evaluation process of an aircraft design is presented in Fig. 1. The process consists of Main wing aerodynamic performance evaluation module, propulsion performance evaluation module, and so on. We assume fixed weight is 1.2 (kg) as shown in Table 1.

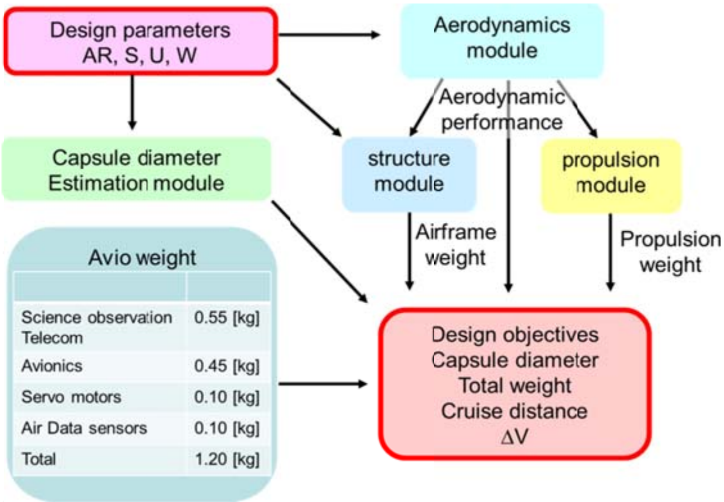

Fig. 1. Flowchart of the performance evaluation process of an aircraft design

Table 1

	Weight
Transmitter	100(g)
Antenna	150(g)
Processor	100(g)
Avionics	450(g)
Servomotors	100(g)
magnetometer	100(g)
Camera	100(g)
total	1200(g)

A. Aerodynamic performance evaluation

Aerodynamic lift coefficient is given by

$$C_L = mg \Big/ \left(\frac{1}{2} \rho U^2 S \right) \tag{1}$$

Corresponding drag and angle of attack is obtained from linear interpolation of a wind tunnel experiment database[5]. According to the reference 6, aerodynamic drag of the horizontal and vertical tail wing and fuselage drag are given by

$$D_t = 0.5 \rho U^2 (2S_t) C_f FF_t \tag{2}$$

$$FF_t = 1.34 M^{0.18} \left(1 + \frac{0.6}{(x/c)_{\max}} \left(\frac{t}{c} \right) + 100 \left(\frac{t}{c} \right)^4 \right) \tag{3}$$

$$D_f = 0.5 \rho U^2 (2S_f) C_f FF_f \tag{4}$$

$$FF_f = \left(1 + \frac{60}{f^3} + \frac{f}{400} \right) \tag{5}$$

$$f = l_{fuselage} / (2 \times r_{fuselage}) \tag{6}$$

B. Propulsion performance evaluation

Required power at cruise is equal to the aerodynamic drag at cruise.

$$T_{cruise} = D \tag{7}$$

The maximum thrust is assumed to be 1.5 times larger than the required thrust at cruise.

$$T_{\max} = 1.5 \times T_{cruise} \tag{8}$$

We assume four motors, each of that has four blades as shown in Fig.2.

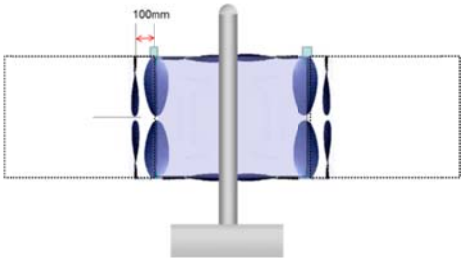

Fig. 2. Propeller blades folded to be fit in the entry capsule.

The propeller blade performance is obtained from the following equations those are derived from the propeller blade design database in which the propeller blades are designed in several design conditions based on the design optimization method of Adkins and Liebeck[7].

$$P = 1.6 \times U \times T \tag{9}$$

$$m_{prop} = 0.1 \times T \tag{10}$$

$$\Omega = (1300 - \frac{25000}{U})\frac{1}{U \times r} \tag{11}$$

$$Q = \frac{P}{2\pi\Omega} \tag{12}$$

Weight of the required motors is obtained from the equation (13) that is derived from the database of the existing motors for utilization in space

$$m_{motor} = 1.25(1.5Q + 0.1)N \tag{13}$$

C. Structural weight estimation

The main wing is consisted with a spar, ribs, and skin.

$$m_{wing} = m_{spar} + m_{rib} + m_{skin} \tag{14}$$

Magnesium (ZK60A) or Aluminum (Al7075) is adopted as wing structure material. Assuming the spanwise lift distribution is constant, weight of the spar can be obtained by the equation (14)

$$m_{spar} = \frac{1.25 \times \rho}{\sigma \times b}\frac{4}{7t}nmgb^2 \tag{15}$$

Rib and skin weight is given by

Magnesium: $m_{spar} = 16.6 \times c \times t \times b$ (16)

