

PERFORMANCE ESTIMATION OF SOLAR-POWERED LONG ENDURANCE AIRPLANE AT THE STAGE OF PRELIMINARY DESIGN

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Abstract. Analytical investigation was made for the optimal mass definition of solar-powered airplane. Masses of spar, photovoltaic cells, wing skin, powerplant, energy storage and payload as functions of aircraft performance parameters and flight conditions were taken into account. The problem of airplane gross weight minimization was analyzed. Two models of spar mass were investigated. Expressions for the main flight conditions and aircraft parameters as functions of altitude, solar radiation intensity, aircraft construction material properties have been obtained. It was found that some definite maximal mass of payload as part of gross weight exists for which the solar-powered airplane can be designed.

Keywords. HALE, preliminary design, optimization, solar power

1 Introduction

For the present state of the art the design of the solar-powered airplane for multi-day mission is a serious problem because of the moderate value of the solar radiation intensity, the low efficiency and high density of the solar cells, insufficient energy density of the energy storage and some other factors.

There are two ways of increasing the performance of such an aircraft. These are design optimization and flight path optimization. The second one was investigated in [1] and a set of results obtained there will be used below.

Optimal design of airplane can significantly improve its performance. It is also well known that the stage of preliminary design gives the results which significantly affect the aircraft performance. So, the designer must have a set of recommendations and necessary conditions for this stage of design.

2 Necessary relationships

First of all it is necessary to understand if the presence of energy storage is required for the multi-day mission. In other words, is it possible to increase altitude during the daytime and use this potential energy for the flight at the nighttime to be at the initial altitude after 24 hours of flight. If one even not take into account the energy balance it is rather simple to estimate that for the 12 hours of night

flight and the altitude difference of 10 km the vertical velocity must be 0.25 m/s. On the other hand, the minimal gliding vertical velocity is defined as

$$V_Z = \frac{C_D}{C_L^{3/2}} \sqrt{\frac{2mg}{\rho S}}, \quad C_L = \sqrt{3\pi\lambda C_{D0}}$$

For the HALE compatible with “Helios” [2] at 18km altitude ($\rho=0.01 \text{ kg/m}^3$) with $C_{D0}=0.01$, $\lambda=30$, $m=1000\text{kg}$, $S=300\text{m}^2$ the value of V_Z is 2m/s. So, the energy storage devices are required.

Now the main part of energy storages convenient for HALE are of chemical nature (accumulators, fuel cells etc.). So, suppose that the energy storage in this investigation is only of chemical nature. For the convenience let's name the energy storage as “fuel cell”.

It was shown [1] that the optimal path gives about 37% of flight time increase for the altitude difference of more than 10 km compared to the level flight at the „optimal altitude“. The lower altitude difference gives lower increase. It is evident that for the „optimal altitude“ of 18 km the altitude difference of 10 km (i.e. 28 km altitude) is unreal. For the lower altitude difference there is no sufficient difference in energy consumption between the level flight and „optimal“ one. So, for the first stage of evaluation assume that the flight altitude is constant. From this, the air density is constant and mean intensity of solar radiation (averaged by 24 hours) is also constant. Also assume that the total efficiency of powerplant η_P corresponding to the level flight is near its maximum, so it can be assumed as constant at first approximation.

Let's define the efficiency of the energy transformation from solar one to electrical one as η_{PH} and the efficiency of the energy transformation from electrical one to the chemical one in energy storage and backwards to the electricity for the electrical drive as η_{FC} .

Then the „mean intensity“ I of solar radiation after the photoelements is

$$I = I_0 \eta_{PH}$$

where I_0 is the intensity of solar radiation.

It is well known that the minimal energy consumption for the constant altitude corresponds to the maximal value of $\frac{C_D}{C_L^{3/2}}$.

