CONQUEST OF INTERPLANETARY SPACE /537

Author's Preface

The major portion of the present work was written in 1916, after which it was supplemented and radically revised three times. The author hopes that he has succeeded in presenting the problem of conquering the solar system not so much in the form of theoretical principles, leaving their development and practical application to the science and technology of the future, as in the form of a plan of attack, which, even if not detailed, is outlined with concrete figures that are fully realizable today with current technology once we have performed experiments not presenting any particular difficulties. And this realization, from the first preliminary experiments beginning and ending with flights to the moon, would require, to the extent that this can be adduced beforehand, less material media than the equipment for several of our largest warships.

As to the existence of Engineer Tsiolkovskiy's paper on the same subject, the author later became aware of the work and has only recently had the opportunity to familiarize himself with a portion of the article entitled "The Investigation of Cosmic Space by Reactive Devices," published in the journal Vestnik Vozdukhoplavaniya (Aeronautics Bulletin) for 1911, and I am convinced of Engineer Tsiolkovskiy's prerogative in the resolution of many of the fundamental issues. However, paragraphs that clearly no longer represent new ideas have not been rejected from the cited article, on the one hand not to disrupt the integral unity of the presentation and in order not to refer the interested reader to the now very rare and difficult to locate issues of Vestnik Vozdukhoplavaniya, on the other hand so that the theoretical postulates and formulas themselves, just used in a slightly different way, will occasionally put the entire problem in a new light. Similarly, the author has also not had the opportunity to become acquainted either with the foreign literature on this problem or even with the second part of Engineer Tsiolkovskiy's article, which was published in the 1912 issue of the same journal.

Many of the formulas and almost all of the numerical data cited in this work are given with simplifications and rounding off, on occasion even rather coarsely; the reason for this is that we are still lacking the experimental /538
material needed for a detailed treatment of the problem, so that there is no point in our bothering with hundredths as long as we can still be sure of accuracy to tenths; the sole objective of some of the computations in the present work is to give a notion as to the order of magnitude of the physical quantities with which we will have to deal, and as to the general nature of their variation, since their exact values cannot possibly be computed until suitable experimental investigations have been performed. For similar reasons, there are no construction plans or drawings; the general design principles can be readily enunciated, but we cannot develop the particulars: therefore, any drawing purporting to contain specific design features would hinder rather than aid the scientific concept.

In view of the relative newness of the subject, the author has been compelled to introduce quite a few of his own terms, which are replaced almost throughout, for the sake of brevity, by symbol designations, the application of which is such that the very same symbols denoting numerical values of physical quantities in the formulas and computations replace the corresponding conventional physical or special terms of the present work in the text of the latter. To facilitate matters for the reader, a separate list of all symbol designations used repeatedly in different parts of the work is given at the end of the article. Wherever special notation is not given, the symbols denote the physical quantities expressed in absolute (c.g.s.) units.

June 1925 Yu. Kondratyuk

Second Author's Preface

I am going to discuss a fundamental general problem of this work, which is completely untouched in the original presentation, i.e., the problem of the expected consequences for mankind when it emerges into interplanetary space.

Prof. Tsiolkovskiy, the pioneer researcher in this field, views its significance in the fact that mankind will be able to colonize the vast reaches of the solar system and, when the sun cools, to travel by rockets for the population of still uncooled worlds.

Such possibilities, of course, are not by any means excluded, but all this is the prognostication for a very distant future, some of it extremely distant. There is no doubt that for a long time yet the investment of resources and improvement of living conditions on our planet will be more profitable than colonization beyond it; it should not be forgotten that only a very small part of the total surface of our planet has been populated and exploited. We will examine the problem of man's departure into interplanetary space from a more /539
"present-day" point of view: What specifically can we expect in the near future, at most ten years, after the first flight from earth?

Without dwelling on more or less groundless fantasies, we can expect the following:

1) An incontrovertible enormous enrichment of our scientific knowledge with corresponding ramifications in technology.

2) The possible, more or less probable, although not certain, enrichment of our technology with valuable materials which might be found on other bodies in the solar system and which are absent or are extremely rare on the earth's surface.

3) Other possible riches from the solar system, which we are partially unable to foresee right now and which may or may not exist, as, for example, the results of intercourse with the presumed organic world of Mars.

4) The unquestionable possibility for man to acquire resources which he can use to radically improve the conditions of existence on earth, to exercise reclamation on a grandiose scale, realizing undertakings in the not too distant future of such magnitude as, for example, altering the climate of entire continents.

I am speaking, of course, of nothing other than utilization of the untapped stores or energy in solar radiation, which is so difficult under conditions present on the earth's surface, which make it less profitable than the exploitation of fuels, water, and wind and which, on the other hand, will be immeasurably more profitable in outer space, where there is no atmosphere or apparent gravity. It is the possibility of beginning in the immediate future to improve the economy of our planet that should be regarded as the chief aspect of tremendous importance to us in mastering the outer space of the solar system.

Reflecting on the impressive achievements of science and technology in recent years and, in spite of ourselves, wondering why the problem of interplanetary travel has not been solved in practice before now, the problem, essentially, in comparison with other advances, is not so difficult if approached scientifically and without the eyes prematurely widened in astonishment and fear, nor is it so grandiose in terms of the technological means required; at the same time, however, it is of such vast significance that it will only fail to come to fruition through lack of courage and initiative on the one hand, and through failure to grasp the practical importance of the problem on the other hand. If the cost of this task, given the same difficulty, were expressed more explicitly in dollars, so as not to be overwhelmed by its extraordinary aspect, the Americans would truly have already licked the problem, instead of, like the Germans, just having conducted some very preliminary experiments directed, insofar as one may judge from our newspaper reports, along not altogether proper lines of attack.

In 1921, I arrived at a very unanticipated solution to the problem of /540 setting up a continuous line of communication from earth into space and back again, the materialization of which requires only the use of a rocket such as the one discussed in this book; in 1926, I arrived at an analogous solution of the problem of a rocket developing the initial 1500 to 2000 m/sec of its escape velocity without the expenditure of charge and, at the same time, without a colossal artillery cannon or superpowerful engines, or for that matter any kind of giant apparatus. The indicated chapters are not included in the present book; they are already too near the working project of mastering outer space, too near to be published, not knowing how and by whom these data will be used.

In conclusion, I must express by profound gratitude to Prof. V. P. Vetchinkin, the editor of the present work and its first critic.

October 1928 Yu. Kondratyuk

I. Facts Concerning the Rocket.

Principal Notation

A mechanical definition of the rocket as a reactive device is the following: "A vehicle which, ejecting particles of its own mass with a certain velocity, itself develops a velocity in the opposite direction due to their effect." We will adopt the following terms and notation with respect to rockets:

M Mass of the rocket at a given instant;

M_0 Initial mass of the rocket;

M_f Mass of the rocket on completion of its operation as such; the "final mass";

M_{i_o} Mass of the rocket at the instant it passes the initial point of a given segment (i) of its trajectory;

M_{i_f} Mass of the rocket at the instant it passes the final point of a given segment (i) of its trajectory.

The "exhaust" is the aggregate of particles ejected by the rocket, the reaction of which imparts velocity to the rocket.

u is the "exhaust velocity," or velocity of the ejected particles relative to the rocket at the instant it begins to move independently from it, unless the force of gravity on the rocket is assumed inconsequential. We will assume that u is constant in any given interval of time. If different exhaust particles leave the rocket with different velocities, then we will assume for u an average velocity such that it can replace all of the actual, disparate particle velocities without altering the sum total of their reactive effect on the /541
rocket; this will be the velocity of the center of gravity of the exhaust after an infinitesimal time interval and is equal to

$$u = \frac{\Sigma\,(\alpha \cdot u_\alpha)}{\Sigma\alpha}, \tag{1}$$

where α and u_α are the masses and velocities, respectively, of the individual particles. It is not too difficult to see that, given the same sum of the kinetic energies, equal to $\frac{1}{2}\sum(\alpha u_\alpha^{\ 2})$, u will be a maximum (eq. (1)) when the velocities of all the individual particles are equal.

j_0 is the "intrinsic rocket acceleration," equal to the acceleration that the rocket would have if only one force of exhaust reaction were acting on it. It is readily grasped that $j_0 = \frac{dM}{Mdt}\,u$, where dM is the mass of the ejected particles.

μ is the "rocket charge," or the part of the rocket mass intended for consumption, i.e., for conversion to "exhaust."

n is the "flight load rating" $= M_0/M_f$, whence

$$M_0 = M_f \cdot n; \tag{2}$$

n_i is the "segmental load rating," i.e., the same ratio for a given segment of the flight; $n_i = M_{i_o}/M_{i_f}$, whence

$$M_{i_o} = M_{i_f} \cdot n_i \ . \tag{2a}$$

It is readily seen that always

$$\left.\begin{array}{c} M_0 = M_к + \mu;\ M_{i_0} = M_{i_f} + \mu_i;\ \mu = M_f\,(n-1); \\ \mu_i = M_{i_f}\,(n_i - 1); \\ n = n_a \cdot n_b \cdot n_c \ldots n_i \ldots n_z, \end{array}\right\} \tag{2b)-(3}$$

where a, b, c,..., i,..., z are all segments of the rocket trajectory.

W is the "rocket velocity," equal to $\int_0^{t_f} j_0\, dt,$ where t_f is the burnout time. In other words, the "rocket velocity" is the velocity that the rocket would develop when not subjected to any external forces and with acceleration imparted to it consistently in the same direction.

We use the symbol j_0, therefore, to denote just the absolute magnitude of the acceleration irrespective of its direction.

W_i is the "segmental rocket velocity," equal to $\int_{t_1}^{t_2} j_0 dt,$ /542

consistent with the preceding notation, where t_1 and t_2 are the times at which the beginning and end of a given segment are passed.

II. Formula for the Load Rating

(Ratio of Initial to Final Rocket Mass)

The basic formula of rocket theory, relating the quantities W, u, and n, has already been advanced by Engineer Tsiolkovskiy (but in a slightly different form):

$$\frac{M_{i_0}}{M_{i_к}} = n_i = e^{\frac{W_i}{u}} \tag{4}$$

where e is the base of the natural logarithms. We use the subscript i here to denote any segment of the rocket trajectory or the entire trajectory.

We proceed now with the elementary derivation of this formula. Let a rocket with initial mass M_0 eject a train of particles of its own mass at a velocity u in the same direction, where the particles have masses M_0/k_0, M_1/k_1, $M_2/k_2, \ldots, M_i/k_i$, where M_o, $M_1, \ldots, M_i$ are its masses after each ejection. We then have

$$\frac{M_1}{M_0} = \left(1 - \frac{1}{k_0}\right); \quad \frac{M_2}{M_1} = \left(1 - \frac{1}{k_1}\right); \ldots \frac{M_{i+1}}{M_i} = \left(1 - \frac{1}{k_i}\right).$$

Multiplying all of these equations together, we obtain

$$\frac{M_K}{M_0} = \left(1 - \frac{1}{k_0}\right)\left(1 - \frac{1}{k_1}\right)\left(1 - \frac{1}{k_2}\right)\ldots\left(1 - \frac{1}{k_i}\right)\ldots;$$

the limit of the latter expression for $k_0, k_1, k_2, \ldots, k_i, \ldots = \infty$ will be $e^{-\sum \frac{1}{k_i}}$, or, as we are permitted to write, $e^{-u\sum \frac{1}{k_i} : u}$ Since the velocities of mutually repulsing bodies are distributed as the inverse of the masses, during each injection the rocket will acquire a velocity equal respectively to $u\frac{1}{k_0}$, $u\frac{1}{k_1}, \ldots, u\frac{1}{k_i}, \ldots$ The total rocket velocity acquired, therefore, will be $W = u\sum\frac{1}{k_i}$ Substituting into our expression

$$\frac{M_K}{M_0} = e^{-u\Sigma\frac{1}{k_i}:u}$$

in place of the velocity $u\sum\frac{1}{k_i}$ the symbol W, we obtain the inverse of /543
equation (4). Equation (4) enables us to determine M_0 and μ when M_f, W, and u are given.

We see from equation (4) that when the ratio W_i/u is near zero, n_i becomes almost equal to unity, where $(n_i - 1)$, which difference is proportional to μ_i (eq. (2b)), varies approximately as the velocity ratio W_i/u. Consequently, for $W_i/u \ll 1$, the amount of required charge is insignificant (note 1, Commentary), being approximately proportional to the required rocket velocity and inversely proportional to the exhaust velocity.

For $W_i/u > 1$, n_i will grow as an exponential function relative to W_i (note 2) and can rapidly attain values that would render unfeasible the practical realization of man's flight into interplanetary space. If, for example, to execute the flight, W_i were required to be ten times the value of u that we could attain in actual practice, n_i would acquire a value of about 22,000; for M_f = 1000 kg, we would need the astounding value of 22,000 metric tons for the total mass of the rocket. The practicability of flight in interplanetary space and landing on other bodies in the solar system depends, therefore, on how large we can make u and how small a value of W we can get away with for executing the flight.

III. Exhaust Velocity. The Chemical Material

The store of energy needed to impart the exhaust velocity can be carried with the rocket in very diverse forms, but of all these only latent chemical energy in a compound of some of the lightest and most active elements and the energy of decomposition is in proper proportion to the mass of the substance containing them for a value of u sufficient for the practical realization of flight to be obtained. We have far too scarce reserves of radium, and even then we are not able to control the liberation of its latent energy, which takes place much too slowly for our purposes; of all the possible types of "rocket," therefore, we must discuss the "rocket" in the common sense of the word, i.e., the thermochemical rocket, which is invested with the one very powerful advantage that in it the latent energy can be converted into exhaust kinetic energy in very large quantities and with high efficiency for a relatively low weight and simplicity on the part of the accessory equipment subservient to this conversion.

Still another special type of rocket is possible, one which utilizes energy from without, i.e., from the light of the sun. In practice, however, this method of operating a rocket is inapplicable at the present time, or almost inapplicable because of purely technical difficulties: /544

1) The difficulty of imparting, even with the required energy reserve available, a larger velocity to the exhaust particles than could be produced by the expansion of intensely heated gases in a thermochemical rocket.

2) The difficulty of constructing the necessary reflectors with a suitable area to mass ratio such that the solar energy entrapped by it will be enough to impart sufficient exhaust velocity with sufficient intensity ($d\dot{M}/Mdt$, see above). In view of these difficulties, for now we will shelve the idea of a rocket that functions on the energy of solar radiation.

The transformation of the heat of chemical reaction into kinetic energy for exhaust is based on the expansion of gases, hence gases are needed in the composition of the rocket exhaust; however, we are not limited in our choice of chemical composition of the exhaust to gaseous compounds alone. The rocket can function properly if just a portion of the exhaust is gaseous, the other being made up of denser substances atomized in the gas. The gases, expanding in the rocket tube due to their expansibility and thus acquiring velocity, will entrain particles of the denser substances, at the same time picking up heat from the latter to replace the heat lost in expansion (note 3). In order for this process to go to completion with the greatest efficiency, we need: 1) the most complete entrainment possible of the denser particles by the gases; (2) the most complete transfer possible of heat from the dense particles to the gases. Either of these things will require sufficiently fine and uniform dispersion of the dense particles in the gas and a sufficient interval of time during which they will be in mutual contact, i.e., a sufficiently long rocket tube. The problem of what should be the degree of dispersion, the tube length, and percentage content of dense particles in the exhaust for satisfactory operation of the rocket can only be solved by a series of suitable experiments.

The choice of materials for the charge, therefore, amounts primarily to a choice of a suitable group of materials such that the amount of heat liberated in chemical reaction between its components will be a maximum with respect to each gram of evolved compound, so that we will be able to acquire maximum u. If it should turn out that the reaction products liquify or solidify at temperatures still far from absolute zero, thus losing the expansibility that we require, we would have to supplement the selected group of materials with another, for which the products of reaction between its elements would retain the gaseous state at lower temperatures and thus be capable of converting the liberated heat more completely into kinetic energy. In the simplest case, we might use in place of a second gaseous group the lightest of gases, hydrogen.

The table below lists chemical compounds having the highest heat value per gram mass.

Composition of exhaust	Combustible material		kcal/g	u, m/sec	n_1 (W_1= 22,370)	n_2 (W_2= 14,460)	/545
CO_2			2.1	4200	205	31	
"			2.7	4760	110	21	
H_2O			3.7	5570	55	13	
"			4.4	6080	40	11	
CO_2+2H_2O	CH_4	H_2O liquid	3.3	5250	60	15	
" "	"		3.9	5720	49	12	
CO_2+H_2O	Hydrocarbons (petroleum)		2.6	4670	120	22	
" "	The same		3.2	5160	73	16	
CO_2+H_2O+9N_2	Hydrocarbons (petroleum) and liquid air		0.8	2590	5600	250	

Composition of exhaust	Combustible material		kcal/g	u, m/sec	n_1 (W_1= 22,370)	n_2 (W_2= 14,460)
$2CO_2+H_2O$	C_2H_2		3.0	5020	86	18
" "			3.5	5420	62	14
H_2O			3.2	5160	73	16
"		H_2O steam	3.9	5720	49	12
CO_2+2H_2O			3.1	5070	77	17
" "			3.7	5570	55	13
CO_2+H_2O	Hydrocarbons (Petroleum)		2.5	4580	130	23
" "	The same		3.1	5070	77	17
$CO_2+H_2O+9N_2$	The same and liquid air		0.7	2430	9000	300
$2CO_2+H_2O$	C_2H_2		2.9	4940	95	20
" "	"		3.4	5340	65	15
Li_2O			4.6	6220	36	10
"			5.0	6480	32	9.3
LiOH			4.6	6220	36	10
"			5.1	6540	30	9.1
B_2O_3			4.5	6150	38	11
"			5.0	6480	32	9.3
$B(OH)_3$			4.2	5940	43	12
"			5.0	6480	32	9.3
$B(OH)_3$	BH_3		?	?	?	?
Al_2O_3			3.8	5650	52	13
"			4.1	5870	45	12
$Al(OH)_3$			3.7	5570	55	13
"			4.2	5940	43	12

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Composition of exhaust	Combustible material		kcal/g	u, m/sec	n_1 (W_1= 22,370)	n_2 (W_2= 14,460)
SiO_2			3.6	5500	58	14
"			4.0	5800	47	13
MgO			3.4	5340	65	15
"			3.7	5570	55	13
$Mg(OH)_2$			3.7	5570	55	13
"			4.1	5870	45	12
SiO_2+2H_2O	SiH_4		?	?	?	?

The first column of figures gives the heat of combination in kilocalories per gram, minus the latent heats of vaporization in the case of liquid O_2, O_3, H_2, CH_4, C_2H_2, and liquid air.

The second column contains the exhaust velocities in meters per second corresponding to the data in the first column, i.e., the velocities that would be obtained per gram mass if its kinetic energy were equal to the heat energy indicated in the first column.