Aluminum: $m_{spar} = 25.8 \times c \times t \times b$ (17)

$$m_{skin} = 0.13 \times c \times b \tag{18}$$

Assuming 0.1mm thickness CFRP is used, weight of tail wing and fuselage is given by

$$m_{tail} = \rho_{carbon} \times t_{carbon} \times c_{tail} \times b_{tail} \times 2 \tag{19}$$

$$m_{fuselage} = \rho_{carbon} \times t_{carbon} \times l_{fuselage} \times 2\pi \times r_{fuselage} \tag{20}$$

D. Battery weight and cruise distance estimation

Battery weight is given by

$$m_{battery} = m - m_{fix} - m_{wing} - m_{tail} - m_{fuselage}$$
$$- m_{prop} - m_{motor} - m_{margin} \tag{21}$$

where weight margin is 20% of the total weight. Then the cruise distance is given by

$$E = 424800 \times m_{battery} \tag{22}$$

$$t_f = E \times \eta_{motor} / P_{cruise} \tag{23}$$

$$R = U \times t_f \tag{24}$$

III. Design optimization result

To solve the above design optimization problem, a multiobjective evolutionary computation[7] is used. Population size and number of generations are set to 1000 and 100, respectively. As a result, no optimal solution is obtained. Therefore, we assumed the following

1) Motor weight is 40% lighter than the equation (13).
2) Main wing structure weight is 30% lighter than the equation (14)
3) Energy density of the battery is 30% higher than 118 (Wh/kg)
4) Aerodynamic lift is improved by 20% while aerodynamic drag is kept the same.

Figure 3 is distribution of the Pareto-optimal designs under the above assumptions. This figure shows the tradeoff between the cruise distance maximization and capsule diameter minimization. This figure also shows that a design that can cruise roughly 100 (km) and can be fitted into roughly 1 (m) diameter capsule is feasible. Specifications and weight breakdown of one of the Pareto-optimal design (design A in Fig. 3) are presented in Tables 3 and 4, respectively.

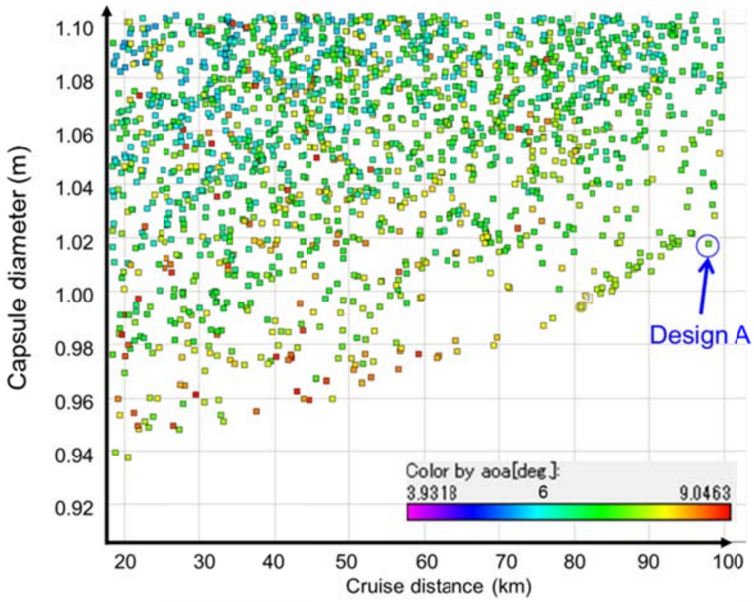

Fig. 3. Distribution of the Pareto-optimal designs.

Table 3. Specifications of the design A

Main wing span length	2.42(m)
Main wing chord length	0.48(m)
Main wing aspect ratio	5.11
Main wing area	1.15(m^2)
Cruise speed	60.0(m/s)
Total weight	4.24(kg)
Cruise distance	95.4(km)
Minimum speed	54.0(m/s)
Cruise Reynolds number (based on the chord)	24,700
C_L	0.643
C_D	0.0439
Main wing lift-to-drag ratio	14.6
Cruise angle of attack	6.79(deg.)
Cruise thrust	1.37(N)
Maximum thrust	2.06(N)
Required torque per motor	0.113(Nm)
Cruise power	132(W)
Maximum power	198(W)

Propeller blade radius	0.318(m)
Propeller rotation speed	46.3(rps)

Table 4. Weight breakdown of the design A

W_{fix}	1.200(kg)	28.3(%)
m_{wing}	0.658(kg)	15.5(%)
$m_{horizontal\_tail}$	0.047(kg)	1.1(%)
$m_{vertical\_tail}$	0.036(kg)	0.9(%)
m_{wing}	0.141(kg)	3.3(%)
m_{prop}	0.206(kg)	4.9(%)
m_{motor}	0.810(kg)	19.1(%)
$m_{battery}$	0.436(kg)	10.3(%)
m_{margin}	0.707(kg)	16.7(%)
Total	4.240(kg)	100(%)

VI. Conclusions

Multidisciplinary design optimization of a Mars exploration airplane is demonstrated. For design optimization, a multiobjective evolutionary computation is used. The design result indicates that a Mars exploration airplane that fit into an entry capsule of 1 m diameter and is capable to fly almost 100 km is with multiobjective design framework is a feasible design if we assume reasonable technology improvement assumption.

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