For the constant altitude

$$C_L \rho \frac{V^2}{2} S = mg. \quad (1)$$

The consumed useful power P is

$$P = 4C_{D0} \rho \frac{V^3}{2} S. \quad (2)$$

Consider the case when the solar cells cover all the upper wing surface (other cases will be investigated below). Then

$$I \cdot S \cdot T = \frac{1}{\eta_P} \left(P T_T + P(T - T_T) \frac{1}{\eta_{FC}} \right) = \frac{P}{\eta_P} \left(T_T + (T - T_T) \frac{1}{\eta_{FC}} \right).$$

Define η_0 as

$$\frac{1}{\eta_0} = \frac{1}{\eta_P T} \left(T_T + (T - T_T) \frac{1}{\eta_{FC}} \right),$$

then

$$S = \frac{P}{I \eta_0}.$$

Using (2) one can find that

$$P = 4C_{D0} \rho \frac{V^3}{2} \frac{P}{I \eta_0}.$$

From this,

$$V = \sqrt[3]{\frac{I\eta_0}{2C_{D0}\rho}}.$$

This formula shows that the (optimal) flight velocity depends only on the flight altitude (through I , C_{D0} , ρ), dimensionless parameters of powerplant (through η_0) and profile parameters (through C_{D0}). Using the obtained formula and equation (1) one can find that

$$mg = C_L \frac{P}{2(2C_{D0})^{2/3}} \left(\frac{\rho}{I\eta_0} \right)^{1/3} \quad \text{or} \quad P = \frac{2mg}{C_L} (2C_{D0})^{2/3} \left(\frac{I\eta_0}{\rho} \right)^{1/3},$$

$$S = \frac{2mg}{\sqrt{3\pi\lambda C_{D0}} \rho^{1/3}} \left(\frac{2C_{D0}}{I\eta_0} \right)^{2/3}.$$

So, the parameter mg/S depends on the same characteristics as V and on the wing aspect ratio.

For the further analysis assume that the total mass of aircraft is the sum of constant mass m_0 (sum of payload and other masses that are not changed with the change of design), wing mass m_W , photovoltaic cells (PV) mass m_{PH} , fuel cell mass m_{FC} and electric drive mass m_D . So, the masses balance can be written as

$$m = m_0 + m_W + m_{PH} + m_{FC} + m_D,$$

Below for the simplicity we will name m_0 as “payload mass” but keep in mind that this is the sum of all constant masses.

Assume that wing mass consists of spar mass m_S and wing skin mass.

Spar mass can be evaluated at least in two ways.

1. From the condition of maximal mechanical tension [3]

$$m_S = 0.15 \frac{mn\rho_S g}{2\bar{c}\sigma} S^{1/2} \lambda^{3/2} = 0.15 \frac{mn\rho_S g}{2\bar{c}\sigma} \sqrt{\frac{2mg}{C_L}} \frac{1}{\rho^{1/6}} \left(\frac{2C_{D0}}{I\eta_0} \right)^{1/3} \lambda^{3/2}$$

where n is total safety factor, ρ_S is the density of spar material, $\bar{c}$ is relative thickness of wing profile, σ – maximal mechanical tension in spar.

As

$$C_L = \sqrt{3\pi C_{D0}\lambda}$$

then

$$m_S = 0.15 \frac{m^{3/2} n \rho_S g}{2\bar{c}\sigma} \sqrt{\frac{2g}{3\pi}} \frac{1}{\rho^{1/6} C_{D0}^{1/12}} \left(\frac{2}{I\eta_0} \right)^{1/3} \lambda^{5/4}.$$

2. From the condition of optimal bending [4]

$$m_S = 0.164 mg \left(\frac{\rho_S}{E} \right)^{2/3} \frac{\lambda^{3/2} (2m)^{1/3}}{\bar{c}^{4/3} \rho^{1/9}} \frac{2^{2/9} C_{D0}^{1/18}}{(3\pi)^{1/6}} \frac{1}{(I\eta_0)^{2/9}}$$

where E is Young modulus of spar.

Wing skin mass is assumed to be proportional to the surface covered by the skin.

Fuel cells must store the energy for the flight at the nighttime T_T , (time with the absence of solar radiation) so the fuel cell mass m_{FC} must be:

$$m_{FC} = \frac{\alpha}{\eta_P} P T_T = \frac{\alpha}{\eta_P} \frac{2mg}{\sqrt{3\pi C_{D0}\lambda}} (2C_{D0})^{2/3} \left(\frac{I\eta_0}{\rho} \right)^{1/3} T_T.$$

where α is proportionality coefficient (fuel cell mass per stored energy).

If we assume that the PVs cover all the upper wing surface, then

$$m_{PH} = \rho_{ph} S = \rho_{ph} \frac{2mg}{\sqrt{3\pi C_{D0}\lambda} \rho^{1/3}} \left(\frac{2C_{D0}}{I\eta_0} \right)^{2/3},$$

where ρ_{ph} – is surface density of photoelements.

