# Estimation of main parameters for solar-powered long endurance airplane at the preliminary design stage

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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. 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

## 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.
Now a set of aircraft with solar cells exist. Some of them are presented in Figures 1-3.

Figure 1: Solar Impulse aircraft [1] (left) and  Zephyr aircraft [2] (right)

Figure 2 : SoLong aircraft (left) [3] and Global Observer aircraft [4] (right)

Figure 3 : Helios aircraft [4]

One can see that all of them are light enough and have the wing with high aspect ratio. This facts prove that the design of such aircrafts is a difficult task.
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 [5] 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 that 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 $V_Z$ is defined as

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

where $C_D$ is drag coefficient, $C_L$ – lift coefficient, $m$ – aircraft mass, $g$ – gravity acceleration, $\rho$ – air density, $S$ – wing area, $\lambda$ – wing aspect ratio, $C_{D0}$ – drag coefficient at zero lift.
For the HALE aircraft compatible with “Helios” [4] at 18km altitude ($\rho$=0.01 kg/m$^3$) with $C_{D0}$=0.01, $\lambda$=30, $m$=1000kg, $S$=300m$^2$ the value of $V_Z$ is about 1.5m/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 [5] 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 $\eta_P$ 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 $\eta_{PH}$ 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 $\eta_{FC}$.

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 $C_D/(C_L^{1.5})$ that corresponds to the conditions

$$C_L=\sqrt{3\pi\lambda C_{D0}}\ ,\ C_D=4C_{D0}.$$

For the constant altitude

$$C_L\rho\frac{V^2}{2}S=mg\ , \tag{1}$$

where $V$ is flight velocity.
The consumed useful power $P$ is

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

Consider the case when the solar cells cover all the upper wing surface (other cases will be investigated below). Then for 24 hour flight

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

where $T$ – total time (24 hour), $T_T$ – amount of time when the sun is below the horizon (night time), $\eta_P$ – efficiency of powerplant from energy source to propeller.
Define $\eta_0$ as

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

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 $I$, $C_{D0}$, $\rho$), dimensionless parameters of powerplant (through $\eta_0$) and wing parameters (through $C_{D0}$). Using the obtained formula and equation (1) one can find that

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

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

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 name $m_0$ as "payload mass" but one must 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 from the condition of maximal mechanical tension [6]

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

where $n$ is total safety factor, $\rho_S$ is the density of spar material, $\hat{c}$ is relative thickness of wing profile, $\sigma$ – maximal permissible mechanical tension in 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 night time $T_T$, (time with the absence of solar radiation) so the fuel cell mass $m_{FC}$ must be:

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

where $\alpha$ 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 $\rho_{ph}$ – is surface density of photoelements.
Assuming that the electrical drive mass is proportional to its power one can obtain

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

where $\alpha_D$ 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 either 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 $\rho_{PH}$ is the total mass of PVs and skin per unit wing surface.

Mass balance equation becomes

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

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 $\lambda$.
Let’s analyse the function *f* (see (3))

$$f = m_0 + 0.15\frac{m^{3/2} n\rho_S g}{2\bar{c}\sigma}\sqrt{\frac{2g}{3\pi}}\frac{C_{D0}{}^{1/12}}{\rho^{1/6}}\left(\frac{2}{I\eta_0}\right)^{1/3}\lambda^{5/4} + \rho_{ph}\frac{2mg}{\sqrt{3\pi C_{D0}\lambda}\rho^{1/3}}\left(\frac{2C_{D0}}{I\eta_0}\right)^{2/3} + $$
$$+\frac{\alpha}{\eta_P}\frac{2mg}{\sqrt{3\pi C_{D0}\lambda}}(2C_{D0})^{2/3}\left(\frac{I\eta_0}{\rho}\right)^{1/3} T_T + \frac{\alpha_D}{\eta_P}\frac{2mg}{\sqrt{3\pi C_{D0}\lambda}}(2C_{D0})^{2/3}\left(\frac{I\eta_0}{\rho}\right)^{1/3} - m = 0.$$

