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OPTIMUM SPACECRAFT LIFT-OFF FROM THE MOON'S SURFACE

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SUMMARY

The problem of optimum landing of a spacecraft from the AMS orbit on the Moon's surface is investigated in [1]. Presented here are the results of the analysis of problem's numerical solution and, in particular, the material for comparison of optimum programs with theoretical dependences [2], previously obtained by the authors.

The present work is dedicated to the investigation of motion dynamics over the subsequent stage of lunar expedition which is the spacecraft's lift-off from lunar surface into the circular AMS orbit. These two maneuvers have much in common from the standpoint of the application of optimization methods. Reference to work [1] was made in a series of cases, thus making a detailed expose unnecessary.

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* *

The problem was solved in the following setup:

1) The motion takes place in the central Newtonian gravitation field of the Moon.

2) The control of the magnitude and direction of the thrust is inertialess.

3) The outflow velocity <u>c</u> is constant.

The equations of optimum motion* coincide with Eq. (1) of work [1], while the boundary conditions at the times $t = 0$ (start

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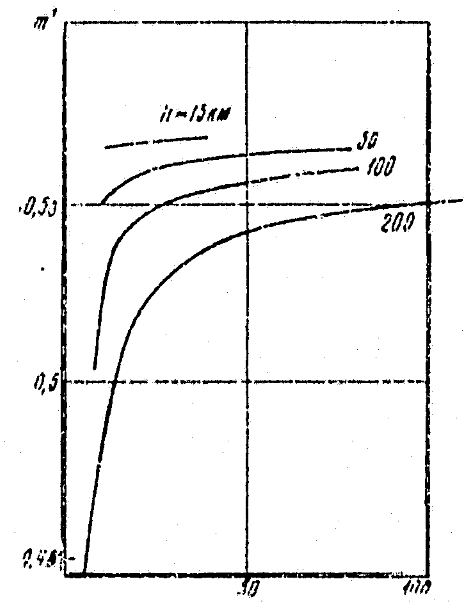
* Descartes' inertial system of coordinates was used with origin at the center of attraction; the denotation of phase coordinates are standard.

from the Moon) and t = T (escape time to the designated orbit) have the form

$$u = 0,\ v = 0,\ x = 0,\ y = y^0,\ m = 1 \quad (t = 0)$$
$$(1)$$
$$u = u^1,\ v = v^1,\ x = x^1,\ y = y^1,\ p_m = -1 \quad (t = T)$$



ϕ degrees
Fig.1

The time T is not fixed and is chosen optimum. At the initial moment of time the values p_u, p_v, p_x, p_y, p_m of the conjugate variables of the variation problem are unknown.

Taking into account the integral $\mathcal{H} = 0$ (see [1]), there remain four unknown parameters of the boundary value problem. For the solution of the latter the Newtonian method was applied with modification described in [3].

The height of circular orbit's allotment varied from 15 to 200 km, and the selenocentric angular range ψ of the lift-off sector from 5° to 180°. The spacecraft's dimensionless final mass[1] (related to the mass M(0) on the Moon's surface) is a monotonically increasing function of total range ψ (Fig.1), as in the case of landing. Values close to maximum are attained at ψ = = 30° - 40°.

From the comparison of analogous results of the landing problem (see [1]), one may see that the relative fuel consumption at lift-off is greater than at landing. The dependences of the final mass on thrust load are plotted in Fig.2 for various altitudes of orbits. The maximum values are practically attained at

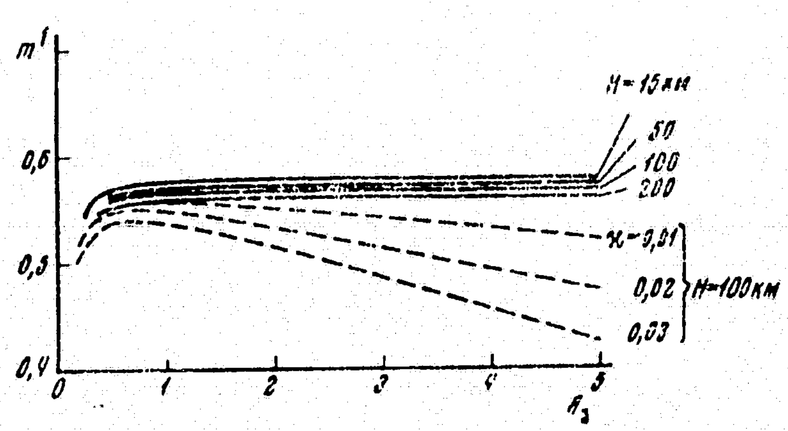


Fig.2

$A_E = P_{max}/g_E = 1.0$ (g_E is the gravitational acceleration on the Earth's surface), which agrees well with the conclusions of the work [4].

Taking into account the engine's weight, the optimum thrust load is still less.

Given by the dashed lines in Fig.2, is the dependence of the relatively effective payload m^1 in the case, when the engine's weight depends linearly on engine's thrust $G_{eng} = \kappa P_{max}$. The optimum value of the thrust load is $A_E^{opt} = 0.6-0.7$ With lesser power supply the relative final mass for the landing is somewhat greater than for the take-off, but starting with the value $A_E^{opt} = 1.0-1.5$, the relative mass consumption for both maneuvers practically coincide. The results of computation for the initial relative thrust load $A_E = 0.4$ are presented below.