The third column gives the values of n_1 for

$$W_1 = 22,370 \text{ m/sec} = 2 \cdot 11,185 \text{ m/sec (note 4).}$$

The fourth column gives the values of n_2 for

$$W_2 = 14,460 \text{ m/sec} = \left(2 - \sqrt{\frac{1}{2}}\right) \cdot 11,185 \text{ m/sec,}$$

calculated according to equation (4) in correspondence with the data of the second column. The velocity values 22,370 m/sec and 14,460 m/sec will be discussed below in section VI, IX, and XII.

Inasmuch as the element oxygen is a part of every one of the compounds in which we are interested, each of the compounds is listed twice, corresponding to the two forms of oxygen, O_2 and O_3, where the upper line gives the data for oxygen, the lower line gives the data for ozone, which has a considerably higher store of energy. From here on, we will refer to the active part of the compounds in terms of their nonoxygen components.

We see from the table that the maximum heat effect is given by the lithium and boron compounds. The use of lithium in the rocket charge is dismissed from the outset, because it is incomparably more expensive than boron, while only slightly surpassing it in heat value. Then come the rest of the compounds, in almost systematic order: aluminum, silicon, magnesium, and hydrogen, if we are concerned with the liquefaction of steam, but in reckoning with the gaseous state of water the hydrogen group is somewhat inferior to the metallic group, whereas in reckoning with the liquid state of water with the simultaneous /547 application of ozone, it is slightly superior. Then come the hydrocarbon compounds which yield a mixture of carbon dioxide with water: marsh gas (methane), acetylene, and petroleum; a still smaller effect is given by the pure carbonic radical, and, finally, the compound consisting of petroleum and air. In view of the cost economy of petroleum, which is more convenient for our purposes and yields greater efficiency, the pure carbonic compound is rejected from the outset. As for the hydrogen radical, the question of its application must be left open because of the difficulties in handling it and the expense of liquid hydrogen. It is very likely that the use of the silicon and boron hydride compounds will be better in every respect, especially since we cannot hope to condense the steam in the rocket tube, i.e., to utilize its latent heat of vaporization, in the period during which the rocket develops the greater part of its velocity, when we cannot count on j_0 and dM/dt being made as small as we like, and in all liklihood it cannot be realized in general, since the liquefaction of steam would require a hundred thousandfold expansion or more between exit from the combustion chamber and emergence from the tube. The use of the metal or boron compounds requires the simultaneous application of the hydrogen, boron hydride, or one of the hydrocarbon compounds, otherwise there will not be any excess hydrogen. If minimal cost is the criterion for formulating the charge, the guiding principle must be as follows: application of the most economical compounds[1] for the parts of the charge that are consumed first, and changeover to compounds with a higher heat value ($q/m = \max$) for the parts of the charged to be expended later. According to this principle and the table above, the rocket charge should consist of compounds in the following order:

I. Petroleum; if liquid oxygen proves to be far more expensive than liquid air, the petroleum-plus-air compound should take precedence over this compound.

II. Marsh gas (methane); if it turns out to be possible to obtain low-cost and nonhazardous liquid acetylene, the acetylene compound should take precedence.

III. Hydrogen; its use is contingent upon the cost of manufacturing and storing liquid hydrogen; it is quite possible that the hydrogen compound will

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[1] In other words, compounds yielding the cheapest reactive effect; the reaction cost is defined by the product $C \cdot q^{-\frac{1}{2}} \cdot m^{\frac{1}{2}}$, where C is the cost of the charge, m is its weight, and q is its heat value.

prove unsuitable and unprofitable and be replaced by the combined use of the marsh gas (methane), metallic (Al, Si, Mg), and silicohydride compounds.

IV. Boron; used together with the hydrogen or borohydride compounds.

The application of the metallic compounds will be discussed further in sections V and IV below. /548

Whether to use ozone and what compound to start with will depend on how cheaply and, mainly, how safely liquid ozone can be prepared; the use of the hydrogen compound will also depend largely on this factor, since the difference between oxygen and ozone becomes the most pronounced in this case.

The compounds O_2, O_3, H_2, CH_4, C_2H_2, SiH_4, and BH_3 can only be carried on the rocket, of course, in liquid form, since in the gaseous state they would require tanks of enormous volume and weight; boron must be carried as an amorphous powder, which, is pulverized in the combustion chamber by a jet of hydrogen or marsh gas, or is mixed in with petroleum before being sent to the combustion chamber. B, Si, and H_2 can be carried in the form of BH_3, B_2H_3, and SiH_4, as well as in the form of boro- and silicohydrides. The author regrets that he has not had the opportunity to locate thermochemical data pertaining to these extremely interesting compounds. The metals can be used in the molten state or, as with boron, in powder form.

It is difficult to estimate beforehand the efficiency of the rocket, i.e., the relative amount of heat that will be converted into exhaust kinetic energy; it will depend primarily on the degree of expansion of the gases in the rocket tube, i.e., on the ratio of the initial to final expansibilities. The latter will depend, however, on the ratio of the exhaust mass (dM/dt) to the exit cross section of the tube and, in addition, cannot be less than the expansibility of the surrounding atmosphere. The efficiency of the rocket will therefore be greater during those periods of the flight when the rocket is a free cosmic body in empty space, when the quantitities j_0 and dM/dt will be sufficient no matter how small, and the efficiency will be lower in those periods of flight when the rocket is within the boundaries of an atmosphere of substantial density and when it will require a j_0 at least as great as some critical value (sections VI and VIII). Under the above conditions, the efficiency will clearly be on the order of 50 to 75%. To raise the efficiency, we will have to have as large an initial pressure as possible (in the combustion chamber) and as low a final pressure as possible (in the end of the rocket tube) (note 5); in order to do this without increasing the tube cross section or the overall cross section of the rocket and concomitant atmospheric drag, it may prove more expedient to replace one exit tube by several distributed in sequence and emerging at a small angle with respect to the lateral surface of the rocket; the stern of the rocket in this case could be made with a pointed, streamlined shape. These exit tubes could be fed from one or more combustion chambers, whatever proves to be best from the design standpoint. Due to the incomplete utilization of the heat of chemical reaction, the actual values of u will be less than indicated in the table. If the efficiency were equal to 50 to 75% /549 the actual value of u would be equal to about 3/4 or 7/8, respectively, of its computed value, in correspondence with which n would have a value of $n^{4/3}$ or $n^{8/7}$ with respect to the computed values.

IV. The Combustion Process, Construction of the Combustion Chamber and Exit Tube

A very important problem concerns the temperatures in the combustion-chamber and in the exit tube. If complete combination of the exhaust components could be immediately realized, the temperature in the combustion chamber would rise to

$$T = 208\ Qm, \tag{5}$$

where Q (kcal/g) is the mean heating capacity per gram of the compound, m is the mean molecular weight of the exhaust, assuming it is gaseous. For solid and liquid products, the temperature would be even higher. The occurrence of molecular dissociation at high temperatures, however, does not permit the chemical reaction to go to completion at once; at some temperature (above 3000°C) chemical equilibrium sets in for all reactions, after which further reaction will only be possible with increasing heat loss by the gases as they expand in the exit tube. Consequently, the thermal energy of the reactions will be realized primarily, not by an adiabatic process but by a more nearly isothermal process. An adiabatic process sets in when the gases, after expanding in the tube, lose so much heat that the reactions can continue to completion without raising the temperature of the mixture to a point where its components suffer appreciable dissociation. These effects are of considerable importance in the construction of the rocket; to evolve the same amount of heat of combination with gradual combustion we must have a larger ratio of final to initial volume occupied by the gases, i.e., larger dimensions on the part of the exit tube. On the other hand, in the combustion chamber and in the beginning of the exit tube, we will have a lower temperature than that which would exist with complete combustion in the chamber. It is clear from equation (5) that, given a certain limiting temperature in the combustion chamber based on design considerations, we will obtain far more complete initial combustion and a faster total combustion for compounds with a lower molecular weight. From this point of view, the most suitable compounds are those with H_2, CH_4, C_2H_2, petroleum, and Li, somewhat less appropriate are SiH_4, BH_3, and the least suitable are the purely metallic compounds with Si, Mg, boron, and, especially, aluminum.

It will be necessary to do the following in constructing the combustion chamber and exit tube: those surfaces which will be exposed to temperatures higher than can be tolerated by the most refractory materials must be metal (copper or one of the high-melting metals like chromium or vanadium) and subjected to powerful cooling from the outside by the liquid gases being fed into the combustion chamber. It does not appear possible to calculate this cooling until suitable experiments have been carried out on the amount of heat that will be acquired by the surfaces of the chamber through radiation and thermal conduction from the hot mixture. All other surfaces can be internally lined with sufficiently refractory materials, insulating them as far as possible from the /550 exterior construction, which could, if necessary, be moderately cooled. If it proves inconvenient or unfeasible to bring the temperature in the combustion chamber and in the beginning of the tube down to a point where appreciable dissociation of the exhaust components will not occur, we can maintain it artificially at some specified level by not feeding one of the charge materials (metals or oxygen) into the combustion chamber right away, instead just part of it, delivering the remainder of it in different parts of the tube as heat is lost from the originally specified misture.

V. The Proportional Passive Load

In the passive mass of the rocket, i.e., the mass μ not belonging to the charge, we can distinguish two essentially different parts:

1) The absolute passive mass m, which includes the personnel and all that is necessary for them to live and perform their allotted tasks, as well as for operation and safe descent to the earth's surface when the rocket ceases to function as such.

2) The proportional passive mass m_1 of all objects subservient to the functioning of the rocket, including: a) the tanks containing the charge, b) the combustion chambers, c) the exit tube, d) the instruments and machines for mixing the charge substance in the combustion chamber, e) all parts connecting the objects in the first four categories and reinforcing the overall construction of the rocket. This part of the passive mass is called the "proportional passive load," in view of the fact that, according to design precepts, its mass must in general be approximately proportional to the mass of the charge with which it is associated, as long as the latter does not exceed some value; for large values of μ the ratio m_1/μ will increase.

The starting point for construction of the rocket is its preestablished mass m, which, once given, must be matched by μ and m_1; m remains constant throughout the flight; u is gradually used up, and m_1 may vary, hopefully, in /551
correspondence with the diminishing masses of the charge (μ) and the exhaust products (dM/dt).

We designate the ratio $m_1/u = q$ and postulate that we have one and the same irremovable passive load m_1 functioning the whole time. Then

$$m_1 = \mu q; \; M_f = m + m_1 = m + \mu q.$$

Substituting this value of M_f into equation (2b), we obtain

$$\mu = (m + q\mu)(n - 1),$$

whence

$$\mu = \frac{m(n-1)}{1 - q(n-1)}, \tag{6}$$

whereas for $m_1 = 0$ we would have $\mu = m(n-1)$.

We see from equation (6) that as long as $q \ll \frac{1}{n-1}$, we will obtain values for μ only slightly different from those which would occur for $m_1 = 0$ (note 6), but with increasing q the mass μ will increase, going to infinity at $q = \frac{1}{n-1}$, which means that it is theoretically impossible to build a rocket for such data. But the practical feasibility is less stringent; for $q = \frac{1}{2(n-1)}$, we would obtain already twice the charge (note 7). But for the mass of the rocket not to increase too much due to the presence of the mass m_1 and the need for imparting to it a velocity consistent with m, it is desirable to have the approximate relation

$$q \ll \frac{1}{5(n_i - 1)} \quad \text{(note 8)}, \tag{7}$$

where n_i is the load rating for that segment along which the same m_1 functions without change and at the completion of which it can be rejected so as not to burden the rocket unnecessarily with its surplus mass, after which another unit m_1 begins to function, of smaller dimensions and smaller mass, in correspondence with the reduced masses of the charge and exhaust. Both sides of the inequality (7) are not identically amenable to our efforts to change them; the quantity q is determined by the degree of technological perfection in building the objects m_1 and may be larger or smaller, depending on a variety of conditions, but it does have a certain rigorous minimum, which, with the materials at our disposal and with the present development of engineering design, we are not now in a position to surmount. We can decrease the quantity n_i at will down to unity (note 9), by dividing the trajectory of the rocket into a larger number of segments with a smaller W_i for each. The number of segments and, accord- /552 ingly, the number of units m_1 is determined as a function of the relative amount of expended charge that we find convenient for the use of one invariant unit m_1; specifically this number should be equal to $\log n : \log n_i$, where n_i is the load rating of each segment of the trajectory. Should we wish to use a one-unit system for the entire flight, we would obtain too insignificant an absolute limit for the quantity q. The theoretical minimum W necessary for completing the flight purely by rocket means is equal, as we shall see below, to 22,370 m/sec; the corresponding values of n_1, calculated on the assumption of 100% efficiency on the part of the rocket, are given in the third column of figures in the table on pages 64 to 66.

Considering all of the sources of energy loss and imperfections, we can say that the actual value of n for W = 22,370 m/sec will be at least 100, and if we wish to cheaply formulate the charge and partially use hydrocarbons, it will be more than 100. Consequently, for q = 1/99, the mass of the charge according to equation (6) would already be infinite, for q = 1/200 it would have doubled, whereas $\mu/200$ is a very compact quantity, more correctly impossible for the mass of the total unit m_1. Even if we let $W = 14{,}460$ m/sec and, accordingly, assume $n_2 = 20$ (page 65), we still obtain twice the charge with the difficult-to-realize ratio $m_1 = \mu/40$. In practice, therefore, the optimum system will be a two-unit one for the machines and instruments and a three-unit version for the tanks, as the bulkier constituents of m_1. If we again let $n = 100$, the absolute limit of q is lowered from 1/99 (for the one-unit system) to 1/9 for the two-unit, and to 1/3.9 for the three-unit system (note 10). A several-unit system, although it provides more space in the construction of the objects of m_1 and saves us from shelving the whole undertaking because of the unfeasibility of building m_1 sufficiently lightweight, it nevertheless does not entirely eliminate the undesirable influence of the masses m_1 on the mass of the rocket; the value of μ according to equation (6) still turns out to be larger than that which we would have if m_1 were entirely absent.

If we assume a multiunit system, dividing the trajectory into several segments with equal W_i for each, then for the total flight we obtain an increase in mass by the factor

$$\left(\frac{1}{1-q(n_1-1)}\right)^K \qquad (6a)$$

(where K is the number of segments), compared with the mass that the rocket /553
would have if m_1 were absent.[2] The power exponent in this equation is based on the addition of $m + m_1$ to the right-hand side of equation (6) and removal of mn_i from the bracketed expression (note 11).

A solution of the problem concerning m_1 can be suggested, for which the harmful influence of the mass m_1 is almost entirely eliminated. This solution is contained in the following. As with the multiunit system, several units m_1 are constructed in successively diminishing sizes; the material for construction is, insofar as possible, predominantly aluminum, silicon, magnesium; parts requiring special refractory characteristics (inner surface of the combustion chamber) are made of suitable kinds of graphite, carborundum, corundum.

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[2] In the limit at $K = \infty$, the fraction in equation (6) assumes a value

$$e^{qx} = e^{q\frac{W}{u}} \quad \& \quad \frac{M_o}{M_f} = e^{\frac{W}{u}} e^{q\frac{W}{u}} = e^{\frac{W}{u}(1+q)}$$

(Noted by V. P. Vetchinkin.)

The units, as they become surplus due to the diminishing mass of the rocket, are not discarded but are broken down and fed into the pilot's compartment for remelting and comminution, so that they can subsequently be used as chemical components of the charge. This solution is the ideal one, because all that remains of the harmful mass m_1 is the last and smallest unit, while all of the previous ones are used for the charge, gradually exhausting the functions of m_1. Since the breakdown and subsequent conversion of the objects m_1 requires a certain amount of time, in such a system the division of the rocket trajectory into segments associated with the invariant units m_1 is no longer arbitrary; the first change of units cannot be performed before the rocket has entered into the free earth satellite state; the last changeover cannot be made after the rocket has lost so much velocity in returning home that it cannot function as a free earth satellite. These two changeovers are best limited so that they correspond to a division of the trajectory into three segments with approximately equal W_i for each. To break down the objects m_1 in empty space and convert them into charge materials requires certain additional equipment. Nevertheless, every effort should be directed toward this solution of the problem of m_1, because it will facilitate the main difficulty of the whole undertaking, reducing the required mass of the rocket, which if very large constitutes a very real obstacle to the conquest of interplanetary space and the bodies in our solar system, and is exceedingly difficult to overcome in practice, although theoretically the objective does not present any particular difficulties.

VI. Types of Trajectories and Rocket Velocities Required /554

We will adopt the following notation:

I_j Segments of the rocket trajectory along which it functions, i.e., imparts acceleration to itself.

w_e The "escape velocity" for a given state of the rocket; that velocity by which the existing velocity of the rocket must be increased in order for it to assume motion in a parabolic orbit to the center of the earth.

w_r The "return velocity" for a given state of the rocket; that velocity which the rocket would have if, continuing in its orbit, it reached the surface of the earth (sea level).

w The "total escape velocity" and "total return velocity," equal to w_e as computed for the state of rest at the level of the earth's surface, or equal to w_r as computed for the state of rest at an infinite distance from the earth or for the rocket moving in a parabolic orbit, equal to the "parabolic velocity" = $\sqrt{2Rg}$ (where R is the earth's radius, g is the acceleration due to the earth's force of gravity) = 11,185 m/sec.

v   Velocity of the rocket relative to the center of the earth (not the earth's surface) at a given instant.

r   distance from the rocket at a given instant to the center of the earth;

$$\bar{r} = \frac{r}{R}.$$

By the term "flight" we mean the motion of the rocket to some point at an infinite distance from the earth, and the return from that point, where the velocity of the rocket at the designated point and at the earth's surface must be equal to zero. We will ignore for the moment atmospheric drag and the presence of other bodies in space besides earth, so that our results of this section will be approximately valid only for segments of the trajectory lying outside the significantly dense part of the atmosphere and not in the vicinity of the moon, as well as for trajectories whose dimensions are considerable in comparison with the radius of the earth's orbit.

It is readily seen that for any state of the rocket we will have the following:

$$w_e = \frac{w}{\sqrt{\bar{r}}} - v; \quad w_r = \sqrt{v^2 + w^2\left(1 - \frac{1}{\bar{r}}\right)}. \tag{8}$$

For a rocket in the earth satellite state with a circular orbit:

$$w_e = v(\sqrt{2} - 1) = \frac{w}{\sqrt{2\bar{r}}}(\sqrt{2} - 1)^{\text{(note 12)}} \quad 12)$$

$$w_r = \sqrt{w^2 - v^2} = w\sqrt{1 - \frac{1}{2\bar{r}}}$$

In the case when the orbit of the rocket does not touch or cross the earth's surface, as in the example of any circular orbit, our definition of the quantity w_r is fictitious. In this even, w_r must be interpreted as the velocity that the rocket would have if to its kinetic energy were added the energy due to its mass and difference in the gravitational potential energy of the earth between its position at a given instant and the level of the earth's surface, regardless of whether or not this summation of energies can actually be performed as the rocket moves in its given trajectory. It is not too difficult to see, then, that w_e has different values for points at different distances from the earth in the same orbit (provided only that the orbit is not parabolic, in which case $w_e = 0$); w_r, on the other hand, has a constant value for all points on the same orbit. The quantities w_e and w_r have the following significance: /555

1) The value of w_e for the perigee (the point of the orbit nearest the earth's center) is the theoretical minimum of W (i.e., computed only the basis of the law of energy conservation) necessary for the rocket, moving in a given orbit, to acquire motion in a parabolic orbit such that the rocket can execute the first half of its "flight," i.e., the motion to a point at infinity.