Assuming that the electrical drive mass is proportional to its power one can obtain

$$m_D = \alpha_D \frac{P}{\eta_P} = \frac{\alpha_D}{\eta_P} \frac{2mg}{\sqrt{3\pi C_{D0} \lambda}} (2C_{D0})^{2/3} \left(\frac{I\eta_0}{\rho} \right)^{1/3}.$$

where α_D is proportionality coefficient (drive mass per drive power).

3 Aircrafts with PVs on all the wing surface

Assume that the PVs cover all the upper surface of the wing. As for the wing skin, it can cover only the lower surface or both upper and lower surfaces. In any case the mass of wing skin is proportional to the wing area. As the PV mass is also proportional to the wing area we assume in this section that ρ_{PH} is the total mass of PVs and skin per unit wing surface.

3.1 First model of spar

Mass balance equation becomes

$$m = m_0 + 0.15 \frac{m^{3/2} n \rho_s g}{2\bar{c}\sigma} \sqrt{\frac{2g}{3\pi}} \frac{C_{D0}^{1/12}}{\rho^{1/6}} \left(\frac{2}{I\eta_0} \right)^{1/3} \lambda^{5/4} + \rho_{ph} \frac{2mg}{\sqrt{3\pi C_{D0} \lambda} \rho^{1/3}} \left(\frac{2C_{D0}}{I\eta_0} \right)^{2/3} + \frac{\alpha}{\eta_P} \frac{2mg}{\sqrt{3\pi C_{D0} \lambda}} (2C_{D0})^{2/3} \left(\frac{I\eta_0}{\rho} \right)^{1/3} T_T + \frac{\alpha_D}{\eta_P} \frac{2mg}{\sqrt{3\pi C_{D0} \lambda}} (2C_{D0})^{2/3} \left(\frac{I\eta_0}{\rho} \right)^{1/3}. \quad (3)$$

This equation is not linear with respect to m . So, it can not be easily solved to obtain the mass m as function of design parameters and flight conditions. On the other hand, one can solve this equation numerically.

One of the design tasks is to make the aircraft „not heavy enough“ or „light enough“. In a set of design cases it is necessary to minimize the aircraft mass. From equation (3) one can see that the design parameter that is strongly affect the mass is the wing aspect ratio. From this, it is useful to analyze the value of m minimization with respect to λ .

Setting the derivative of implicit function $m(\lambda)$ with respect to λ equal to zero one can obtain the optimal aspect ratio as

$$\lambda = \left(\frac{0.5 \left(\rho_{ph} \frac{2g}{\sqrt{3\pi} \rho^{1/3}} \left(\frac{2}{I\eta_0} \right)^{2/3} + \left(\frac{\alpha}{\eta_P} T_T + \frac{\alpha_D}{\eta_P} \right) \frac{2g}{\sqrt{3\pi}} (2)^{2/3} \left(\frac{I\eta_0}{\rho} \right)^{1/3} \right)}{\frac{5}{4} 0.15 \frac{m^{1/2} n \rho_s g}{2\bar{c}\sigma} \sqrt{\frac{2g}{3\pi}} \frac{1}{\rho^{1/6}} \left(\frac{2}{I\eta_0} \right)^{1/3}} C_{D0}^{1/12} \right)^{4/7}. \quad (4)$$

Substituting (4) to (3) one can obtain

$$m = m_0 + m C_{D0}^{1/12} \sqrt{\frac{2g}{3\pi}} \left(\frac{2}{I\eta_0} \right)^{1/3} \left(\left(\frac{2}{5} \right)^{5/7} + \left(\frac{5}{2} \right)^{2/7} \right) \left(0.15 \frac{m^{1/2} n \rho_s g}{2\bar{c}\sigma \rho^{1/6}} \right)^{2/7} \times \left(\sqrt{2g} \left(\frac{2C_{D0}}{\rho} \right)^{1/3} \left(\rho_{ph} \left(\frac{2}{I\eta_0} \right)^{1/3} + \left(\frac{\alpha}{\eta_P} T_T + \frac{\alpha_D}{\eta_P} \right) (I\eta_0)^{2/3} \right) \right)^{5/7}. \quad (5)$$