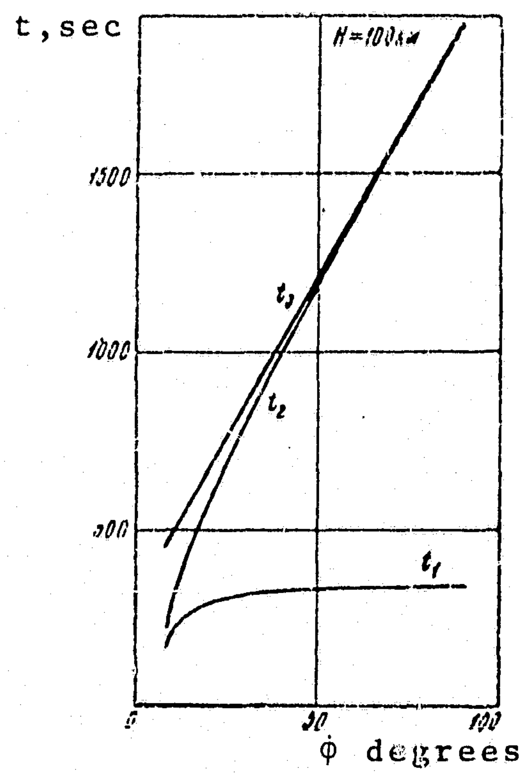


Fig.3

The trajectories are composed of three sectors: two of maximum thrust at the beginning and the end of the take-off and a passive section between them. As for the landing, such condition of thrust magnitude control is optimum for all investigated values of parameters.

The active section $[0, t_1]$, at the beginning of the motion is fundamental as regards the energy consumption. In it the engine is accelerated to a velocity close to circular. The second active sector $[t_2, t_3]$ is too small from the standpoint of time and characteristic velocity (Fig.4). In it the correction is performed in order to obtain the circular orbit parameters. With minor angular ranges at take-off analogous division of functions of active sectors is absent. The total flight time $T = t$ is almost a linear function of the distance.

With the increase of ψ optimum orientation program of the vector of thrust in the initial system of coordinates linked with the horizon at the point of escape into a circular orbit, approaches the linear function of time (Fig.3). The averaged value of angular pitch velocity $\dot{\vartheta}$ decreases the total range and approaches the angular velocity of Moon's satellite at zero height.

The second active portion is small, so that the orientation angle in it has no time to vary somewhat significantly. The averaged direction of thrust is close to lateral. As the orbit height increases, the deflection angle upward from the transverse increases.

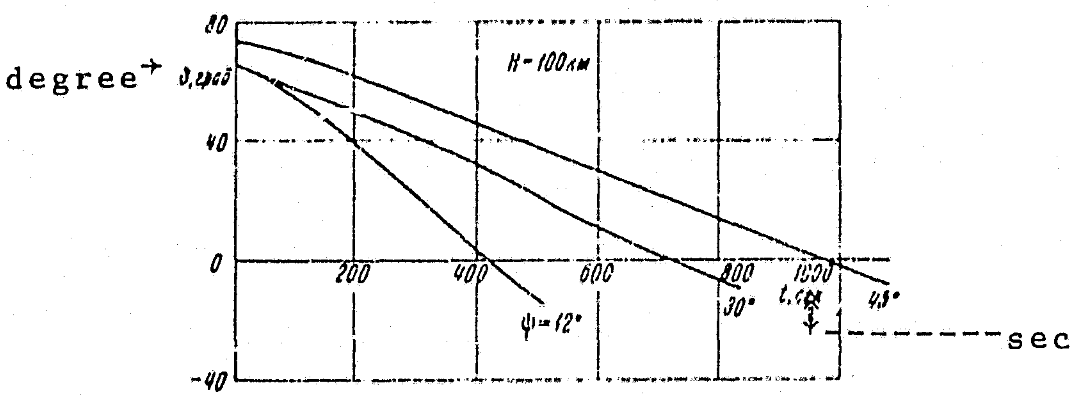


Fig.4

Shown in Fig.5, are the $\vec{p}$-trajectories by whose character the form of optimum orientation program is determined. With minor angular take-off ranges (ψ < 20°) $\vec{p}$-trajectores are close to elliptical. In this case the angular rotation velocity of

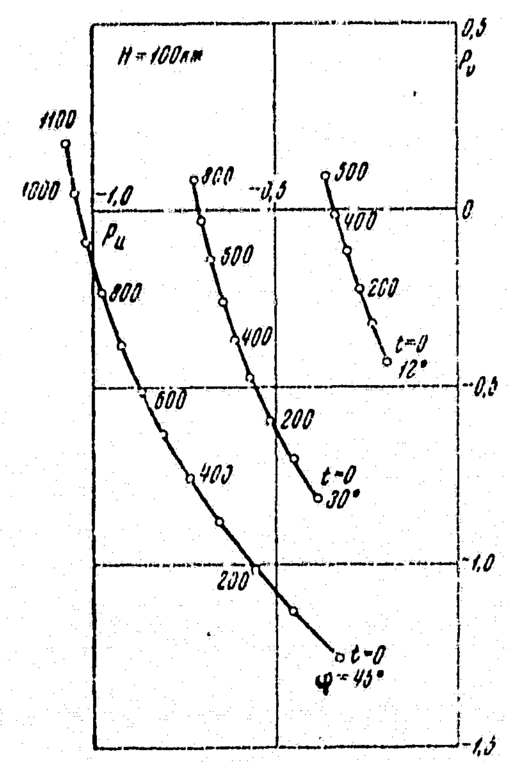


Fig.5

the radius-vector is not constant and the program $\vartheta(t)$ is nonlinear. With the increase of angular range $\vec{p}$-trajectories become circular. Moreover, the rotation velocity of radius-vector $\vec{p}$ (i.e. the angular velocity of pitching) approaches the angular velocity of an AMS, while the optimum program $\dot{\theta}(t)$ approaches the linear function of time.

* * * THE END * * *

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