2) w_r is the theoretical minimum of W necessary for the rocket, moving in a given orbit, to reach the earth's surface with zero velocity and thus complete the second half of its flight.

For proof of the first postulate we compare w_{e_1} and w_{e_2}, computed for two points a_1 and a_2 in the same orbit, where the difference in earth-gravitational potential energy is equal to an infinitesimal quantity α. If for the more distant of the points, say a_1, we have, according to equation (8),

$$w_{\gamma_1} = w\sqrt{\frac{1}{r}} - v,$$

then for the nearer point a_2 we obtain

$$W_{e2} = \sqrt{\frac{w^2}{r} + 2\alpha} - \sqrt{v^2 + 2\alpha}, \text{but } \lim\left[\sqrt{\frac{w^2}{r} + 2\alpha} - \right.$$

$$\left. - \sqrt{v^2 + 2\alpha}\right]_{\alpha=0} = \left(w\sqrt{\frac{1}{r}} - v\right) - \alpha\left(\frac{1}{v} - \frac{\sqrt{r}}{w}\right) =$$

$$= w_{e1} - \alpha\left(\frac{1}{v} - \frac{\sqrt{r}}{w}\right)$$

(since for elliptic velocities $v < w/\sqrt{r}$, it follows that $1/v > \sqrt{r}/w$ and the quantity in the parentheses is positive - noted by V. P. Vetchinkin).

Consequently, in absolute value, which is all that concerns us, $w_{e_2} < w_{e_1}$. Therefore, w_e has a minimum at the perigee of the given orbit, which is the theoretical minimum of the rocket velocity needed for transition to a parabolic orbit, Q.E.D. /556

For proof of the second postulate, we compare two values w_{r_1} and w_{r_2} obtained in two situations: one, in which the rocket moves in a certain orbit and acquires a velocity increment u at the point a_1; another, where it moves in the same orbit with the same velocity and acquires the same negative velocity increment at another point a_2, the difference in gravitational potential energies between the points $\underline{a}_1$ and $\underline{a}_2$ comprising an infinitesimal quantity α. If in the first instance we have, according to equation (8),

$$w_{\text{в}_1} = \sqrt{(v-u)^2 + w^2\left(1-\frac{1}{r}\right)},$$

then in the second instance we obtain

$$w_{r_2} = \sqrt{\left(\sqrt{v^2+2\alpha} - u\right)^2 + w^2\left(1-\frac{1}{r}\right) - 2\alpha}, \text{ but}$$

$$\lim\,[w_{r_2}]_{\alpha=0} = \sqrt{(v-u)^2 + w^2\left(1-\frac{1}{r}\right)} - \alpha u:$$

$$: v\sqrt{(v-u)^2 + w^2\left(1-\frac{1}{r}\right)} = w_{r_1} - \alpha u : vw_{r_1}.$$

Consequently, $w_{r_2} < w_{r_1}$, and, the nearer to earth the points at which deceleration is applied, the smaller will be the value of w_r. We obtain the minimum w_r by applying negative velocity increments to the rocket at the level of the earth's surface. In order for the rocket to complete its flight, we must eliminate at the earth's surface the entire velocity that the rocket possesses, which will equal w_r for a given orbit, Q.E.D.

The two foregoing postulates can be explicated as follows.

A certain expenditure of charge on the part of the rocket imparts to it some definite positive or negative velocity increment, independently of the state of rest or motion of the rocket itself, but inasmuch as the energy of the rocket relative to earth - its kinetic energy - is proportional to the square of its velocity relative to the earth, a certain given velocity increment will constitute a larger positive or negative increment in the kinetic energy when /557 it occurs for a larger initial velocity of the rocket; for instance, a velocity increment equal to 4, applied to a velocity of 2, will mean an increment in the kinetic energy of

$$\frac{6^2 - 2^2}{2} = 16,$$

whereas the same velocity increment, equal to 4, applied to a velocity of 20, will represent a kinetic energy increment of

$$\frac{24^2 - 20^2}{2} = 88.$$

Consequently, from the point of view of the rocket's energy relative to earth, the exhaust will produce a greater reaction on the part of the rocket the higher the velocity of the rocket itself. But the velocity of the freely moving rocket will be greatest at the point of closest approach to earth, hence the reaction at this point will be most favorable, both when it is necessary to impart to the rocket sufficient energy for escape from earth and when it is necessary to deplete its energy for favorable descent to earth.

Thus we see that W can attain a minimal value 2w only under the necessary (but still insufficient) condition that all accelerations and decelerations be executed at the earth's surface; since this is impossible, W will be smaller, the nearer to the earth's surface the segment I_j is located. Hence, nearness to the earth's surface of all segments I_j of the rocket's self-imparted acceleration if the prime requirement that we must impose on the rocket trajectory to preclude a superflous increase in the rocket velocity W. We call the difference W-2w the "rocket velocity surplus" and denote it by the symbol L. We donote by L_i the surplus of a given segment, or that part of the total surplus L which was a nonminimal result of the conditions under which the rocket traversed a given segment of its trajectory.

In general,

$$L_i = W_i \pm \left\{ v_2 - v_1 + w \left(\sqrt{\frac{1}{\bar{r}_1}} - \sqrt{\frac{1}{\bar{r}_2}} \right) \right\}, \tag{9}$$

where v_1, v_2, $\bar{r}_1$, $\bar{r}_2$ are the respective values of the quantities at the beginning and the end of the ith segment. The upper sign should be used for "elliptic" rocket velocities ($v < w\sqrt{1/\bar{r}}$) for the first half of the "flight"; in all other cases the lower sign should be used.

If the difference in gravitational potential energy at the ends of a given segment is equal to the infinitesimal quantity α, we will have the following in a flight without resistance of the medium: /558

$$L_i = \pm \alpha \left(\frac{1}{v} - \frac{\sqrt{\bar{r}}}{w} \right). \tag{10}$$

The upper sign should be used for elliptic velocities, the lower for nyperbolic velocities. The parabolic trajectory does not yield a velocity surplus in itself, since always in this case

$$w \sqrt{\frac{1}{\bar{r}}} = v.$$

A subscript on the symbol L will be used to denote the particular physical factor giving rise to the surplus velocity. For example, in equation (10) we have L_{ig}, because the surplus is due to gravitational acceleration; the subscript s will be used to indicate the summed effect of all factors, d the effect of atmospheric drag, with two subdivisions d_p and d_v, of which more will be said below in section VIII. According to the discussion, of all the possible trajectories, L necessarily gives those whose elements are the elements of free orbits that do not touch or intersect the surface of the earth, because with such an element present in the trajectory, the "first requirement" (see above) is clearly not fulfilled. Maximum L gives the presence of a circular orbit of some finite radius in the trajectory.

The second requirement which we must impose on the rocket trajectory in order to attain the smallest possible L is that the angle β between the direction of the reactive force and the tangent to the trajectory be as small as possible. The absolute value of v varies not as a function of the total self-imparted acceleration of the rocket j_0, but only as its tangential component, equal to $j_0 \cos\beta$; we obtain, therefore,

$$L_{i\beta} = W_i(1 - \cos\beta). \qquad (11)$$

We divide the trajectory of the total flight arbitrarily into three segments.

1) T_e, the "escape trajectory," or the segment beginning at the earth's surface and ending at some infinitely distant point.

2) T_c, the "connecting trajectory," or the segment beginning at the end of T_e and ending at some other, infinitely distant point.

3) T_r, the "return trajectory," or the segment beginning at the end of T_c and ending at a point on the earth's surface. In correspondence with the indicated notation, we will adopt the notation W_e, W_c, W_r. We also denote the following: /559

Θ The angle between the trajectory at a given point and the horizon plane.

β The angle between the direction of intrinsic acceleration j_0 and the trajectory at a given point of the latter.

$\lambda = \Theta + \beta$ is the angle between the direction of j_0 and the horizon plane. The angles Θ and β are considered positive when the tangent to the trajectory is directed upward from the horizon plane, while j_0 is directed upward from the tangent to the trajectory.

The reasoning behind our division of the trajectory is the following. At infinity from the earth the earth's gravity is inconsequential and there is no drag from the earth's atmosphere. Consequently, T_c can take on any shape whatsoever and, regardless of the shape, it can be negotiated by the rocket with arbitrarily small j_0, v, and W_c, because it lies entirely at an infinite distance from the earth.

In practice, a segment of the trajectory situated at a distance a few multiples of ten times the earth's radius from the earth can be equated to T_c. W_c is determined in practice largely by the amount of time that we agree is convenient for traversal of T_c.

T_e and T_r, on the other hand, have parts lying within the sphere of influence of earth's gravity and parts within a resistive medium, the atmosphere. Consequently, a particular quantity L, and hence W, will depend entirely on the geometric shape and velocities that we choose for T_e and T_r; in our subsequent analysis of trajectory types, therefore, we will be concerned only with the segments T_e and T_r, ignoring the comparatively unimportant segment T_c.

Since, in the absence of resistance from a medium, segments T_e and T_r which are identical in configuration and absolute value of the velocities at similar points require equal accelerations at similar points in order to be identical, the quantities W_e and W_r for these T_e and T_r will also be equal. The computations given below pertain identically to T_e and T_r, since they lie outside the limits of a significantly dense atmosphere.

It is not difficult to grasp the impossibility of constructing a trajectory that would simultaneously correspond to both requirements outlined above for minimizing the velocity surplus L. A type of trajectory that fully meets the "second requirement" is a "radial" trajectory, for which T_e and T_r represent continuations of the earth's radii. In accordance with the "first requirement," we must minimize I_j in a radial trajectory, imparting to the rocket at large a value of j_0 as possible, starting with the point of departure and continuing to the point at which the rocket attains the parabolic velocity $v = w\sqrt{1/\bar{r}}$; in returning from the corresponding point, it will be necessary to begin with j_0, the "intrinsic deceleration" of the rocket.

We will assume for simplificity that the gravitational acceleration over the entire extent I_j is the same as on the earth's surface, g. We denote /560

$$\overline{j_e + j_p} = j \;\&\; j : g = \bar{j},$$

where j_0 is the intrinsic acceleration, j_ρ is the deceleration imparted to the rocket by the force of atmospheric resistance, and j is their vector sum (in the present case, with a radial trajectory, it is equal to the algebraic difference), which we will call the "mechanical acceleration," in correspondence with which $\bar{j}$ is the ratio of the mechanical acceleration to the gravitational acceleration. With these assumptions and notation, we have from equation (9)

$$L_g = w\left(\sqrt{\frac{\bar{j}}{\bar{j}-1}} - 1\right) \text{ (note 13)} \tag{12}$$

or, simplifying for $\bar{j} \gg 1$,

$$L_g = w\frac{1}{2(\bar{j}-1)}$$

These values, which are somewhat larger than the actual values for finite $\bar{j}$, are taken as approximate values of the surplus from the influence of the force of gravity in a radialtrajectory, assuming $\bar{j} \geq 5$ (for small values of $\bar{j}$, a radial trajectory is unsuitable everywhere).

The type of trajectory corresponding to the "first requirement" is a "tangential" trajectory (see fig. 1); from the point of departure 0 to the point b the rocket moves parallel to the earth's surface via the arc of a large circle; the rocket attains horizontal motion by directing j_0 at an angle β with respect to the horizon and trajectory such that the force $Mj_0 \sin \beta$ will counteract the amount by which the gravitational force on the rocket exceeds its centrifugal force; up to the point d_1, the angle β must be positive, but after this point, at which $v = w\sqrt{1/2}$, β is made negative, since the centrifugal force will already exceed the force of gravity. The circular motion continues until the angle β required for its maintenance, ever increasing (in absolute value) with increasing velocity and centrifugal force, attains a value such that L_β (eq. (11)) becomes an appreciably deleterious part of the surplus. When β (corresponding to the rocket velocity) attains such a value, the rocket moves for a certain time at constant β, moving further away from the earth's surface at an ever-increasing angle θ. When at the point b_1 the factor L_g becomes essentially disadvantageous due to the ever-increasing difference in potential energy between the position of the rocket at a given instant and /561
the perigee of the orbit along which the rocket would have moved had j_0 been discontinued (for the effect of this difference on W_e, see page 74), the rocket ceases to function, and from the point b_1 to the point b_2 the rocket moves freely along an elliptic orbit. At the point b_2, which is symmetrical to b_1 (relative to the major axis of the ellipse), j_0 is renewed at $\beta < 0$, such that

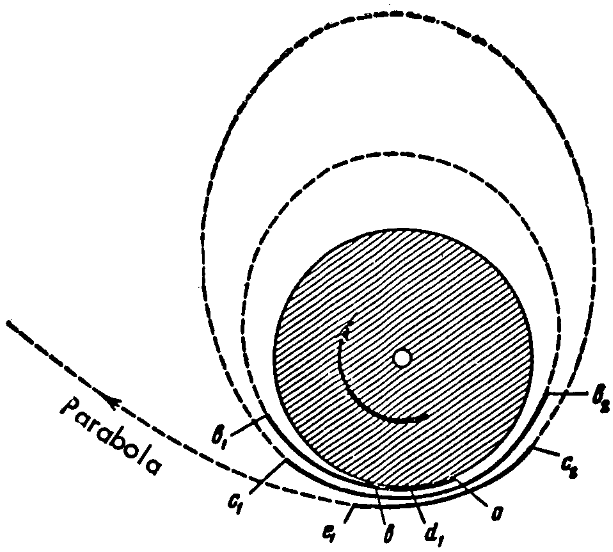

Figure 1. Tangential Trajectory.

The heavy lines indicate T_g; the dashed lines indicate the segments of free flight along elliptic and parabolic orbits.

I_j will be traversed as near the earth as possible, and is continued to the point c_1 corresponding to the same condition as the point b_1; after the point c_1 it again follows a free elliptic orbit $c_1 - c_2$, then again an $I_j = c_2 - e_1$ near the earth's surface, and so on, until in traversing the last I_j we attain the required parabolic velocity and orbit. In the tangential trajectory, the $L_{g\beta}$ which is attained after the rocket passes the point d_1 can theoretically be made as small as we like by moving the points d_1 and b, b and b_1, b_2 and c_1, c_2 and e_1, etc., sufficiently close together; the only thing to consider is that the number of elliptical intervals and flight time will be increased. We will neglect this part $L_{g\beta}$ as being largely dictated by our wishes; by contrast, the l_β obtained by the point d_1 has a definite theoretical minimum, equal approx- /562
imately (for $\bar{j} \gg 1$) to

$$L_\beta = w \frac{1}{6\bar{j}^2} . \tag{13}$$

We will use this approximate value in our later analysis. Besides having a surplus of less than $3\bar{j}$, the tangential trajectory has the further advantage that, by launching and returning the rocket in the equatorial plane from west to east, we can utilize the earth's rotation about its own axis to obtain for the whole flight an economy of the rocket velocity W, equal to twice the velocity of motion of the earth's surface:

$$2U = 920 \text{ m/sec}.$$

In addition to the difficulty of precise control of the required tangential trajectory, it has one added disadvantage, which makes its application in pure form for departure impossible; the tangential type T_e requires points of departure outside the significantly dense atmosphere, because otherwise, due to the considerable length of the segments situated on a level with the point of departure and somewhat above it, L_d would increase by an incredible amount, many times exceeding the savings in W obtained from the smaller surplus $L_g\beta$ for a tangential trajectory and from utilization of the earth's rotational velocity. In practice, therefore, the most favorable type of T_e is not tangential, but a certain compromise, beginning with a spiral arc, as shown in the example of figure 2. The angle θ for this spiral should be smaller, the smaller the admissible value of j_0 (and, hence, the larger will be our value of $L_g\beta$), and the smaller the deceleration j_ρ elicited by atmospheric drag. For this middle type of trajectory, $L_g\beta$ will have a value midway between $w/2(\bar{j}-1)$ and $w/6\bar{j}^2$.

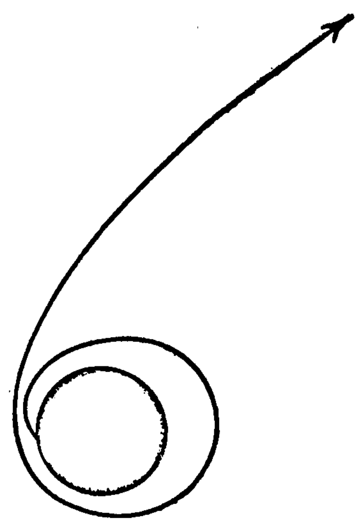

Figure 2. A Hybrid Type Trajectory.

Below, we will assume that for $\theta<30°$ and for $\bar{j}>3$, provided the rocket does not use airplane wings, or for $\bar{j}>1$ when the rocket does use them,

$$L_g\beta = \frac{w \sin\theta}{3\bar{j}} \tag{14}$$

with the necessary condition that aviation type lifting surfaces are used, provided only that $j < 2$. As for the tangential type T_r, it is applicable in essentially pure form and can yield very considerable economy in W_r, due to the useful aspect of atmospheric resistance in the return flight, which helps to attenuate the return velocity of the rocket. This will be discussed separately below, in section IX. /563

VII. Peak Acceleration

We see from equations (12), (13), (14) that L, hence W and n, decrease with increasing j and $\bar{j}$; it is important, therefore, for us to find out what is the maximum mechanical acceleration j that we can impart to the rocket. The "mechanical acceleration" is the acceleration elicited by the resultant forces acting exclusively on the external parts of the rocket, which is then the acceleration felt inside the rocket, whereas the gravitational acceleration, applied identically to all parts of the mass of the rocket, will not be perceptible inside it. The limiting value of j can be hypothesized in terms of four factors: 1) the capability and endurance of the rocket construction; 2) endurance of the pilot's organism; 3) atmospheric drag, which increases with increasing velocity and can make the application of smaller j more advantageous while passing through the atmospheric layers of appreciable density, equations (12), (13), and (14) notwithstanding; 4) design problems in the building of sufficiently lightweight and portable components of the proportional passive load (tanks, pumps, burners, etc.), which will have sufficient performance capabilities to impart large acceleration to the rocket.

The third factor may be of significance only for a relatively small segment near the earth's surface; this will be discussed further in section VIII. The endurance of the rocket depends on how durable we wish to build it. Factors which may contribute to the upper limit of j for a large part of T_e are, therefore, endurance of the human organism, this factor being the least prone to our efforts to enhance it, and the dimensions of the objects m_1, which we can make lighter more portable than some limit dictated by modern mechanical engineering.