This equation have the solution not for all values of m_0 . Maximal m_0 can be found from the following: function $f(m)$

$$f(m) = m_0 + m C_{D0}^{13/84} \sqrt{\frac{2g}{3\pi}} \left(\frac{2}{I\eta_0} \right)^{1/3} \left(\left(\frac{2}{5} \right)^{5/7} + \left(\frac{5}{2} \right)^{2/7} \right) \left(0.15 \frac{m^{1/2} n \rho_s g}{2\bar{c} \sigma \rho^{1/6}} \right)^{2/7} \times \\ \times \left(\sqrt{\frac{2g}{\rho}} \left(\frac{2}{\rho} \right)^{1/3} \left(\rho_{ph} \left(\frac{2}{I\eta_0} \right)^{1/3} + \left(\frac{\alpha}{\eta_P} T_T + \frac{\alpha_D}{\eta_P} \right) (I\eta_0)^{2/3} \right) \right)^{5/7} - m$$

have the minimum with respect to m . If this minimum of f is below the zero then there are two solutions of the equation

$$f(m)=0.$$

If this minimum is equal to zero then only one solution exists. If the minimum is above zero then there is no solution. So, the maximal possible value of m_0 corresponds to the case of one solution. For this case

$$m = \left(\frac{8}{7} C_{D0}^{1/7} \sqrt{\frac{2g}{3\pi}} \left(\frac{2}{I\eta_0} \right)^{1/3} \left(\left(\frac{2}{5} \right)^{5/7} + \left(\frac{5}{2} \right)^{2/7} \right) \right)^{-7} \left(0.15 \frac{n \rho_s g}{2\bar{c} \sigma \rho^{1/6}} \right)^{-2} \times \\ \times \left(\sqrt{\frac{2g}{\rho}} \left(\frac{2}{\rho} \right)^{1/3} \left(\rho_{ph} \left(\frac{2}{I\eta_0} \right)^{1/3} + \left(\frac{\alpha}{\eta_P} T_T + \frac{\alpha_D}{\eta_P} \right) (I\eta_0)^{2/3} \right) \right)^{-5}. \quad (6)$$

Substituting this expression to (5) one can found that in this case the value of m_0 is 1/8 of m :

$$m_0 = \frac{m}{8} = \frac{(3\pi)^{7/2} \cdot 5^5 \cdot \rho^2 \bar{c}^2 \left(\frac{\sigma}{n \rho_s} \right)^2 (I\eta_0)^4}{2^{30} \cdot 0.15^2 g^8 C_{D0} \left(\rho_{ph} + \left(\alpha T_T + \alpha_D \right) \frac{I\eta_0}{\eta_P} \right)^5}, \quad (7)$$

So, the maximal value of payload can be found from (7) for the defined flight conditions, material properties and parameters of powerplant. Corresponding value of aspect ratio is

$$\lambda = 2^{28/3} C_{D0}^{1/3} g^2 \frac{\left(\rho_{ph} + \left(\frac{\alpha}{\eta_P} T_T + \frac{\alpha_D}{\eta_P} \right) I\eta_0 \right)^2}{75\pi \rho^{2/3} (I\eta_0)^{4/3}}.$$

It should be mentioned that formula (7) gives the maximal possible value of m_0 for the model investigated. If m_0 is less than maximal then m_0/m will be higher than 1/8.

Expression (7) also shows the influence of aircraft parameters on the value of maximal payload.

3.2 Second model of spar

In the same way as it was made above

$$m = m_0 + 0.164mg \left(\frac{\rho_s}{E} \right)^{2/3} \frac{\lambda^{3/2} (2m)^{1/3}}{\bar{c}^{4/3} \rho^{1/9}} \frac{2^{2/9} C_{D0}^{1/18}}{(3\pi)^{1/6}} \frac{1}{(I\eta_0)^{2/9}} + \\ + \frac{2mg}{\sqrt{3\pi C_{D0} \lambda}} (2C_{D0})^{2/3} \left(\frac{\rho_{ph}}{\rho^{1/3}} \left(\frac{1}{I\eta_0} \right)^{2/3} + \frac{\alpha}{\eta_P} \left(\frac{I\eta_0}{\rho} \right)^{1/3} T_T + \frac{\alpha_D}{\eta_P} \left(\frac{I\eta_0}{\rho} \right)^{1/3} \right).$$