As

$$\frac{df}{d\lambda} = \frac{\partial f}{\partial \lambda} + \frac{\partial f}{\partial m}\frac{dm}{\partial \lambda} = 0\,,$$

then setting the derivative of implicit function $m(\lambda)$ with respect to $\lambda$ equal to zero one can obtain the optimal aspect ratio from

$$\frac{df}{d\lambda} == 0$$

In the following form

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

The additional condition for this procedure is

$$\frac{\partial f}{\partial m} \neq 0\,.$$

With the help of (3) one can obtain this condition in the form of

$$m_0 \neq 0.5 m_S\,.$$

We will check this condition below.
Substituting (4) to (3) one can obtain

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

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

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

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

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

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

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

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

$$\lambda = 2^{28/3}C_{D0}{}^{1/3}g^2\frac{\left(\rho_{ph}+\left(\frac{\alpha}{\eta_P}T_T+\frac{\alpha_D}{\eta_P}\right)I\eta_0\right)^2}{75\pi\rho^{2/3}(I\eta_0)^{4/3}}. \tag{8}$$

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 available value 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.
It should be mentioned that the condition of

$$\frac{\partial f}{\partial m} \neq 0 \tag{9}$$

is valid for all the λ except the (8). But in the case of (8) the condition $f$=0 is valid only in the one point corresponding to (8) so the expression (9) have no sense.
Also one can conclude that for (8) the condition for spar mass is

$$m_S = 2m_0.$$

Formulas (7) and (8) can help to understand the influence of aircraft parameters on $m_0$. For example, influence of strength parameter $\sigma/\rho_S$ 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 on all the wing surface. General case

The results obtained in previous chapter and among them the expression $m_0=m/8$ strongly depends on the models used. (For example, the above investigation was made for the case of equal tension in spar. But also one can use the condition of optimal spar bending as in [7]). Of cause, if masses of aircraft components depend on $m$ and λ in another way the result is not the same.
Let the relationship (3) is of the following form

$$m = m_0 + Am^{\delta}\lambda^{\beta} + Bm^{\gamma}$$

where $A$, $B$, β, γ, δ – some constants.
Making the same computations as in previous chapter one can obtain

$$m_0 = \frac{\gamma(\delta-1)}{\delta\gamma-\beta}m\,.$$

Corresponding value of $m_0$ is

$$m_0 = \left[\left(\frac{\gamma}{\delta\gamma-\beta}\right)^{\gamma-\beta}\left(\frac{-\gamma}{\beta}\right)^{\beta}\frac{B^{\beta}}{A^{\gamma}}\right]^{\frac{1}{\gamma(\delta-1)}}\frac{\gamma(\delta-1)}{\gamma\delta-\beta}\,.$$

## 5. 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 seams 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 $\rho_{WS}$. The surface density of solar cells we denote $\rho_{ph}$ 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

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

and 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 can obtain

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

For the 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

$$\frac{S_{ph}}{S} = \frac{4\rho_{WS}}{\rho_{ph} + \frac{\alpha}{\eta_P} T_T I\eta_0 + \frac{\alpha_D}{\eta_P} I\eta_0} . \qquad (10)$$

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 (10) 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 $I\eta_0$.

## 6. 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 $m_0$.

Table 1: Data for the set of aircrafts

| | Payload | TO weight *m*, | Payload/*m**100% |
|---|---|---|---|
| Zephyr | 2.5 kg [8] (2kg [9]) | 53 kg [8] (30 kg [9]) | 5% (6.7%[9]) |
| Global Observer GO-1 [10] | 400 lb (172 kg) | 4000 lb | 10% |
| Global Observer GO-2 [9], [10] | 1000 lb | 9870 lb | 10.1% |
| Helios [9] | 100 kg | 825 kg | 12% |
| Pathfinder Plus [9] | 11.3 kg | 218 kg | 5.1% |
| SolarImpulse [11] | | | ~12% |

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.

## 7. 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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