Too large a value of j can prove harmful and even fatal for the pilot, in that all fluids of the living organism and, above all, the blood tend toward those parts of the body which are situated opposite to the direction of apparent gravity created by the acceleration j. If, for example, to a man 200 cm (about $7\frac{1}{2}$ feet) tall we were to impart an acceleration $j = 10$ g for a sufficient period of time in the lengthwise direction of his body, from toe to head, a difference of about two atmospheres would be developed in the blood pressure between the soles of the feet and forehead region, which is probably quite enough for the head to become completely drained of blood, and the feet would become charged with blood vessels, unless special precautions were taken against these effects. The first condition for the organism to be able to bear the acceleration j is to reduce the height of the blood column as much as possible in the direction of acceleration, i.e., to recline the body in an attitude perpendicular to the direction of j. The inflow to the "lower" (i.e., lying opposite the direction of j) parts of the body and drainage of blood from the /564 "upper" parts can be inhibited by counteracting the internal difference in blood pressure with the same difference in external pressures on the part of a liquid with the same density as the blood, in which the body would have to be immersed. Otherwise, the movement of the blood masses can be inhibited by seating the body in a smooth, hard, everywhere tightly form-fitting contour.

Any method that serves equally to prevent fluid drainage in the external surfaces of the body (when large accelerations are applied), is completely inapplicable to the internal surface of the lungs. Yet it is in the internal surface of the lungs that the most delicate blood vessels are found, intimately intermingled with the air sacs, without even the most diaphanous tissues to separate them. Since the absolute density of the air in the lungs is insignificant in comparison with the blood density, the pressure difference dhj obtaining between the "upper" and "lower" surfaces of the lungs, where d is the absolute density of the blood, h is the dimension of the lungs in the direction of j, cannot in any way be compensated from without, i.e., from the space of the lung vacuoles. If this difference exceeds the limiting resistance of the capillary vessels and tissue of the lung vacuoles, first there will be rupture, after which the "lower" surface of the lungs will become engorged with blood.

By its structure, the chest cavity presents still another impediment to the development of large accelerations; it contains a number of organs or relatively disparate densities, the heart and lungs. With the communication of acceleration to the body, the heavier heart will suffer a displacement in the opposite direction inside the chest, which can, if the effect is intense enough, have a pernicious effect on the activity of the heart and its neighboring left lung, the latter becoming deformed. Consequently, the permissible limiting acceleration for the human organism will be dictated by the resistance of the internal surface of the lungs to rupture and the resistance of the attaching members of the heart to displacement. Thus, in whatever direction the heart will best withstand stress, forward or backward, will determine whether the man should have his chest or his back in the direction of acceleration. The endurance of the lungs can be enhanced considerably by turning the body about its long axis, which will be perpendicular to the direction of acceleration. With such rotation, we would probably find that the blood would not have a chance to flow to /565
any part of the lung, since all would be changing their positions gradually relative to the direction of apparent gravity. With such rotation of the body, the heart would no longer suffer a constant displacement in one direction, but would tend to move in a circle, which would affect both the organ itself and its neighboring left lung, although in just what measure is not known.

An exhaustive, well-grounded investigation of the endurance to j on the part of the human organism can be carried out very well on a large centrifuge, the most practical and least expensive type of which for our purpose would be something like a "giant stride" with two cables, one of which holds the experimental chamber with pilot, the other holding a counterweight. We can obtain some indications as to the magnitude of the permissible j from experiments on the giant stride and present-day aviation tests. On the giant stride, the acceleration attained is often as high as 2g and can be continued for a fair amount of time, while aviators performing stunt flying withstand short-term accelerations up to 8g, as well as fairly prolonged accelerations to 2g. In either case, no really harmful effects are observed (note 14). Bearing in mind that in swinging on the giant stride and in airplane flights, the position of the human body relative to the direction of j is usually lengthwise, i.e., in the most unfavorable position, in that the dimensions of the lungs in the direction from the shoulders to the pelvis are the largest, we have good reason to assume that under favorable conditions, namely with the body primarily in a transverse attitude, the human being could stand an acceleration $\bar{j} = 5g$ for a period of three minutes (more is not required) without any particular harm. If it should prove possible to apply rotation of the body about its long axis, the value of permissible $\bar{j}$ might even surpass 10g. The value of $L_{g\beta}$ corresponding to $\bar{j} = 5g$ will be: for a radial trajectory w 0.125 and for a tangential trajectory w 0.007. To the value $L_{g\beta} = 0.125$ w for 2w : u = 5, which is the relation we will approximately realize in actuality, corresponds a 1.87-fold increase in n. As for the design capabilities in building the objects comprising the proportional passive load so that they are sufficiently portable with a high performance rating in attaining an appropriately large j_0, this problem must await the corresponding engineering research. In all probability, it will be this design factor that will impose the practical upper limit on j_0.

VIII. Effect of the Atmosphere on the Rocket during Departure

/566

In departure, an important factor contributing to the rocket velocity surplus L is the resistance of the atmosphere, which, first of all, reduces the actual acceleration I of the rocket relative to the center of the earth $(I = \overline{j_0 + g + j_\rho} = \overline{j + g})$ and thus decreases v, and, second, compels us to give the angle θ a value greater than zero in order to avoid too large a rocket velocity within the dense regions of the atmosphere, hence in order to avoid too large a value of L_d. However, increasing θ, according to equation (14), causes an increase in $L_{g\beta}$. Furthermore, we may be forced to diminish j and v over a certain initial segment of T_e in order to preclude catastrophic overheating of of the surface of the rocket.

The effects of resistance from the medium and heating of the moving surfaces have been theoretically investigated only very meagerly, and there is almost no experimental material relating to velocities expressed in kilometers per second. Consequently, all that we might know beforehand about these effects is their approximate magnitude, determined on the basis of simplified laws governing the dependence of the drag and heating of the moving surfaces on their shape, slope angle and velocity of motion, as well as the density, chemical composition, and temperature of the medium. We cannot discuss the exact computation of these effects right now, because they are not amenable to such treatment, even for velocities at which variations in the density of the medium surrounding the body can be neglected. We will base our computations on the following equation, which in general is approximately valid:

$$Q = Skv_1^2\Delta 10^{-4}c, \tag{15}$$

where Q is the drag force in kg, S is the cross sectional area of the body in m^2, k is a proportionality factor, which, according to experimental data for velocities near the velocity of sound, where it is a maximum, is equal to 0.25; v_1 is the velocity of the body relative to the air in cm/sec (in our case, neglecting the wind, $v_1 = v - U$, where U is the velocity of rotation of the earth's surface), c is a coefficient depending on the shape of the body and equal to unity for a normally oriented plane, and $\Delta = \rho_h/\rho_0$ is the ratio of the atmospheric density at the position of the rocket at a given time to its value at sea level.

Since it has been more to our advantage to work with accelerations throughout the present article, rather than with the forces giving rise to them, in the present case as well we will go from the resistance of the atmosphere to /567 the deceleration that it induces on the part of the rocket, denoting this deceleration by j_ρ. Expressing all quantities in absolute units, substituting $k = 0.25$, and introducing in place of S the transverse load of the rocket P, we obtain from equation (15)

$$j_\rho = 2{,}5\cdot 10^{-3}\frac{c}{P}v_1^2\Delta = k_1 v_1^2\Delta, \tag{16}$$

where $k_1 = 2.5\cdot 10^{-3}\frac{c}{P}$ (j_ρ in cm/sec^2, P in g/cm^2, v_1 in cm/sec).

In the resistance of the atmosphere and in the heating of the moving surface, we can distinguish two essentially discrete parts, which are the result of different factors: 1) resistance (drag) and heating due to the pressure of the medium on surfaces inclined at an angle with their trajectory; 2) resistance and heating due to viscosity of the medium as it slides along the moving surfaces. The first two effects are the outcome of adiabatic compression of the air ahead of the frontal surfaces of the body and adiabatic expansion of the air behind the rear surfaces. The second two effects are the outcome of internal friction in the medium as it slides along the surfaces of the body. For the first two effects, we will use the notation d_p and h_p, for the second two, d_v and h_v. Equation (16) applies only to d_p, which in general is proportional to the square of the velocity and the first power of the density, whereas d_v, in those layers of the atmosphere where the mean free path of the gas molecules is negligible in comparison with the dimensions of the moving body, is proportional to the one and one-half power of the velocity of the body and the square root of the medium density. Since, according to experimental data, d_p turns out to be larger than d_v for bodies that do not possess a particularly elongated profile,[3] at velocities of several meters per second in the sea level atmosphere, the drag d_v, which is less dependent on the velocity, will be made insignificant relative to the quantity d_p at the hundred- and thousand-meter per second velocities that the rocket will have even in the lower layers of the atmosphere (at the beginning of the path, the ratio $d_v/d_p = kv^{-\frac{1}{2}}\rho h^{-\frac{1}{2}}$ will decrease rapidly).

At altitudes of several tens of kilometers, the drag d_v, being less dependent on the density of air than d_p, can also be made a relatively appreciable quantity, but at such altitudes, due to the inconsequential density of the air, both d_p and d_v will no longer be significant in absolute value, in spite of the increasing velocity. The principal part of the total drag $d_s = d_p + d_v$ will therefore be d_p for about the first 30 to 40 km above sea level. In /568 order to formulate an overall approximate notion as to c and j_ρ, we will therefore confine ourselves to the theoretical investigation of d_p only.

The main factor representing atmospheric effects is the density of the atmosphere. If we regard the gravitational acceleration, chemical composition of the atmosphere, and its temperature as identical at all altitudes, its density will be a decaying exponential function of the height, which we can express with fair accuracy in a form suitable for approximate calculations as follows:

$$\rho_h = \rho_0 2^{-h/5}. \qquad (17)^4$$

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[3]The cross section of the rocket must include the pilot compartment, consequently, it has a definite minimum of about 4 m^2 in area; the shape of the rocket, therefore, cannot be too elongated.

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[4] Assuming a constant temperature t = -50°C, which is observed at altitudes above 10 km. Equation (17) is normally written $\rho_h : \rho_0 = e^{-h/7.2} = 10^{-h/16.5}$, where h is the altitude in kilometers above sea level, ρ_0 is the density of the atmosphere at sea level (noted by V. P. Vetchinkin).

There are no precise empirical data relating to the composition of the atmosphere at large altitudes, but, according to existing data, the temperature and buoyancy of the air as we move upward do not follow an adiabatic law, instead they fall off more slowly than adiabatic. This indicates that in the atmosphere there is a limit above which the intermingling ascending and descending air currents cannot penetrate. Above this upper limit of the atmosphere with its constant percentage composition, the partial densities of all the gases must no longer decrease proportionately on moving upward, but in conformity with their molecular weights; the percentage content, and according to the latest research the absolute partial density at certain altitudes, of the lightest component of the atmosphere - helium - must almost double every 5 km of height. This factor is in our favor during departure, if the takeoff is executed by means of airfoils (wings or fins) but will work against us when we discontinue the use of airfoils. In the first instance, this density would provide support for the airfoils (the problem of overheating of the surfaces can become acute only with respect to a nitrogen-oxygen atmosphere, of which more will be said below), while in the second case it would yield excess resistance to the motion of the rocket once it has developed considerable velocity. This resistance, however, cannot be comparable with the magnitude of the resistance of the lower, dense nitrogen-oxygen layers of the atmosphere. To gain a general notion as to the variation of j_ρ during departure, we will assume: θ_1 = const (the angle θ_1 corresponding to the velocity v_1 is the angle between the velocity v_1 and horizon plane; in departure upward and to the east, $\theta_1 > 0$); I = const; then

$$v_1^2 = 2\cdot 10^5 Ih \frac{1}{\sin\theta_1} \text{ (}v_1\text{ in cm/sec, I in cm/sec}^2\text{, h in km).}$$ /569

The ratio $\rho_h/\rho_0 = \Delta$ is given in equation (17). Substituting the expression for v_1^2 from the foregoing equation and the value of Δ from equation (17) into equation (16), we obtain

$$j_\rho = F(h) = k_1\cdot 2\cdot 10^5 hI \frac{1}{\sin\theta_1} 2^{-\frac{h}{5}} =$$

$$= \text{(with substitution of } k_1\text{)} = 500\frac{cI}{P\sin\theta_1}h\cdot 2^{-\frac{h}{5}} = k_2 h 2^{-\frac{h}{5}}, \tag{18}$$

$$\text{where } k_2 = 500\frac{cI}{P\sin\theta_1}.$$

This function will be used to characterize j_ρ in terms of the height above sea level, assuming that the point of departure is situated at sea level; this is shown graphically in figure 3, for $k_2 = 10$; as it increases from 0 at h = 0, j_p assumes maximum values for $9 > h > 6$, then decreases, becoming similar in its behavior to the function $2^{-h/5}$. Integrating F(h), we obtain the negative work done by the atmosphere on the rocket in dyne-kilometers per gram mass of the rocket:

$$\int_0^h F(h)\,dh = k_2\left\{\frac{25}{(\lg 2)^2} - \frac{5}{\lg 2}2^{-\frac{h}{5}}\left(h + \frac{5}{\lg 2}\right)\right\};$$

$$\int_0^\infty F(h)\,dh = k_2\left(\frac{5}{\lg 2}\right)^2 \approx 50k_2\ (\text{dim: } 10^5\ \text{erg/g}).$$

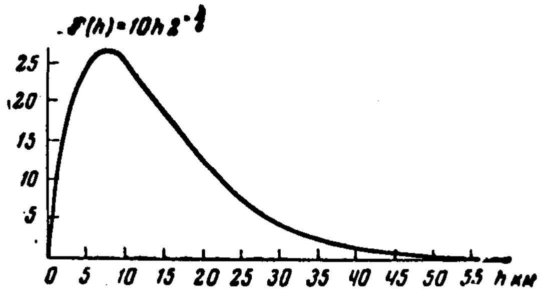

Figure 3

Replacing the factor h by h - h_0 in F(h) and taking $\int_{h_0}^{h_0+h} F(h)dh$, which would /570 correspond to a transfer of the departure point to h km above sea level, we obtain values smaller by a factor of $2^{-h/5}$, hence the negative work on the part of the atmosphere, as well as the values of L_{d_p}, are proportional to the density of the atmosphere at the point of departure. This law is valid for all trajectories which are identical in configuration and velocities and differ only in the altitude of the departure point. From this (and only from this) point of view, the altitude of the departure point is significant. But for the quantity w_e, this altitude is of relatively minor importance within the range of variation that we can possibly realize; thus, for example, moving the point of departure upward 10 km decreases w_e only by about 35 m/sec.

To find the value of L_{d_p}, we must integrate j_ρ with respect to the time. Substituting I·t in place of v_1 in equation (16), expressing Δ in terms of h, and h, in turn, in terms of t and I, as $h = 10^{-5}\frac{1}{2}It^2 \sin\theta_1$, we obtain

$$j_\rho = F(t) = 2{,}5\cdot 10^{-8}\frac{cI^3}{P}t^3 2^{-10^{-6}It^2\sin\theta_1} = k_3\cdot t^3\cdot 2^{-10^{-6}It^2\sin\theta_1}, \tag{19}$$

where

$$k_3 = 2{,}5\cdot 10^{-8}\frac{cI^3}{P}.$$

For the time variation, we will assume arbitrary data convenient for computation: $I = 5000\ \text{cm/sec}^2$ and $\theta_1 = 90°$, whereupon

$$j_0 = 62500\frac{c}{P}t^2 2^{-0{,}005t^2} = k_4 t^2 2^{-0{,}005t^2} \tag{20}$$

where $k_4 = 62{,}500c/P$. The function $j_0 = F(t)$ is graphically illustrated in figure 4 for $k_4 = 1/3$.

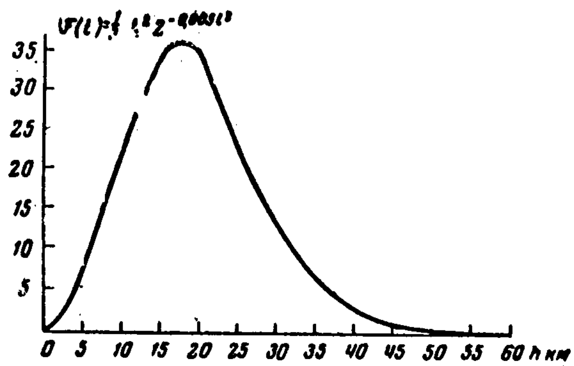

Figure 4

The value of $\int_0^\infty F(t)\,dt$ according to equation (20) (or L_{d_p} for $I = 5000$ cm/sec^2 and $\sin\theta_1 = 1$) is equal to about 2000 k_4. It is readily seen that L_{d_p} must be proportional to $I^{\frac{1}{2}}$ and $(\sin\theta_1)^{-3/2}$. Consequently, for any values of I and θ_1 we will have /571

$$\begin{aligned} L_{dp} &= 2000\,k_4\cdot\sqrt{I:5000}\cdot\sin^{-\frac{3}{2}}\theta_1 = \\ &= 1{,}75\cdot10^6\frac{c}{P}I^{\frac{1}{2}}\sin^{-\frac{3}{2}}\theta_1 = z\sin^{-\frac{3}{2}}\theta_1, \end{aligned} \tag{21}$$

where

$$z = 1{,}75\cdot10^6\frac{c}{P}I^{\frac{1}{2}} = L_{dp}\ (\text{for}\ \theta_1 = 90°).$$

The optimum angle θ_1 is that angle for which

$$L_{g\beta d} = L_{g\beta} + L_d = \text{min.}^{5} \qquad (22)$$

We will assume for simplicity that $\theta_1 = \theta$, i.e., we will neglect the rotation of the earth about its axis. Then the angle θ_1 must satisfy the equation

$$z \sin^{-\frac{3}{2}} \theta_1 + w \frac{\sin \theta_1}{3\bar{j}} = \text{min.}$$

From this, we find

$$\sin \theta_{1_{\text{optim}}} = \left(\frac{9\bar{j}z}{2w}\right)^{2/5}. \qquad (23)$$

Inasmuch as we do not, in actuality, have to make θ_1 = const over the entire I_j, but since, on the other hand, we cannot vary it abruptly, especially at large velocities, since this would require a large angle β and large L_β, it follows that, according to equation (23), $\sin\ \theta_{1_{\text{optim}}}$ can only be an average for a segment I_j lying within the appreciably dense portion of the atmosphere. At the beginning of this segment it is more favorable to take $\theta_1 > \theta_{1_{\text{optim}}}$, then, gradually diminishing it, go over to $\theta_1 < \theta_{1_{\text{optim}}}$, since this diminution can be achieved by the action of the force of gravity and a small deviation of the rocket axis from the trajectory (in order for L_β not to be large, it is necessary that $\beta \le 5$ or 10°). For better penetration through the atmosphere and for obtaining the lowest possible L_d, the rocket should have a long, pointed configuration, and its discharge tube can only be aligned with the long axis. Consequently, along the segment of T_e on which L_d can attain sizable /572 values, namely beginning with the point at which the rocket velocity v_1 attains values of several hundred m/sec and ending with an altitude of about 60 km, the long axis of the rocket, as well as the axis of the discharge tube and the direction of reaction, must be aligned with the direction of the trajectory

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[5] L_α, i.e., the surplus rocket velocity that depends on the opposing reaction of the supporting surfaces inclined at an angle α with respect to the trajectory, is not included here, as it is almost totally independent of the angle θ_1.

in order to obviate excessively large atmospheric drag. Hence, the exhaust-reactive force component normal to the trajectory, being equal to $j_0 M \sin\beta$, and the angle β must be nearly zero; with this stipulation, unless some other normal force is acting on the rocket, the trajectory will curve under the influence of the normal component of the force of gravity, equal to $Mg\cos\theta$, where the radius of curvature will be equal to $\rho = v^2/g\cos\theta$. For velocities $v < 2000$ m/sec and for θ not too near 90°, this curvature of the trajectory could cause the rocket to head back toward earth before it had managed to reach the atmospheric regions of negligible density, wherein the angle β may be given any value we like, without creating considerable atmospheric drag. A force to counteract the normal component of the force of gravity might be the air pressure on lifting surfaces, with which the rocket would be equipped. These would have to be steel surfaces covered with a thermal insulation (aluminum, probably, will probably be unsuitable due to its low-melting characteristic), extending along the body of the rocket and having a surface area such that their load will be approximately 200 kg/m^2. At velocities beginning with $v_1 = 100$ m/sec, a small angle of attack (angle α between the lifting surfaces and trajectory of the rocket) will be sufficient ($\sin\alpha < 1/10$) in order for the lifting force developed by the supporting surfaces to equalize the normal component of the force of gravity and thus to prevent the rocket trajectory from curving downward more than is desirable. The opposing action of the surfaces (i.e., the projection of the force due to air pressure onto the rocket trajectory) will also be relatively minor, namely $Mg\cos\beta\tan\alpha$. It will decrease the translational acceleration of the rocket by an amount

$$g\cos\theta\tan\alpha = \frac{j\cos\theta}{j\cot\alpha}, \tag{24}$$

where, as the rocket picks up velocity, the angle α can be decreased (until the rocket enters the rarefied layers).