Optimal value of aspect ratio is

$$\lambda = \sqrt{\frac{C_{D0}^{1/9} \bar{c}^{4/3} \rho^{1/9} 2^{10/9} (I\eta_0)^{5/9} \left(\frac{\rho_{ph}}{I\eta_0 \rho^{1/3}} + \left(\frac{\alpha}{\eta_P} T_T + \frac{\alpha_D}{\eta_P} \right) \left(\frac{1}{\rho} \right)^{1/3} \right)}{3 \cdot 0.164 \left(\frac{\rho_S}{E} \right)^{2/3} m^{1/3}}}$$

Then the maximal value of aircraft mass, corresponding to the maximal payload, is

$$m = 0.0715 \frac{\bar{c}^4 \rho^{10/3} (E/\rho_S)^2 (I\eta_0)^{20/3}}{C_{D0}^{5/3} g^{12} \left(\rho_{ph} + (\alpha T_T + \alpha_D) \frac{I\eta_0}{\eta_P} \right)^9}. \quad (8)$$

Corresponding value of payload is

$$m_0 = m/13.$$

3.3 Discussion

One can see that the value of m_0/m corresponding to the maximal m_0 depends on the spar mass model. More accurately, this ratio depends on the power for the aspect ratio in the expression for spar mass. The absolute value of m_0 also vary with the model of spar. This means that one must calculate the value of m_0 according to all the models and chose the smallest one.

Formulas (7) and (8) can help to understand the influence of aircraft parameters on m_0 . For example, influence of strength parameters σ/ρ_S and E/ρ_S is strong enough. Increase of these parameters twice will give the increase in m_0 by four times.

It should be also mentioned that there can be other mathematical models of spar mass, and the results for those models will differ from the obtained ones.

4 Aircrafts with PVs covering not on all the wing surface

Note that functions (7) and (8) have extremums (minimums) with respect to I . In other words, while increasing I the maximal payload mass at first increases and then decreases. This situation seems illogical. It can be explained as follows: starting with some value of I the optimal construction corresponds to the case when not all the wing surface is covered by the solar cells. Moreover, this value of I is lower then corresponding to the extremums of (7) and (8).

Let's investigate these cases. Assume that the wing skin covers all the wing surface and define the total wing skin surface density (upper and lower) as ρ_{WS} . The surface density of solar cells we denote ρ_{ph} as before.

According to (2), if the area of solar cells S_{ph} is

$$S_{ph} = \frac{P}{I\eta_0},$$

Then the mass of PV cells is

$$m_{PH} = \rho_{ph} S_{ph} = \rho_{ph} \frac{P}{I\eta_0} = 4C_{D0} \rho \frac{V^3}{2I\eta_0} S \rho_{ph}.$$

Similarly, mass of fuel cells is

$$m_{FC} = \frac{\alpha}{\eta_P} P T_T = 4 \frac{\alpha}{\eta_P} T_T C_{D0} \rho \frac{V^3}{2} S,$$

Mass of drive is

$$m_D = \alpha_D \frac{P}{\eta_P} = 4\alpha_D C_{D0} \rho \frac{V^3}{2\eta_P} S.$$

From (1) one obtain

$$S = \frac{2mg}{C_L \rho V^2} = \frac{2mg}{\sqrt{3\pi C_{D0}} \lambda \rho V^2}$$

For the first model of spar mass

$$m_s = 0.15 \frac{mn \rho_s g}{2\bar{c}\sigma} S^{1/2} \lambda^{3/2} = 0.15 \frac{m^{1.5} n \rho_s g^{1.5}}{\sqrt{2\bar{c}} \sigma V} \left(\frac{1}{\sqrt{3\pi C_{D0}} \rho} \right)^{1/2} \lambda^{5/4}.$$

Mass balance equation looks like

$$m = m_0 + 0.15 \frac{m^{1.5} n \rho_s g^{1.5}}{\sqrt{2\bar{c}} \sigma V} \left(\frac{1}{\sqrt{3\pi C_{D0}} \rho} \right)^{1/2} \lambda^{5/4} + \left(4C_{D0} \rho \frac{V^3}{2I\eta_0} \rho_{ph} + 4 \frac{\alpha}{\eta_P} T_T C_{D0} \rho \frac{V^3}{2} + 4\alpha_D C_{D0} \rho \frac{V^3}{2\eta_P} + \rho_{WS} \right) \frac{2mg}{\sqrt{3\pi C_{D0}} \lambda \rho V^2}$$