Considering $\alpha = \text{const}$ and $\sin\theta \ll 1$ (L_α can only have significant values for small slopes of the trajectory, i.e., for prolonged flight in the atmosphere), we have approximately /573

$$L_\alpha = \frac{w}{3j}\,\frac{\tan\alpha}{\cos\theta} \qquad (24a)^6$$

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[6] In this equation, as in equations (13) and (14), the factor 3 in the denominator is attributable to the following: 1) The surplus occurs in the interval in which the rocket develops only the first 8000 m/sec of its velocity, since beyond this point the rocket becomes a free body; 2) as the velocity builds up from 0 to 8000 m/sec, all drag effects decline to zero, since they are directly related to the apparent gravity of the rocket, but the latter quantity reverts to zero at v = 7909 m/sec at sea level with v in the horizontal direction.

under the condition that the (apparent) gravity of the rocket is inhibited the whole time only by the action of the lifting surfaces. The lifting surfaces are desirable for the initial development of velocity, if we have $2 < j_0 < 3$, and are always necessary for $j_0 < 2$, since for $j_0 = 2$, even for purely tangential flight, L_β amounts to about 600 m/sec, while for $j_0 = 1$, L_β would go to infinity if we were to try and counteract the weight of the rocket solely with the exhaust reaction. Nevertheless, it is quite possible that it will be difficult from the design point of view to create an initial value of $j_0 > 2$; in this event, then, the prolonged application of airfoils is mandatory. In our favor, in this case, is the fact that the ratio j_0/g_a (where g_a is the apparent gravitational acceleration of the vehicle (its weight minus the centrifugal force) will decrease steadily and fairly quickly, on the one hand because of the decrease in g_a as centrifugal force is developed, on the other hand because of a possible increase in j_0 as the mass of the rocket is diminished. Since, for a certain period of time after takeoff, all that functions is the one initial unit m_1, we can maintain its absolute performance rating at the same level and thereby obtain an ever-increasing relative exhaust intensity dM/Mdt and, accordingly, increasing j_0. Thus, for example, at the instant the rocket develops a velocity $v = 5000$ m/sec ($v_1 \approx 4500$ m/sec), its apparent gravitational acceleration will decrease by a factor of 5/8, the mass by a factor of about 2/5, so that, with the rest of the reaction force constant, j_0 will increase fourfold relative to g_a. This fact shortens considerably the period in which the airfoils are utilized, because they are more essential the nearer j_0/g_a is to unity, while for $j_0/g_a > 2$ we can get by without them altogether, countering the gravitational force on the rocket with the vertical component of the reaction force.

The theoretical investigation of the utilization of airfoils at velocities $v_1 > 1000$ m/sec is very difficult for lack of appropriate experiments and investigations, both with respect to the laws of drag and heating of moving bodies at high velocities and with respect to the composition of the atmosphere at heights of several tens of kilometers. If we were to rely on the data of modern aviation, the outlook for the use of airfoils would seem very promising. But, in all probability, at velocities several times the speed of sound, the drag as a function of the angle of attack approaches the Newtonian formula $F/s = k \sin^2 \alpha$, so that the lifting force of the supporting surfaces will be much less than according to the formulas used in aviation, their aeronautical efficiency falling off accordingly. Due to the reduction in lifting force coefficient at large rocket velocities, it would not be able to escape from the comparatively dense atmospheric layers before reaching a velocity of about 700 m/sec (at which the apparent gravity already begins to fall off sharply). /574

Consequently, it is necessary to pay special attention to the problem of the additional drag due to viscosity of the atmosphere d_v and heating of both the frontal portions of the rocket due to adiabatic compression of the air in front of it, and of the sloping surfaces due to the viscous forces. Therefore, leaving open the question of the possible limits of applicability of winged flight, we will consider that the rocket has a ratio $j_0/g_a > 2$ at the instant the rocket develops a velocity $v_1 = 4500$ m/sec. At the very inception of velocity buildup to 100 m/sec, we should make $\beta > 0$ if we wish to have $\bar{j} < 2$, otherwise the initial acceleration of the rocket will be executed by some mechanical means. In the first instance, the axis of the rocket would not at all coincide with the tangent to the trajectory, but at small velocities a certain deviation still would not produce too great a deceleration due to atmospheric drag.

The optimum rocket velocity at a given point of its trajectory, i.e., for given θ and h, is that velocity at which minimum L_s is attained for the element of the trajectory nearest this point. We have, consequently, the equation

$$L_s = L_g + L_d + L_\alpha = \min, \tag{25}$$

where in the functions L_g, L_d, and L_α we must take v_1 as the variable, assuming $\theta = \text{const.}$[7] According to equation (10),

$$L_{ig} = ig\sin\theta\frac{1}{\bar{r}^2 v} - ig\sin\theta\frac{1}{w\bar{r}\sqrt{\bar{r}}} \tag{26}$$ /575

(since α in equation (10) is equal to ig sin $1/\bar{r}^{2}$). The optimum velocity problem is of practical significance only for the segment near the earth's surface in the medium of a dense atmosphere, so that we can let $\bar{r} = 1$ with only minor error. According to equation (16),

$$L_{id} = t/_e = \left(\frac{i}{v}\right) k_1 v_1^2 \Delta;$$

[7]The following computations, like the concept of optimum velocity itself, are only applicable so long as we have $\theta > \alpha$, i.e., so long as the opposing action of the force of gravity at a given point of the trajectory (projection of the gravitational force on the trajectory) is greater than the opposing action of the lifting surfaces, since at an angle θ small in comparison with the attack angle α, the altitude of the rocket at a given instant is directly dependent on its velocity at that instant and vice-versa, while the angle of ascent θ is determined by the way in which the velocity increases, so that the problem of finding an optimum velocity for a given altitude and angle of ascent becomes superfluous.

substituting here the value $v_1^2 = v^2 + U^2 + 2vU \cos\theta$, we obtain

$$L_{id} = ivk_1\Delta + i\frac{U^2}{v}k_1\Delta + 2iUk_1\Delta\cos\theta. \qquad (27)$$

According to equation (24), we have

$$L_{i\alpha} = \left(\frac{i}{v}\right) g \cos\theta \tan\alpha \qquad (28)$$

The third term in equation (27), like the second term in equation (26), does not contain v, hence is constant in this case. Substituting into equation (25) the values of L_{ig}, L_{id}, and $L_{i\alpha}$, excluding the constant terms, we obtain

$$\frac{i}{v} g\sin\theta + \frac{i}{v} U^2 k_1\Delta + \frac{i}{v} g\cos\theta \tan\alpha + ivk_1\Delta = \min.$$

Solving this equation and substituting the value of Δ according to equation (17) and the value of k_1 from equation (16), we get

$$v_{optim} = \sqrt{2^{\frac{h}{5}} \cdot 400P \cdot g\, \frac{1}{c}\,(\sin\theta + \cos\theta\tan\alpha) + U^2}. \qquad (29)$$

v_{optim} is the value of the velocity that should not be exceeded in flight, at any rate not exceeded by too great an amount. Should it happen that, for the I and θ that we have chosen on a certain ith segment, the rocket velocity turns out to be much greater than optimum for the given h and θ, I would have to be decreased at the beginning of this segment until the rocket had attained considerable altitude, at which v_{optim} is then made larger (equation (29)).

Substituting the value of z from equation (21) into equation (23) and neglecting the difference between j_0 and I (we can do this without dangerous error, since flight is possible in general only as long as the difference between j_0 and I is not too great, i.e., as long as L_s is not particularly great), we obtain

$$\sin\theta_{optim} = 0{,}14\left(\frac{c}{P}\right)^{1/8} I^{3/8}. \qquad (30)$$

Substituting this expression for sin θ into equation (21), we obtain /576

$$L_{dp} = 1{,}75\cdot 10^6\,\frac{c}{P} I^{1/4}\left\{0{,}14\left(\frac{c}{P}\right)^{1/8} I^{3/8}\right\}^{-3/4} = 34\cdot 10^6\left(\frac{c}{PI}\right)^{5/6}. \qquad (13)$$

Substituting the value of sin θ from equation (30) into equation (14) and again neglecting the difference between j_0 and I, we obtain

$$L_{g\beta} = \frac{w}{3I}\,0{,}14\left(\frac{c}{P}\right)^{2/5} I^{3/5}\,981 = 5\cdot 10^{7}\left(\frac{c}{PI}\right)^{2/5} \qquad (32)$$

Adding equations (31) and (32), we obtain $L_{g\beta d}$ as a function of the acceleration I, under the condition that the rocket follows a trajectory with an ascent angle $\theta = \arc\sin\theta_{opt}$ = const, and for I = const:

$$L_{g\beta d} = 84\cdot 10^{6}\left(\frac{c}{PI}\right)^{2/5}. \qquad (33)$$

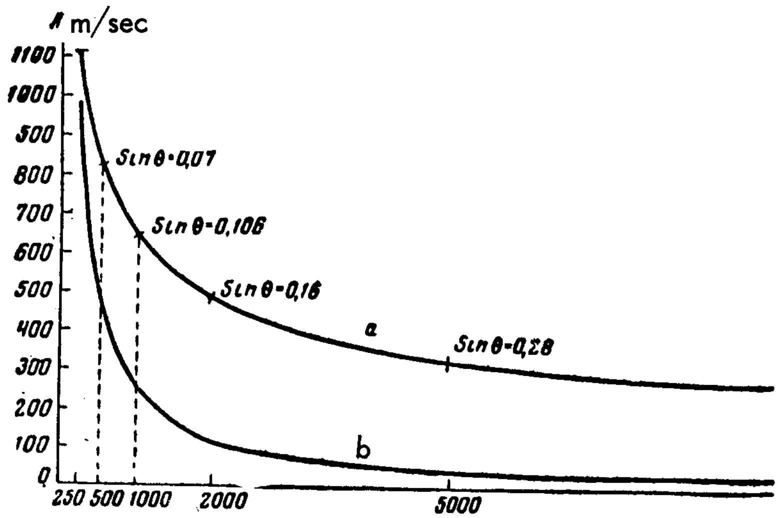

Figure 5

a) $L_{g\beta d} = 84\cdot 10\ (1/\ 62{,}500I)^{2/5}$; b) $L_{\alpha} = (1{,}120{,}000/3I)\tan\alpha$

(L in m/sec; I in cm/sec^2; $\tan\alpha = 0.1$).

A graph of this function (equation (33)) is given in figure 5 for c/P = 1/62,500 (c = 0.04, P = 2500, these values being approximately probable data). In the same graph is given the function

$$L_{\alpha} = F(I) = \frac{w}{3I}\frac{\tan\alpha}{\cos\theta}, \qquad (24a)$$

where in the latter we neglect the divisor cos θ (which in prolonged winged flight is necessarily very near unity) and, as in the preceding equations, we assume $I = j_0$. /577

The quantities $L_{g\beta d}$ from equation (33) and L_α from equation (24) cannot be added one to the other, since the assumptions underlying the derivation of these equations are mutually exclusive; if the prolonged use of airfoils (L_α) necessitated by small j_0 holds, then we cannot have θ = const, but if j is large and sin θ is correspondingly not to small, airfoils are not used continuously and we can have α = const. In the first situation, we must orient more according to equation (24a), in the second situation, more according to equation (33); the changeover point is $j_0 \approx g$.

In the present section, we have permitted a number of simplifications (though always toward increased resistance; in particular, equating θ to the larger angle θ_1, we increased the calculated loss in velocity $L_{g\beta}$, while taking the maximum value of the coefficient k in equation (15), we increased the calculated loss in velocity L_{d_p}), and in equation (33) (see fig. 5) we introduced more or less probable yet nevertheless arbitrary data (c = 0.04, P = 2500), doing the same in equation (24) (α = 0.1). With this in mind, as well as the fact that in departure at an angle $\theta_1 < 30^\circ$ (from all indications, θ_1 would never be greater than 30° in any case) the economy in W_e from utilization of the earth's velocity of rotation about its axis will amount to about 450 m/sec, we may regard the following as a safe result of the calculations in the present section: the required rocket velocity W_e, taking into account all resistance effects, will not be more than 12,000 m/sec and, in all liklihood, will be somewhat less.

As far as heating of the rocket surface is concerned, this problem will clearly not be as acute in departure, a fact which may be inferred from the following considerations:

Let us assume

$$p_v = 0.02\,\rho v^2, \qquad (34)$$

where p_v is the pressure in atmospheres in a plane moving perpendicular to itself at a velocity v in m/sec, ρ is the atmospheric density in g/cm³, v is the velocity in m/sec, 0.02 is the drag coefficient for the largest velocity investigated;

$$P_0 = 80\ \frac{\rho T}{m}, \qquad (35)$$

P_0 is the buoyancy of the atmosphere in atmospheres, T is the absolute temperature, O is the same as in equation (34), m is the molecular weight (mean) of the gases constituting the atmosphere. The adiabatic compression formula is

$$\frac{T}{T_1} = \left(\frac{p}{p_1}\right)^{\frac{k-1}{k}} \tag{36}$$

where k = 1.41.

Regarding heating as the result of adiabatic compression, we obtain for velocities $v > 700$ m/sec, at which $p_v \gg p_0$ (in an oxygen-nitrogen atmosphere; for other gases, the lower limit of applicability of the following equation is proportional to their molecular velocity): /578

$$T_1' = 0{,}09T^{\frac{1}{k}} v^{\frac{2(k-1)}{k}} m^{\frac{k-1}{k}} = 0{,}09T^{0{,}71} v^{0{,}582} m^{0{,}291} \tag{37}$$

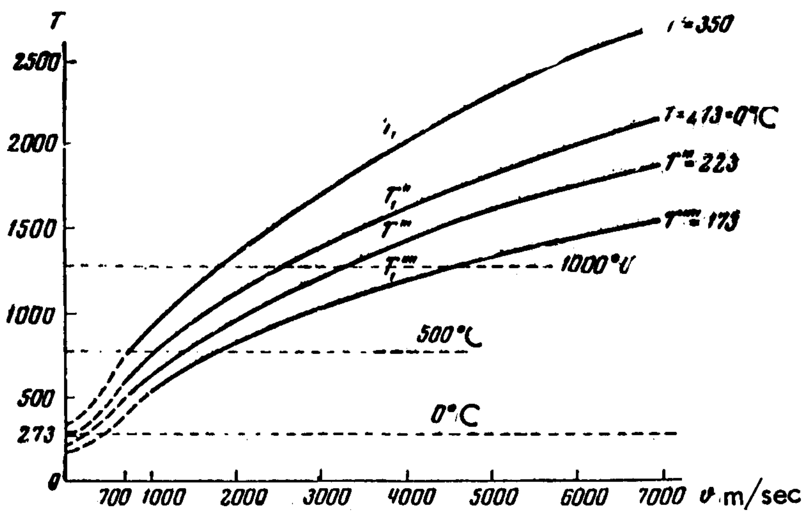

Figure 6. $T_1 = 0.09\ T^{0.71}\ v^{0.582}\ m^{0.291}$; m = 29.3

A graph is constructed according to this equation for m = 29.3 (fig. 6). The equation gives the air temperature in front of a surface normal to the trajectory; this temperature will obtain only ahead of the frontal parts of the rocket, the nose and leading edges of the foils, while near sloping surfaces the pressure and corresponding temperature will be considerably less. If we protect the oncoming frontal regions with some kind of refractory material, the remaining external surfaces of the rocket, if made of steel, should be able to withstand velocities up to 4500 m/sec, even without special refractory protection. Calculations of the temperatures of rapidly moving bodies are given below in section IX. Here we will use a second means of calculation - according to equation (37), but with allowance for the ameliorating circumstance that we do not make our surfaces normal to the trajectory but with a slight angle of attack, so that the air compression ahead of them, hence their temperature, will be still considerably lower. At the instant the rocket acquires a velocity of 4500 m/sec, it will be moving through the rarefied /579 layers of the atmosphere, in addition to which (see page 92) there is no longer a need for airfoils.

The data are no less favorable if we proceed from the fact that dumdum bullets, charged with mercury fulminate, do not fly apart spontaneously in air, having an initial velocity up to 700 m/sec and being so small that they are able to heat up quite considerably during their flight. The detonation temperature of mercury fulminate is 185°C, wherefore it may be assumed that the bullets are not heated to more than 150° above the air temperature. We presume that the absolute temperature of the surfaces of a moving body is proportional to some power (x) of the mean (square) molecular velocity of the gaseous medium relative to this body. Then, knowing that the mean molecular velocity of air at 0°C is 460 m/sec, we determine the mean velocity of the same molecules relative to a bullet flying at a speed of 700 m/sec:

$$v = \sqrt{460^2 + 700^2} = 837 \text{ m/sec}$$

We formulate the equation

$$\left(\frac{837}{460}\right)^x = \frac{T_1}{T};$$

substituting $T = 300°K$ and $T_1 < 450°$, we obtain $x < 0.7$. Consequently, we obtain

$$T_1 < T\left(\frac{u^2 + v^2}{u^2}\right)^{0.35}$$

(where u is the mean molecular velocity, v is the velocity of the moving body).

According to this equation, with $v = 4500$ m/sec, we obtain $T_1 < 800°C$ for $T = 220°K = -53°C$.

IX. Extinction of the Return Velocity by Atmospheric Resistance

On returning to earth, we need to reduce the velocity of the rocket to zero; consequently, the resistance of the atmosphere will consistently work to our advantage, and our only task is to utilize it optimally without burning up the rocket from motion through the atmosphere at velocities of several km/sec. The resistance of the atmosphere has a twofold application: 1) to extinguish the entire return velocity w_r = 11,185 m/sec; or 2) to extinguish just the "circular velocity," which is

$$7909\ \mathrm{m/sec} + \alpha = \frac{w}{\sqrt{2}} + \alpha,$$

where α is for the moment, lacking reliable information on the upper layers of the atmosphere, an indeterminate multiple of ten meters per second; the latter /580 is somewhat simpler technically; we will first consider the extinction of the last 7909 m/sec + α. We will adopt the following as our starting assumption: The rocket moves along a parabolic or prolate elliptical orbit, the vertex of which is situated at a distance of 400 to 600 km from the earth's surface, Depending on how precisely we are able to control the direction of the rocket; we must not only guarantee that the rocket will not fall to earth, but that it will not be torn apart in the dense layers of the atmosphere. The subsequent conversion of the trajectory is executed in application to its tangential type, except in the reverse order of that shown in figure 1. As soon as the rocket is in the segment of closest approach, it imparts deceleration to itself, reducing the eccentricity of the orbit and remaining at approximately the position of closest approach. When the eccentricity is diminished enough that it can no longer be detected by the pilot, the rocket will continue to impart small decelerations on arbitrary segments of its almost circular orbit. Each deceleration must be so slight that the resultant eccentricity will be barely noticeable; after each deceleration, the orbit is again traversed (the time to circle the earth is 1½ hr) and, in the event an appreciable eccentricity is detected, it is corrected by a slight deceleration on the segment of closest approach. In this way, the orbit of the rocket will be continually narrowed, consistently maintaining its circular pattern, insofar as permitted by the error of detection. This constriction continues until the orbit is in atmospheric layers of such density that j_ρ attains a value of about 0.1 cm/sec^2. From this moment on, the rocket ceases to function as such, and all objects of the proportional passive load are ejected. The construction of the rocket by this time should be as follows (see fig. 7): 1) pilot compartment; 2) an elliptical lifting surface, the design of which will be discussed below; the major axis of the ellipse should be perpendicular to the trajectory, the minor axis inclined at an angle α (about 40°) yielding the maximum lifting power; 3) a long tail section emerging from the pilot compartment to the rear at an angle α with respect to the minor axis of the elliptical lifting surface; at the end is a tail in the form of two plane surfaces forming an included angle of about 60°, its edge parallel to the major axis of the elliptical lifting surface and its bisecting plane parallel to the trajectory; 4) a surface for automatic maintenance of lateral stability, in the form of an angle similar to the tail but with less opening (about 45°), located above the pilot compartment and having its edge perpendicular to the trajectory and edge of the tail. This surface automatically maintains lateral stability of the vehicle by turning to the right and left /581 about its edge under the control of a gyroscope located in the pilot compartment. The axis of the gyroscope is fixed beforehand parallel to the axis of

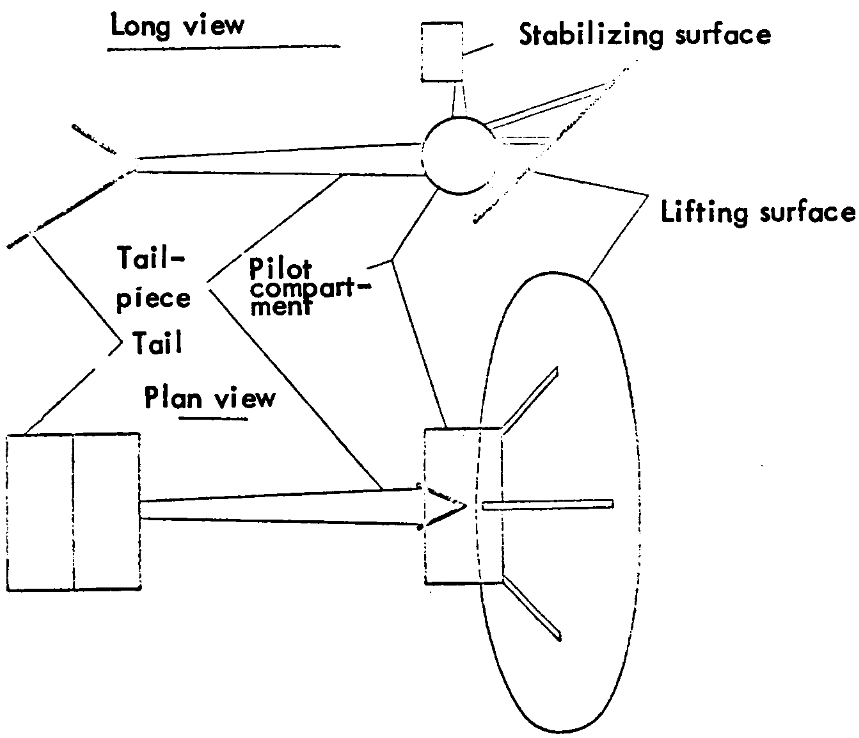

Figure 7. Diagram of a Vehicle for Suppressing the Return Velocity by Atmospheric Resistance.

earth's rotation. It is probably not feasible to achieve lateral stability of the vehicle at very large velocities in the rarefied layers of the atmosphere by purely aerodynamic means, wherefore some kind of automatic control device will be needed, such as the one indicated above. All of the indicated external parts should be carried aboard the rocket in disassembled form and assembled at the instant the orbit, or the part of it nearest the earth, passes through significantly dense atmosphere. A gliderlike vehicle of the above construction (differing from the glider by its much greater angle of attack, tail construction, and lateral stability mechanism) will have the attribute of always remaining in atmospheric layers of such density that, at its present velocity, the vertical component of the air pressure on the lifting surface will be equal to the apparent gravity of the vehicle, i.e., the surplus of its gravity over /582 the centrifugal force that it develops will be equal to

$$K = gM\left(1 - \frac{2v^2}{w^2}\right) \tag{38}$$

(horizontal motion along the arc of a large circle is assumed). As the velocity of the vehicle decreases due to the retarding effect of the atmosphere, it drops into the denser layers of the atmosphere, maintaining the balance between the apparent gravity and lifting force developed by the lifting surface. If we assume that the vehicle executes its return in the equatorial plane in an easterly direction (v_1 = v - U) and that the load on the lifting surface is equal to p kg/m^2, then, according to equations (15) and (38), we have

$$p\left(1-\frac{2v^2}{w^2}\right)=K\cdot 10^{-4}(v-U)^2\Delta c_\alpha, \tag{39}$$

where c_α is a function of the slope angle of the lifting surface. The left-hand side of this equation represents the apparent gravity of the vehicle per square meter of lifting surface, the right-hand side is the vertical component of the atmospheric resistance, i.e., the lifting force per square meter. On the basis of this equation, with p = 200 kg/m^2, c_α = 0.7 (α = 40°), and k = 0.1 (we choose the smaller of the experimental values of k, being the less favorable, in lieu of data relating to such high velocities), we draw a graph of the function h = F(v_1) according to equations (39) and (17) (fig. 8). The figures given with the curve denote the ratios $\Delta = \rho_h/\rho_0$ corresponding to the values of v_1 plotted on the horizontal axis. The part of the curve for v_1 < 1000 m/sec, is not plotted, since it is of no importance as far as we are concerned, for reasons discussed below. Extinction of the return velocity by atmospheric resistance is possible so long as the vehicle does not burn up in the air like a meteor at the values of v and h that will be encountered during descent according to equation (39); let us examine this condition: Inasmuch as the amount of heat given off (primarily through radiation) by the lifting surface of the vehicle at the highest temperature that it can withstand will not be /583

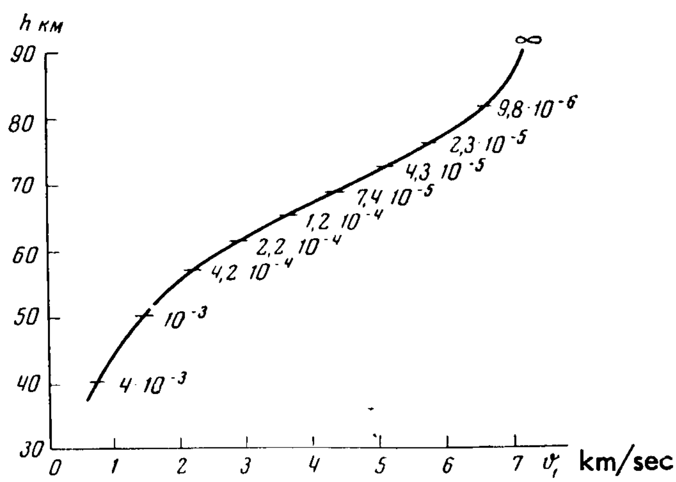

Figure 8. The Numbers on the Curve Denote the Ratios $\Delta = \rho_h/\rho_0$ Corresponding to the Values of v_1 Plotted on the Horizontal Axis, h is Computed from the Values of Δ According to Equation (17).

less than the amount of heat it acquires from the volumes of air in front of it, which is heated to incandescence by adiabatic compression, given different combinations of v and h corresponding to equation (39). We cannot formulate an exact notion as to the indicated phenomena for the lack of precise information on the effects occurring in an elastic medium near a moving body or on the radiative power of gases at temperatures of several thousand degrees. Since the radiation intensity increases as the fourth power of the absolute temperature, the surfaces of the vehicle supported by the atmosphere, above all its lifting surface, must have the greatest heat resistance, which means increasing their weight per square meter p. The most logical construction for the supportive tail and stabilizing surfaces is the following: a metal framework, thickly coated with a tile of some highly refractory material, as, for example, graphite, retort carbon, limestone, or porcelain. The tile should be on the side of the surface facing forward, so as to protect the metal frame. The parts of the frame coming in direct contact with the tile should be made of one of the higher-melting metals, the base of which might be tubular steel cooled from within by water and steam (the hazardous period of the descent will last less than 20 min) and protected against radiation from the back side of the tile by a porcelain lining. There is clearly no danger of considerable scorching of the carbon-bearing tile, because when the vehicle is traveling at a speed of several km/sec only the molecules of a very thin adjacent air /584 layer will come into direct contact with its surface. The total amount of air in the volume described by the contour of the vehicle will only be a few times the mass of the vehicle as the latter decelerates from v_1 = 7000 m/sec to 2000 m/sec (hazardous interval). It is very probably that at altitudes $100 > h > 50$ km, the atmosphere is very impoverished of oxygen, whose molecular weight is higher than the molecular weight of nitrogen, and the hazardous velocities will occur at heights $100 > h > 50$ km.

In view of the fact that the hazardous velocities are several times the velocity of sound in air, only the surfaces of the vehicle facing forward will be exposed to the intense action of the atmosphere, while near the backward-facing surfaces will be almost complete void in comparison with the density of the surrounding atmosphere. In particular, the metal frame of the surfaces and the entire pilot compartment will be located in this void if properly constructed; the compartment should only be protected against radiation from the back side of the tile.

An approximate comparison of the possible quantities of heat liberated and gained by the lifting surface seems to indicate that the vehicle can return comfortably to earth, extinguishing the return velocity beginning with v = 7909 m/sec = $w/\sqrt{2}$. The work done by the vehicle on the atmosphere (independently of the inaccurate equations (17) and (15)) attains a maximum of Q, equal to about $3\ p\ 10^{11}$ erg/sec per square meter of lifting surface, with V_1 equal to about 4500 m/sec. Less than half of this work will be dissipated in one direction by the lifting surface: $Q_1 > 1.5\ p\ 10^{11}$ erg/sec, whereas the other, larger part will be radiated by the compressed volumes of air in the other direction - into space; if we assume that during the passage of air alongside the surface of the vehicle (in the most hazardous period of flight this time will not be more than 0.002 sec), it will radiate a part of its heat, equal to qQ, where Q is the total amount of heat acquired by it in compression, then the lifting surface will acquire at most

$$qQ_1 < 1.5 \text{ pq } 10^{11} \text{ erg/sec} \tag{40}$$

of radiative energy.

According to the Stefan-Boltzmann law, the radiation intensity from a complete black body is equal to $0.57T^4$ erg/sec per square meter of surface. We use a black body here, since in the foregoing case we assumed total absorption of rays by the lifting surface; affecting absorption and radiation identically, the absorption coefficient is of no importance for our purpose. If we let p = 200 kg/m^2, which is a representative and probable figure, and T = 3000°K = /585 = 2730°C (a value near the maximum possible temperature), it turns out that the radiation power per square meter of lifting surface could attain a value of $9.2 \cdot 10^{13}$ erg/sec in both directions, whereas the absorbed energy will not be more than 3.10^{13} q erg/sec (eq. (40)). Judging from the fact that the gases in the cylinders of internal combustion engines, during a period on the order of 0.1 sec, are only able to give up half of their heat to the cylinder walls, we are safe in assuming that the ratio q has a value expressed in less than hundredths. We therefore obtain a very large allowance for reducing T = 3000°K and for increasing the surface load p = 200 kg/m^2.

We now present an alternate calculation of the temperature of the lifting surface. According to equation (37), at a velocity of 4.5 km/sec (we choose this velocity as the one yielding maximum work due to friction), the temperature of the air compressed adiabatically at an initial temperature of 0°C is T_1 = = 1800°K. Since the lifting surface absorbs thermal radiation in one direction, whereas both sides radiate energy, and since the amount of heat radiated must be equal to the amount absorbed, we have the relation

$$aT_1^4 = 2bT_2^4,$$

where a and b are coefficients proportional to the absorption coefficients of the heated gases and lifting surface, T_2 is the unknown temperature of this surface. Assuming a = b and substituting T_1 = 1800°, we find T_2 = 1500° = = 1227°C. Actually, the absorption coefficient for a solid will be higher than for a gas, so that T_2 should be even less. It follows from the preceding calculations that the lining of the lifting surface can be made of porcelain or corundum tile.

After the velocity of the vehicle drops to $v_1 = 2000$ m/sec, any danger from overheating will be eliminated (see eq. (33) and fig. 5).

Further loss in velocity takes place just at the moment the vehicle is found at an altitude of one or two kilometers above the earth's surface level. Since we are unable to calculate precisely beforehand the position of descent and since it will be impossible to know beforehand in the first flights whether to land the vehicle on the ocean or dry land, a direct landing on the surface of the earth at the velocity v_1, which is equal to several multiples of ten m/sec, would present danger to the life of the pilot; the vehicle should therefore be equipped for descent by parachute. If it proves convenient to carry a parachute of sufficient areal span, it can be used to bring down the entire vehicle; but if such a parachute is too bulky, then one will have to be devised for just the pilot, letting the vehicle make a separate landing. /586
If the descent is made over the ocean, landing can be made directly on the water. In this case, the steepness of descent and, hence, the abruptness of landing should be minimized ahead of time at altitudes of 10 to 20 km by decreasing the angle of attack of the lifting surface by rotating the tail section downward through some angle. The landing speed (horizontal) will thus be increased, but the impact will be diminished. For the case of maneuvering in air, which is necessary for descent onto the ocean, the tail section or tail itself must be steered by controls in the pilot compartment. Considering the possibility of descent over the ocean, the vehicle must be provided with whatever is needed staying afloat; it should have a sail, equipment for imparting stability on the water, if this is necessary, a small fuel supply in the form of compressed marsh gas (methane), and a lightweight low-power motor.

With these means, utilizing the tradewinds, the vehicle should reach the nearest land in a reasonable period of time, unless it is picked up earlier by some ship. To facilitate floating, the lifting surface and other similar parts should be ejected or disassembled and packed into the compartment. To extinguish the total return velocity by atmospheric resistance, the initial status should be the same as in the first case (see page 100); the control of the rocket is also the same as before, with the additional fact that its lifting surface has a variable angle of attack from +40° to -40° and is furnished with an automatic mechanism, which orients it in a positive angle of attack when the rocket enters the deeper layers of the atmosphere, at zero angle when the rocket flies parallel to the earth, and at a negative angle when, moving away from earth, the rocket flies into more rarefied layers of the atmosphere. This mechanism can be regulated by a control cable from a special fin situated on the outside perpendicular to the motion of the rocket. When the encountered atmospheric pressure increases on this surface, the mechanism should operate in one direction, imparting a positive angle of attack to the lifting surface; when the pressure diminishes it should operate in the opposite direction. In order not to subject the back side of the lifting surface to the action of the atmosphere, it may be possible, instead of imparting a negative angle of attack, to induce the entire rocket to rotate about its longitudinal axis. Cautiously, with small decelerations at the maximally distant point of the initial ellipse, the orbit of the rocket is constricted, the point of closest approach finally entering the confines of the significantly dense atmosphere. This entry should take place at a distance from the earth's surface such that the rocket will operate with guaranteed safety, with allowance for possible control errors and errors in determining the data of its orbit, against overheating at velocities up to 11 km/sec. /587 Also depending on this requirement is the choice of dimensions for the axis of the initial ellipse; the smaller the major axis, the greater will be the accuracy with which the point of closest approach to the earth can be calculated and the more delicate will be the approach to it (in particular, such that the perturbing influence of the moon will be minimized,) but, clearly, the greater will be the part of w_v that must be preliminarily extinguished purely by the function of the rocket. At the instant the segment of closest approach enters the rarefied layers of the atmosphere, the rocket begins to follow a trajectory completely analogous to the trajectory of the preliminary (external with respect to the atmosphere) phase of the return flight in extinguishing the velocity $w/\sqrt{2} + \alpha$ prior to the transition into circular orbit (see page 100), the only difference being that deceleration will not be provided by the action of the rocket on the segment of closest approach, but by the resistance of the rarefied atmospheric layers, which the rocket will pass through several times with an ever-diminishing major axis on the part of its orbit.

The automatically varied angle of attack of the lifting surface will play the following role in this operation. On penetrating deeper into the atmosphere, when the pressure on the control surface increases the angle of attack will be positive and the lifting surface will function to prevent the rocket's approach to earth, maintaining it in atmospheric layers rarer than the rocket would otherwise penetrate. When the rocket begins to emerge from the atmosphere and the pressure on the control surface diminishes, the angle of attack becomes negative and the lifting surface prevents the rocket from moving away from earth, thus emerging from the atmospheric layers at a smaller angle, so that the next entry therein is at a smaller angle and so that the penetration into the atmosphere on the next pass of the segment of closest approach is shallower. Consequently, by means of a variable angle of attack on the part of the lifting surface, it is possible to move the segment of closest approach away from the earth into more rarefied layers of the atmosphere, beginning with the first entry of the orbit into the significantly dense part of the atmosphere and continuing until the rocket goes over, as the result of the slowing action of the atmosphere, into a circular (essentially spiraliform) orbit contained entirely within the atmosphere, after which the remainder of the descent is executed exactly as in the case when the return velocity is extinguished by atmospheric resistance according to the first method.