As above, one can obtain the condition of minimal m as:

$$V^3 = \frac{2\rho_{WS}}{\left(\frac{\rho_{ph}}{I\eta_0} + \frac{\alpha}{\eta_P} T_T + \frac{\alpha_D}{\eta_P} \right) C_{D0} \rho}.$$

As

$$S_{ph} = 4C_{D0} \rho \frac{V^3}{2I\eta_0} S,$$

then

$$\frac{S_{ph}}{S} = \frac{4\rho_{WS}}{\rho_{ph} + \frac{\alpha}{\eta_P} T_T I\eta_0 + \frac{\alpha_D}{\eta_P} I\eta_0}. \quad (9)$$

If this expression is lower then unity, then for the optimal aircraft mass the solar cells must cover not all the wing surface. This corresponds to

$$I\eta_0 > \frac{4\rho_{WS} - \rho_{ph}}{\frac{\alpha}{\eta_P} T_T + \frac{\alpha_D}{\eta_P}}.$$

If (9) is higher than unity, then either there must be some additional surfaces with solar cells or the wing surface must be increased that corresponds to the case investigated in the previous chapters. From the last formula one can make the conclusion that in the case of

$$4\rho_{WS} < \rho_{ph}$$

the solar cells must cover all the wing surface for any value of $I\eta_0$.

Similarly, for the second model of spar mass,

$$m_s = \lambda^{5/3} S^{1/3} \sqrt[3]{\frac{2\rho_s}{448E^2\bar{c}^4}} mg^{2/3} = \lambda^{3/2} \left(\frac{2}{\sqrt{3\pi C_{D0}} \rho V^2} \right)^{1/3} \sqrt[3]{\frac{2\rho_s}{448E^2\bar{c}^4}} m^{4/3} g$$

For this case

$$V^3 = \frac{8}{7} \frac{\rho_{WS}}{\left(\frac{\rho_{ph}}{I\eta_0} + \frac{\alpha}{\eta_P} T_T + \frac{\alpha_D}{\eta_P} \right) C_{D0} \rho},$$

and for

$$I\eta_0 > \frac{\frac{16}{7}\rho_{ws} - \rho_{ph}}{\frac{\alpha}{\eta_P}T_T + \frac{\alpha_D}{\eta_P}}$$

the solar cells will not cover all the wing surface.

5 Results comparison with existing aircrafts

It is useful to compare the results obtained with the data of existing solar aircrafts. Table 1 gives the data available from open sources. It should be noted that in these data the term “payload” is only the payload without any other constant masses, so it is lower than m_0 .

	Payload	TO weight m ,	Payload/ m *100%
Zephyr	2.5 kg [5] (2kg [6])	53 kg [5] (30 kg [6])	5% (6.7%[6])
Global Observer GO-1 [7]	400 lb (172 kg)	4000 lb	10%
Global Observer GO-2 [6], [7]	1000 lb	9870 lb	10.1%
Helios [6]	100 kg	825 kg	12%
Pathfinder Plus [6]	11.3 kg	218 kg	5.1%

Table 1: Data for a set of solar-powered aircrafts.

One can see from Table 1 that the ratio of Payload/ m is lower than 12.5% (or 1/8).

Unfortunately there is no information about the properties of materials used in the construction of mentioned aircrafts so one can't compare the predicted and real masses of planes.

6 Conclusion

A set of useful results were obtained above.

1. For the solar powered aircrafts some value of maximal payload exists. This value depends on the design parameters, flight altitude and characteristics of the materials used. The specific shape of this relationship depends on the mathematical model (and corresponding style of aircraft design) of masses estimation used in the investigation.
2. Relationships for the aircraft parameters corresponding to the case of maximal payload were obtained. They enable to evaluate the main parameters of solar aircrafts and corresponding masses of the main parts of construction.
3. For the some conditions the optimal construction corresponds to the case when the PV cells cover not all the surface of the wing. The possibility of this case depends on the characteristics of the PV cells and the wing skin and the intensity of the solar radiation and the efficiency of powerplant elements.
4. Formulas can help to understand the influence of flight conditions, materials characteristics and efficiencies of powerplant components on the value of maximal payload.
5. The results obtained can be used not only for the aircrafts flying over the Earth but also for those that are planned to fly over the other planets.

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