Thus, by the second method, we extinguish 11,185 m/sec $- \beta$ instead of 7909 m/sec $+ \alpha$ by atmospheric resistance, where β is the amount of rocket deceleration expended in transition from T_c to the initial ellipse and entry of the point of closest approach of the initial ellipse into the confines of the atmosphere; β is a quantity which theoretically can be as small as we like /588 and essentially is determined by the accuracy with which the rocket can be controlled and its orbit data computed. Approximately speaking, considering the thickness of the atmosphere to be insignificant in comparison with the radius of the earth,

$$\beta = \sqrt{2\frac{R}{r_1}Rg}\left(1-\sqrt{\frac{r}{r+r_1}}\right)+$$
$$+\sqrt{2\frac{R}{r}Rg}\left(\sqrt{\frac{r_1}{r+r_1}}+\sqrt{\frac{R}{R+r}}\right),$$

where R is the radius of the earth, r_1 is the distance from the center of the earth to the point of closest approach (perigee) of the initial ellipse, r is the corresponding distance of the point of farthest approach (apogee). The first term represents the rocket deceleration necessary for transition from T_c to the initial ellipse; the second term is the deceleration required for entry of the initial ellipse perigee into the atmosphere. If we assume as approximate data $r_1 = 2R$ and $r = 20R$, we obtain approximately $\beta = 0.05\sqrt{2Rg} = 0.05\ w =$ = approx. 550 m/sec. Consequently, we can utilize atmospheric resistance to quench a portion of w_v equal to 10,360 m/sec, and W becomes equal to about 12,550 m/sec.

X. An Interplanetary Base and Rocket-Artillery Device[8]

Velocities less than half the exhaust velocity u generated by whatever chemical compound is employed, i.e., up to about 2500 m/sec, overlooking the petroleum-air group (see page 64), can be developed more economically from the viewpoint of fuels and materials expended (in the objects m_1) by artillery means, but man is totally unequipped to take artillery type accelerations. It would be desirable, therefore, to deliver cargo and all objects of the passive load capable of withstanding accelerations of several thousand m/sec² (with suitable packaging, everything except delicate instruments) into interplanetary space by a rocket-artillery device, separately from the human passengers. With the rocket-artillery transportation of loads into space, we would economize as much as 50% on charge materials.

The difficulty of such a device lies in the problem of locating such a relatively small body in space as a rocket vehicle fired from earth. Looking ahead to the time when flights will be made more or less regularly, we propose the following technique for organizing them and arranging them so as to yield considerable material economy. /589

A rocket of large mass is sent from earth with a supply of active load for the development of a velocity W of about 12,000 m/sec. The final mass M_f of

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[8] The author, unfortunately, did not have on hand a manual giving the optical power of modern telescopes, and it must be remembered that the problem of signalization with a "rocket-artillery device" has to be developed on the basis of data that are not quite reliable.

this rocket, due to the smaller required W, will be $\sqrt{n_1}$ times the final mass that the rocket could have if it had the same mass M_0 but designed for flight with a return trip to earth without absorbing the return velocity by atmospheric drag. This rocket then becomes a moon satellite with as large an orbit as is possible without succumbing to the hazard of being drawn back to earth, after which it unfolds a large signaling surface of material with a large visible-light reflecting power in relation to its weight per square meter. The unfolded surface may be as much as a hundred thousand square meters in area, since with a material thickness of 0.1 mm and absolute (bulk) density equal to unity, it would provide exactly 10,000 m^2; this surface will be freely discernible and locatable from terrestrial observatories. Near this signal surface there should be set up an interplanetary base for flights throughout the solar system. The realization of this base, irrespective of its rocket-artillery equipment, will provide the enormous advantage that we will not have to transport materials, instruments, machinery, and personnel with accessory compartments from earth into space and back again on every single trip, just as we will not have to discard objects of the first categories just to avoid the expense of transporting them back to earth. All of this will be stored on the base, whereas flights from the base to anywhere and back again will require $1/\sqrt{n_1}$ times the material expenditure that the same trip from earth would require. Rockets will be sent from earth into interplanetary space only to equip and supply the bases and to alternate the personnel staff after fairly prolonged periods of time. If rocket-artillery transportation proves feasible, then, above all, we will obtain about a 50% savings by setting up the equipment on a base in interplanetary space.

The base should initially have the following items:

1) Personnel - a minimum of three men with a chamber for themselves and everything needed for their sustenance.

2) A powerful telescope (a reflector, as possibly being more lightweight with the same diameter).

3) A small two-man rocket with a store of charge for developing W = 2000 m/sec and two telescopes of successively smaller power but larger field of view than the large base telescope.

In order to preclude rocking or oscillation of the base, which could /590
interfere with observations in a large astronomical instrument, its mass should be divided into four parts, arranging them at the vertices of a tetrahedron and joining them with aluminum girders (these girders are not required to have great strength or, consequently, large mass, since there will be no external forces acting on the base and the gravitational force on it will not be of consequence). So constructed, the base will have an incomparably larger moment of inertia relative to any axis and a correspondingly larger stability in space. Should the personnel be adversely affected by continued absence of an apparent gravity, only a compartment for telescope observations need be connected with

the described tetrahedron, and the living quarters could be constructed separately and attached by a cable several dozen meters in length to a counterweight. If to this system rotation is imparted about a common center of gravity, a centrifical acceleration will be created, which will be perceived just as gravity on the earth. In order to provide the living quarters with as much space as possible for the same mass, the air pressure inside it must be minimized insofar as possible. Experiments will have to be conducted toward this end, in order to evaluate the habitation of less dense air than that which we breath, but with a higher oxygen content.

Communication from earth to the base is effected by light signals, using a high-powered projector with a low scattering angle, set up on earth at a place known to the base; the signals from this projector must be detectable by the large base telescope. The base can communicate with the earth by means of a lightweight metal reflector of large area,[9] aimed so that the sun's rays will be reflected toward some observatory on earth. The area of this reflector should not be too large for the signals to be observable in a large telescope.

The rocket-artillery delivery of cargo to the base is carried out as follows.

On command or at a preappointed time, a rocket-vehicle is fired from a cannon on earth, of which more will be said below, carrying equipment supplies for the base. The flight of the rocket-vehicle is computed so that it will land on the base; inasmuch as such precision is not actually possible, the path of the rocket-vehicle will pass a thousand or hundred so kilometers from the base. The relative velocity of the rocket and base at the instant of closest /591
approach must be minimized, hence the instant of closest approach must coincide with the instant of maximum distance between earth and the base. The orbit of the rocket-vehicle relative to the moon should be hyperbolic with as small an angle as possible between the asymptotes. From the moment the rocket-vehicle is fired, light signals are sent out automatically and periodically, possibly by detonations of a mixture of magnesium and saltpeter. The period from one signal to the next should be such that the rocket-vehicle cannot escape from the field of view of the large base telescope during the interim, because once the vehicle is lost from sight, to find it again would be impossible except as a matter of pure luck. After the rocket-vehicle traverses its segments I_j, it automatically unfolds a signal surface of lightweight white fabric, similar to the base signal area. From the instant of firing, the large base telescope, which is aimed beforehand toward the point from which the firing is to take place, does not let the rocket-vehicle stray from its field of view, tracking it by its light signals on the interval I_j, then by the signal surface.

Shortly before the rocket-vehicle makes its closest approach to the base, when the rocket first becomes clearly discernible in the larger of the two telescopes on board the small base rocket, the latter is sent out to meet the

[9] A rational construction for the reflector would be a thin planar reflective metal sheet stretched out over a lightweight metal Duralumin skeletal framework.

rocket-vehicle, reducing its velocity to zero as it approaches, attaching to it, and towing it to the base, using the charge available on the rocket-vehicle if necessary.

Because the rocket-vehicle necessarily carries certain instruments and mechanisms which, in their assembled form, could not favorable withstand accelerations of several ten thousand m/sec^2, the cannon for firing the rocket-vehicle must be of considerable length, approximately 2 km. With this length, the required acceleration can be reduced to approximately 100g. Specially designed mechanisms can tolerate this kind of acceleration. The cannon could be a tunnel built in solid rock; in order to render the motion of the vehicle perfectly straight, the entire length of the tunnel should be fitted in quadrants with four carefully aligned metal guide strips, whereas the finishing of the intermediate regions can be fairly coarse. Owing to the considerable length of the cannon and the correspondingly low pressure of the gasses therein, compared with present-day artillery pieces, and owing to its large cross section, the burst of escaping gases through the 1 or 2 mm gap between the tunnel walls and vehicle will not be appreciable in comparison with their total quantity.

XI. Control of the Rocket, Measurement and Orientation Instrument /592

For control of the rocket and orientation (navigation), the ship must have the following instruments:

1. An indicator of the apparent gravity inside the rocket, designed on the principle of spring-loaded weights with a suspended load; the indicator pointer will read the value of the apparent gravity directly. To the indicator should be attached a rotating drum for permanent recording of the readings. The area bounded by the resultant curve will be expressed by

$$\int_0^t (j_0 - j_p)\, dt = W - L_d.$$

This indicator should be connected to an automatic exhaust intensity control, so that the acceleration I_0 on the interval I_j will maintain the required value (equal to I_{max}). There should be two such indicators: one for the large accelerations up to and including I_{max}, another for small accelerations, from 0.01 to 10 cm/sec^2. The first indicator will serve for departure on I_j and throughout the remainder of the flight, the second for entry of the rocket orbit into the atmosphere on the return trip. It would be unsuitable to measure accelerations of 1000 cm/sec^2 and decelerations of 0.01 cm/sec^2 on the same instrument.

2. An indicator of atmospheric resistance, in the form of a plate projecting externally from the rocket, connected by cables with the interior of the rocket. Due to friction in the linkages, such an instrument for determining

atmospheric resistance will not be used in place of the first type of indicator at the beginning of entry into the atmosphere, because it could have sufficient sensitivity.

3. An indicator of the rocket mass, giving readings that depend on the readings of instruments for taking into account the expenditure of charge. Connecting the second and third indicators, we obtain an indicator of the deceleration associated with the force of atmospheric drag. Connecting this indicator with the first, we obtain an indicator of the intrinsic rocket acceleration of the rocket j_0; integration of the latter record yields the magnitude of the velocity developed, W.

In order to automatically prevent the rocket from rotating about its own axis, which can happen as the result of insignificant, random errors in the construction of the rocket, it should have a gyroscope whose axis is aligned perpendicular to the axis of the rocket. The axis of this gyroscope should be free and able, by its movements relative to the body of the rocket, to control rotating surfaces which are inserted in the exhaust stream. In order to im- /593
part automatic stability or an automatic preset rotation of the longitudinal axis of the rocket, the latter should be equipped with a second gyroscope whose axis is parallel to the rocket axis and which controls other rotating surfaces placed in the exhaust stream.

For orientation of the pilot, special types of astronomical instruments and techniques need to be developed so as to achieve the most rapid and precise determinations of the position of the rocket and data relating to its orbit relative to the earth. These determinations are of utmost importance and require extreme precision just before attenuating the return velocity by atmospheric resistance. In order to impart greater stability to the axis of the rocket during free flight in empty space, procedures similar to those indicated on pages 107 to 109 may be instituted.

XII. The General Outlook

The essential factor governing the future potentials of space travel, at least in its first exploratory phase, is the amount of passive load, i.e., n, since this quantity dictates the economic aspect of the matter, which theoretically does not present any real difficulties. The amount of charge, or fuel, so to speak, and hence the approximate cost of flight (utilizing the object of the proportional passive load, (see page 72)) are proportional to the quantity (n - 1). In the table (pages 64 to 66) are given the values of n corresponding to the total calorific value of various chemical compounds and rocket velocities W_1 = 22,370 m/sec and W_2 = 14,460 m/sec. The first velocity corresponds to flight from the earth into interplanetary space and back again without quenching the return velocity by atmospheric resistance, the second corresponds to the same flight with quenching of the last 7900 m/sec of velocity by atmospheric resistance. Until suitable experiments have been done, we cannot know what the rocket's performance will be, nor do we know just what chemical compounds or what percentage proportions of the latter will prove most suitable to use. Right now, we will use for our approximate calculations an average efficiency value of 0.8 for the total flight, this being a fairly probable value, according to conjectural computations, which we will not give here, as well as data on the power developed by heated gases in internal combustion engines. We assume 3.3 kcal/g as an average value of the total calorific value. With these data, we obtain u = 4700 m/sec (note 15); this hypothetical value of the exhaust velocity, given the present lack of opportunity to gain a more dependable value, will be used as the basis of the ensuing calculations, assuming that the error in calculating n does not exceed the factor $n^{1/10}$ in either the plus or minus direction. In view of the relative insignificance of the velocity L_s, as explained in section VIII, we will let W_e = 12,000 m/sec, neglecting differences for which the value and even the sign are not known and which will most likely be in our favor (section VIII, page 99). /594

With these data and with the mandatory requirement for utilization of the objects m_1 (in the event that it is necessary to use a multiunit system; see section V) for purely rocket-powered flight from earth into interplanetary space and return to earth without quenching the return velocity by atmospheric resistance, we obtain from equation (4) n = 120, i.e., about 120 weight units of fuel **per weight unit of payload,** a significant part of the former being in the form of liquid oxygen or ozone, the remainder in the form of liquid CH_4, C_2H_2, SiH_4, BH_3, as well as one, not too minor portion, equal to $q\mu$, in the form of metal (mainly Duralumin) objects of the highest quality, namely the objects m_1. The cheapest petroleum compound of the charge will also be of use, but this advantageous (despite the required increase in mass of the charge) use is severely limited by the fact that, corresponding to the increased charge mass, the mass of the most expensive of the expendable portions of the rocket - m_1, the proportional passive load - must also increase. For a flight under the same conditions and with the same data, with stopover on the moon, n = 1000, or the same with stopover on Mars, n = 3000 (using the tangential type of trajectory, continued until the required hyperbolic velocity relative to earth is acquired). The latter figures can be reduced somewhat favorably by the advantageous use of more expensive and thermally efficient compounds - of the boron and borohydride types. It would be impossible to regard such potentials as satisfactory; each flight would require tremendous expenditures of material, and the possibility of taking along such heavy loads, materials, and machines would be completely lacking, due to the same economic aspect of the problem. Even the transportation of a large modern astronomical instrument would require colossal expenditures.

The key to the actual conquest of outer space is contained in the following: first of all, extinction of the return velocity by atmospheric resistance (section IX), then the building of an interplanetary base (section X) and, if it can be done, the necessary light signaling and artillery rocket transportation of supplies to the interplanetary base. The extinction of the return velocity by atmospheric resistance according to the first method, diminishing W to 14,460 m/sec, provides a sixfold reduction in n for all flights: from earth into interplanetary space and back, n = 20; the same with stopover on the moon, n = 160; the same with stopover on Mars, n = 500. With extinction by the second method, when we have W = 12,500 m/sec, a 12-fold reduction in n is obtained, whence we have $n_e = 10$, $n_m = 80$, $n_M = 250$, respectively (note 16). The reduction in n in this case can be compared favorably with the use of cheap /595 petroleum compounds in large relative amounts in the charge or with a smaller consumption of objects from the proportional passive load as charge.

For the same flights from an interplanetary base, we would have values of n reduced still more by a factor of 1/11: $n_e = 2$ (return from the base to earth); with n so near unity, we no longer ignore the difference between n and (n - 1); (n - 1) = 1 in this case, i.e., one unit of fuel per unit of payload (assuming extinction of the return velocity by the first method; in the second method, the extinction of return velocity requires an altogether inconsequential quantity of charge). $n_m = 15$; $n_M = 45$; the nonreturnable transportation of loads from the base would require: to the moon n = 4, to Mars n = 7.

The transportation of loads from the earth to the base by purely rocket-powered means yields n = 11; by an artillery rocket device, n = 7; for a value of $n < 20$, in all probability, we could use just one cheap petroleum compound with greater economic advantage; for n = 10-15, it is no longer necessary to consume objects of the proportional passive load. Under such conditions, the payloads - high-quality materials and machines - with delivery to the moon and even Mars would be just slightly more costly than to earth. We have assumed throughout that landing on Mars will be executed without the aid of velocity extinction by resistance of the atmosphere on that planet. However, on Mars there is clearly a rather dense atmosphere, whose resistance could be utilized by the rocket for gliding descent, just as indicated for earth in section IX. The force of gravity on Mars' surface is only half as great, while the velocity w_M is less than half the value for earth; the operating power of the gliding rocket above Mars' atmosphere at the instant maximum value is reached will be, consequently, one sixth the value for gliding into the earth's atmosphere, so that danger of heating of the rocket surface will be nonexistent. The only remaining hazard stems from the structure of the surface on Mars, of which we know little, and from the presumed inhabitants thereon. In descending to Mars with velocity extinction by atmospheric resistance, the transportation of loads to Mars would cost about the same as to the moon, which is devoid of a dense atmosphere.

XIII. Experiments and Investigations

Considering the deficiencies in our knowledge in certain areas and lack of experience in the building of rockets for large velocities, before we can set about to build or design rockets for flights into interplanetary space, we need /596 to perform certain scientific and technological investigations; the main ones include:

I. Investigation of the functioning of the combustion chamber and exit tube of the rocket in media of various density and expansibility; determination of the best combustion chamber and exit tube constructions; determination of the most favorable shapes and length for the exit tube, techniques for injecting the charge materials into the combustion chamber, the relations between exhaust mass dM/dt, dimensions of the combustion chamber, and the cross section of the exit tube.

Investigations of the operation of the rocket in an atmosphere of low expansibility (buoyancy) can be performed by leading the exit tube of a small model into a chamber from which the gases have been evacuated by a pump with a large volume capacity. For reducing the pressure without further increasing the dimensions of the evacuation pump, the chamber should contain a thick water spray, which will condense all the constituent parts of the exhaust except carbon dioxide, while the latter will be cooled, greatly enhancing the evacuation process. For even greater rarefaction, chemical compounds can be used which do not produce any carbon dioxide in the exhaust; however, with a pressure in the chamber of 0.01 atm, the functioning of the rocket will already be scarcely different from that in empty space.

II. Determination of the best constructions for all objects of the proportional passive load and techniques for their utilization as charge materials.

III. Investigation and development of production techniques for the charge materials until routine factory methods are available, for example, for the production of BH_3, SiH_4, O_3, C_2H_2, CH_4.

IV. Determination of the best designs of compartments for personnel and all attendant instruments.

V. Determination of the best designs for automatic control and orientation (navigation) instruments.

VI. Investigations of the endurance of the human organism with respect to mechanical acceleration and life in air under reduced pressure with an elevated oxygen content.

VII. Determination of improved methods and types of astronomical instruments for rapid pilot orientation relative to the position of the rocket and its orbital data. Cautious training in such flight conditions in a simulated environment; the earth or other celestial body should be replaced by a large hemisphere, around which the trainee, housed in a chamber of the same dimensions and construction as the one intended for the rocket, will float in still water on a slowly moving stable raft.

VIII. Investigation of the atmosphere at altitudes to 100 km. This /597 investigation can be performed by means of projectiles or rocket-missiles fired from conventional large (naval) artillery pieces. On reaching the apex of its trajectory, the projectile should automatically eject as large a parachute as possible, made of lightweight white fabric with a weight hung on it. By observing the rate of descent of this parachute from earth, we gain some notion as to the density of the atmosphere at different heights. If we equip the parachute, instead of with a weight, with an instrument that automatically collects a sample of the air, we can formulate a precise (in every aspect) notion as to atmospheric data at different heights.

IX. Investigation of the heating of surfaces of moving bodies and the resistance of an appreciably dense atmosphere ($\rho = \rho_0$). This investigation can be carried out with projectiles for smaller velocities, while for large velocities it should be carried out with rocket-missiles fired from artillery guns at a small angle with respect to the horizon, calculated so as to land in the water where they could be recovered. The surface of these missiles ought to be coated with materials having different high-melting indices, insulating them from the metal body of the missile by a layer of porcelain. The maximum heating temperature can be estimated from the state of this surface of the missile after it has completed its flight.

X. Investigation of the heating of the surfaces of bodies at large velocities in a rarefied atmosphere (see section IX), as well as investigation of atmospheric resistance at large velocities and the endurance of various constructions of supporting (lifting) surfaces, conducted with small (up to 10 tons) rocket test models. The beginning of the trajectory of these test flights is calculated as the value of T_e for flight into interplanetary space, but with the attainment of heights from 60 to 100 km (depending on the meteorological data obtained in investigation VIII) the trajectory should automatically assume a horizontal direction, going into a gliding descent on its lifting surface when the rocket charge is completely spent.

During ascent, the angle of attack of the lifting surface, i.e., the angle between its minor axis and tailpiece, must be small, increasing gradually to the full value (about 40°) at the instant of burnout. To determine the maximum heating temperature of the rocket surface, the same procedure as in investigation IX can be used. In order to automate control of the test rockets, they should be equipped with the two gyroscopes described (in section X) for the actual rocket. These test flights should be executed with a gradually stepped-up maximum v_1 at the instant of charge burnout; the same rocket can be used for these. Just one petroleum compound with $n < 6$ needs to be used for the charge. After the maximum v attains a value of 7500 m/sec and the test model descends /598 safely into the lower layers of the atmosphere, it will be possible, according to tests on objects of the proportional passive load of appropriate dimensions, to go directly to flight with personnel into interplanetary space, flying around, for example, the unknown backside of the moon.

COMMENTARY /662

This work by Yu. V. Kondratyuk was first published as a separate book in Novosibirsk in 1929. Work was begun on it, however, much earlier. According to the author's attestation, this book, which represents the outgrowth of an earlier paper ("To Whomsoever Will Read in Order to Build," included in the present collection), was written in 1920, then revised and re-edited in 1923-24. In 1925, the manuscript was sent to V. P. Vetchinkin, who gave it the following review in April 1926:

Review by Mechanical Engineer V. P. Vetchinkin of Yu. Kondratyuk's article, "On Interplanetary Voyages"

In his preface, the author states that he was unable to become acquainted with the achievements of foreign scientists in this field, nor did he even have access to fundamental works of Tsiolkovskiy. Nevertheless, this did not prevent the author from obtaining all the results that had been obtained by researchers in interplanetary travel as a whole, a fact which is indeed to his /663
credit.

At the same time, the totally unique language used by the author and his rather unusual expressions and notation, in terms of what scientists are accustomed to, gives every reason to believe that the author is self-educated, having studied at home the basics of mathematics, mechanics, physics, and chemistry.

Both of the above circumstances affirm the fact that mechanic Yu, Kondratyuk represents an enormous innate talent (of the same type as F. A. Semenov, K. E. Tsiolkovskiy, or A. G. Ufimtsev), having been cast off in some god-forsaken hole and having had no opportunity to utilize his capabilities in the right place.

We now turn to the work itself.

Section 1 presents the definitions pertinent to the rocket, to its load, and to the different segments of its trajectory.

Section 2 presents without proof the formula of K. E. Tsiolkovskiy relating the weight of the rocket and its fuel supply to the magnitude of the required velocity and reactive properties of the fuel.

In section 3, the problem of the possible exhaust velocity of the combustion products for various fuel agents is investigated in detail from the thermochemical point of view, insofar as this is possible with a total lack of experimental data and in light of Comrade Kondratyuk's lack of opportunity to perform the appropriate tests themselves.

Section 4 gives a formula indicating not only the advantage but the actual necessity of using several rockets in sequence (Obert proposes two rockets), since in the case of one rocket the fuel tanks must be so relatively lightweight that they could not possibly be built; here also he submits the proposal, similar to the proposal of F. A. Tsander (Moscow), that the tanks be burned up for fuel as they are spent, i.e., that the tanks be constructed from materials that can eventually be advantageously burned in the rocket. Obviously, the formula contained on page 15 (page 72 of the present collection) is in error and should be written in the form

$$\left[\frac{1}{1-q(N_c-1)}\right]^n$$

instead of the form given by the author:

$$\left[\frac{1+q\left(1+\frac{1}{N_c}\right)}{1-q\,(N_c-1)}\right]^n$$

In section 5, the very troublesome problems of the types of trajectories for rocket flight, the transition from one trajectory to another, the velocities required to do this, and the possible trajectories for escaping from earth and returning again are discussed. The very novel presentation and uncommon notation somewhat cloud the reading of this section; all of its results are correct, however; even in the problem of choosing a trajectory, Kondratyuk goes beyond the published literature, arriving at F. A. Tsander's notion of equipping the rocket with wings for flight in the atmosphere.

In section 6 is considered the problem of man's capability to withstand large accelerations in rocket flight. The author implies the proposition of Tsiolkovskiy that the pilot should desirably be situated in a reclining position and in a vessel containing water, but he adds to this the desirability of slowly rotating the man relative to his own lengthwise axis, so that the rush of blood and concomitant tendency of the blood to drain keep changing places in the human body, hence never have a chance to occur. The author bases his arguments on experiments with swings and giant strides (maypoles), indicating the possibility of imparting an acceleration of 3 g to a man without impairing his health. Stunt flying in modern acrobatics and military aviation have demonstrated the possibility of impart accelerations of up to 8 g; just how long /664
accelerations can be endured over a protracted time period, which is only possible on rotating (centrifuge) machines, cannot be said, due to insufficient experimentation.

In section 7, the effects of the atmosphere on the rocket are discussed. Apart from his inadequate knowledge of the laws of aerodynamics (the use of the time-honored Lössl formula) and the latest research in the composition, temperature, pressure, and density of the atmosphere, the author demonstrates a tremendous capability of coping independently with all of the indicated difficulties and, proceeding from the most general physical considerations, he computes the density of the atmosphere, the energy of its resistance, and the conditions anent heating of the rocket during flight through the atmosphere at high velocities, proposing that the rocket be outfitted with wings and rudders - all with complete validity yet obviously with total ignorance of the present state of aviation.

In section 8, the problem of quenching the return velocity by resistance of the atmosphere is investigated in more detail, the author giving a completely correct trajectory of descent. Here, however, the author's utter lack of acquaintance with aviation structures, control techniques, etc., crop up. The author again treats the problem of heating of the rocket at an altitude of 4 to 6 km above the earth and arrives at rather reassuring results.

In section 9, the author talks about a base station, which would have to be a satellite of the moon, and about sending materials and supplies there by means of an artillery rocket device, carrying no passengers.

In section 10, he talks about control of the rocket and the instruments needed, wherein the problem is quite properly stated but without a constructive analysis in particular.

Section 11 gives the overall outlook and expectations, talking not only about flights around the earth and moon and to the moon, but also about flights to Mars. Despite the very favorable weight conditions for flight to Mars (almost the same as for flight to the moon), I would classify any such considerations as premature because of the long duration of the flight and the tremendous weight of the supplies (air, water, food, fuel) implied by this in order to sustain the rocket passengers during the time of the flight (which could not be less than six months).

Otherwise, we cannot accuse the author of delving into outlandish fantasy.

In section 12, the experiments and investigations that must be carried out before rocket flight into outer space are indicated. Once again, the considerations herein are sufficiently well thought out.

The work of Comrade Kondratyuk can be published as it now stands. Later on, his work could be unified with the work of other authors on the same problem (K. E. Tsiolkovskiy, F. A. Tsander, myself, and, probably, still others) so as to publish a good collective work; but such a book cannot be written quickly, and in order to preserve the priority for the USSR, printing of the finished work should not be postponed in face of the possible writing of newer and better material.

With this in mind, it is vitally essential to procure copies written by the author himself, since the copy sent to me for review does not stand up to criticism in the sense of revision, nor have illustrations been furnished, although reference is made to same in the text. Errors in writing the formulas are admitted, in that the dimensions of acceleration are written in the form $cm \cdot S^2$ instead of $cm \cdot ^{-2}$, and α is written as L, etc.

Besides publication of the work of Comrade Kondratyuk, he himself should (in the event that he would agree to do so) be sent to work in Moscow, in closer proximity to the scientific centers; here his talents could be utilized many times more fruitfully than on a grain elevator, here Kondratyuk could continue his self-education and work productively in his chosen field. Such enormous native talents are exceedingly rare, and to leave them unattended, from the point of view of the State, would be a manifestation of wastefulness on the highest scale.

Moscow, 12 April 1926 V. Vetchinkin

On the advice of V. P. Vetchinkin, Kondratyuk slightly modified his system /665 of notation and terminology, adding to the book a previously uncited derivation of the fundamental formula of rocket flight and a whole fourth chapter, "The Combustion Process, Structure of the Combustion Chamber and Exit Tube," which had not been a part of the earlier manuscript.

In spite of the favorable review given by Vetchinkin, The Glavnauka (Main Administration of Scientific Instutions) not only denied Kondratyuk a grant of funds for publication of this book, but administrative assistance as well, so that he was compelled to publish the book at his own expense with a local Novosibirsk printer. In 1947, in other words after the death of Kondratyuk, the work was reprinted by Oborongiz (State Publishing House of the Defense Industry) under the editorial supervision of P. I. Ivanov, at which time a number of editorial changes were incorporated therein.

In the present volume, the work is published in the same form as it was printed in 1929 during the author's lifetime. Only the list of symbols given at the end of the book have been omitted. Some of the comments made by the editor of the first printing, V. P. Vetchinkin, are included as footnotes.

The comments made in the 1947 printing by the editor, P. I. Ivanov, are given below:

Note 1, page 62. Actually, if $W_i/u \ll 1$, then $n_i = e^{W_i/u}$ can be written as the first two terms of a series, i.e.,

$$n_i = e^{\frac{W_i}{u}} = 1 + \frac{W_i}{u}.$$

Then, replacing the value of n_i by the two-term expansion in the expression $\mu = M_f(n_i - 1)$, we have

$$\mu = M_к \frac{W_i}{u}.$$

Note 2, page 63. Assuming u is constant.

Note 3, page 64. The problem, as stated by the author, appears reasonable at first glance, In actuality, however, the mixing of solid or liquid substances with the gaseous products of the exhaust results in a reduction of the exhaust velocity due to drag losses. Insofar as the process of heat exchange takes time, one could hardly expect, in the short period of time that the exhaust products are in the nozzle, that they could even compensate for the lost velocity. Other than that, a molten metal moving with the gas glow will have a higher velocity and will therefore cause mechanical destruction of the nozzle.

Note 4, page 66. The author interprets W_1 as twice the parabolic velocity W relative to the earth's surface, assuming that the velocity on the earth's surface is equal to zero and the rocket trajectory has the earth as its focus. In this case, $W = \sqrt{2gR}$, where R is the radius of the earth, g is the gravitational acceleration.

Substituting the values of R and g, we obtain

$$W = 11{,}185 \text{ m/sec.}$$

The author interprets W_2 as the difference between W_1 and the circular velocity $W_c \approx 7910$ m/sec.

The coefficient in front of the number 11,185 is obtained from the following considerations. Since /666

$$W_1 = 2W, \text{ a } W_c = \sqrt{gR} = \frac{W}{\sqrt{2}},$$

it follows that

$$W_1 - W_c = 2W - \frac{W}{\sqrt{2}} = W\left(2 - \sqrt{\frac{1}{2}}\right) = \left(2 - \sqrt{\frac{1}{2}}\right) 11\,185 \text{ m/sec.}$$

Note 5, page 68. The author's idea is correct in principle, since the efficiency of an engine will in fact be increased when the pressure in the combustion chamber is elevated. However, considering account the weight of the combustion chamber at high pressures and the weight of the ancillary fuel injection equipments, it hardly makes sense to raise the pressure in the combustion chamber.

The author's reference to the hydrates of oxides is incorrect, as they cannot be formed in the combustion chamber. He overlooked berryllium, the most calorific metal.

Note 6, page 71. In fact, equation (6) can be written in the form

$$\mu = \frac{m}{\frac{1}{n-1} - q};$$

hence, if $q \ll \frac{1}{n-1}$, μ will be near $m(n - 1)$. If q is increased the difference $\frac{1}{n-1} - q$ will tend to zero and $\mu \to \infty$ under the condition that the same m_1 functions throughout the entire flight.

Note 7, page 71. Here the author is speaking of μ being doubled by comparison with the μ for $m_1 = 0$.

Note 8, page 71. The condition $q \ll \frac{1}{5(n_i - 1)}$ shows that in choosing q in accord with this condition we will have a value of u proportional to $\mu_0 = m(n - 1)$ in the following sequence:

$$\mu = \frac{5}{4}\mu_0;\ \mu = \frac{6}{5}\mu_0;\ \mu = \frac{7}{6}\mu_0;\ \mu = \frac{K+1}{K}\mu_0$$

and the higher the value of K, the nearer μ will be to μ_0.

Note 9, page 71. Since $n_i = M_{i_o}/M_{i_f}$, while M_{i_f} includes μ for the (i + 1)th segment, it is not meaningless to speak of $n_i = 1$.

Note 10, page 72. In order to obtain the figures indicated by Kondratyuk, q = 1/9 for a two-unit system and 1/3.9 for a three-unit system, it is necessary to recall that the author assigns a unit to each segment, while each segment has the same W_i, so that for a single-unit system we have W, for the two-unit system $\frac{1}{2}W = W_i$, and for the three-unit system $W_i = \frac{1}{3}W$. Since $n = e^{W/u}$, it follows that $n_i = e^{\frac{1}{2}W/u}$ for the two-unit and $n_i = e^{1/3\ W/u}$ for the three-unit system.

Consequently, we can represent n_i for the multiunit system in terms of n /667 one-unit systems as follows:

$$n_i = \sqrt[n]{e^{\frac{W}{u}}} = \sqrt[n]{n}.$$

For the three-unit system, the author gives a value of 1/3.9, but it should be 1/3.65.

Note 11, page 72. The fuel supply for $m_1 = 0$, according to equation (6) is

$$\mu = \frac{m(n-1)}{1-q(n-1)}.$$

The total weight, therefore, will be u + m, but since for $m_1 + 0$ we have $u = m(n-1)$, it follows that $u + m = mn$. Consequently, comparing the weight of the rocket for $m_1 \neq 0$ with $m_1 = 0$, we have

$$\mu + m + m_1 = \frac{m(n-1)}{1-q(n-1)} + m + m_1;$$

but

$$m_1 = q\mu,\ \&\ \mu = \frac{m(n-1)}{1-q(n-1)},$$

so that we have

$$\frac{m(n-1)}{1-q(n-1)}+m+\frac{m(n-1)}{1-q(n-1)}$$

and, after some manipulation, we obtain

$$\frac{mn}{1-q(n-1)} \text{ or } mn\frac{1}{1-q(n-1)};$$

while, on the other hand, for a multiunit system $n = n_i^K$, and we can write therefore

$$mn_i\left[\frac{1}{1-q(n-1)}\right]^K.$$

Note 12, page 74. The equation cited by the author, $w_e = v(\sqrt{2} - 1)$, can be derived under the condition that the parabolic velocity coincide with the direction of the circular velocity. v must be interpreted as $w_c = \sqrt{gR}$. Under these conditions, w_e will be equal to the velocity given by the author. In the event that the parabolic velocity does not coincide with the direction of the circular velocity, then

$$w_e = w_c\sqrt{3-2\sqrt{2}\cos\gamma},$$

where γ is the angle between the directions of the parabolic and circular velocities. The return velocity

$$w_r = \sqrt{w^2 - v^2}$$

transforms to $w_r = w\sqrt{1 - 1/2\bar{r}}$, if we let $v = \sqrt{gR}$ and $w = \sqrt{2gR}$.

Note 13, page 80. Equation (12) is obtained from equation (9) as follows: /668
Letting $j = j_0 + j_\rho$ and $g = g_0$, we write

$$\frac{dv}{dt} = j - g \text{ or } \frac{vdv}{dr} = j - g.$$

Since $v = dr/dt$, we have $v^2/2 = jdr - gdr$. Integrating with the initial conditions $v = 0$ and $r = R$, we have

$$\frac{v^2}{2} = jr - jR + gR - gr,$$

but, since

$$w^2 = 2gR,$$

replacing v by w, we obtain

$$jr - jR - gr = 0;$$

hence

$$r = R\frac{j}{j-g}.$$

But, since

$$W_i = \sqrt{2gr},$$

replacing r by $R\frac{j}{j-g}$ here, we have

$$W_i = \sqrt{2gR}\sqrt{\frac{j}{j-g}} = w\sqrt{\frac{j}{j-g}};$$

We now turn to equation (9) and determine L_g. Inasmuch as j_0 is imparted to the rocket in the immediate vicinity of earth, $g = g_0$, v_2 is the velocity of the rocket at infinity, and, consequently, $v_2 = 0$; v_1 is the velocity on the earth and is also equal to zero, $\bar{r}_2 = \infty$, $\bar{r}_1 = 1$, since $r_1 = R$ by virtue of the fact that velocity is imparted near the earth. From these assumptions, therefore, we have

$$L_g = W_i - w. \text{ But } W_i = w\sqrt{\frac{j}{j-g}},$$

hence

$$W_i - w = w\left(\sqrt{\frac{j}{j-g}} - 1\right),$$

or, introducing the notation $j/g = \bar{j}$, we have

$$L_g = \left(\sqrt{\frac{j}{j-1}} - 1\right)w,$$

i.e., we have derived equation (12).

Series expansion of the expression under the radical with the condition $\bar{j} \gg 1$ leads to an expression of the following form: $L_g = \frac{1}{2\bar{j} - 1}$, rather than the relation cited by Kondratyuk, i.e., /669

$$L_g = w \frac{1}{2(\bar{j} - 1)}.$$

Note 14, page 85. Recent investigations have shown that a human being, lying on his back, can tolerate accelerations considerably higher than stated by the author.

Note 15, page 112. The exhaust velocity u = 4700 m/sec is much too high in contrast with what is actually feasible.

Note 16, page 113. Here, n_E, n_m, n_M are the values of n for earth, the moon, and Mars, respectively.