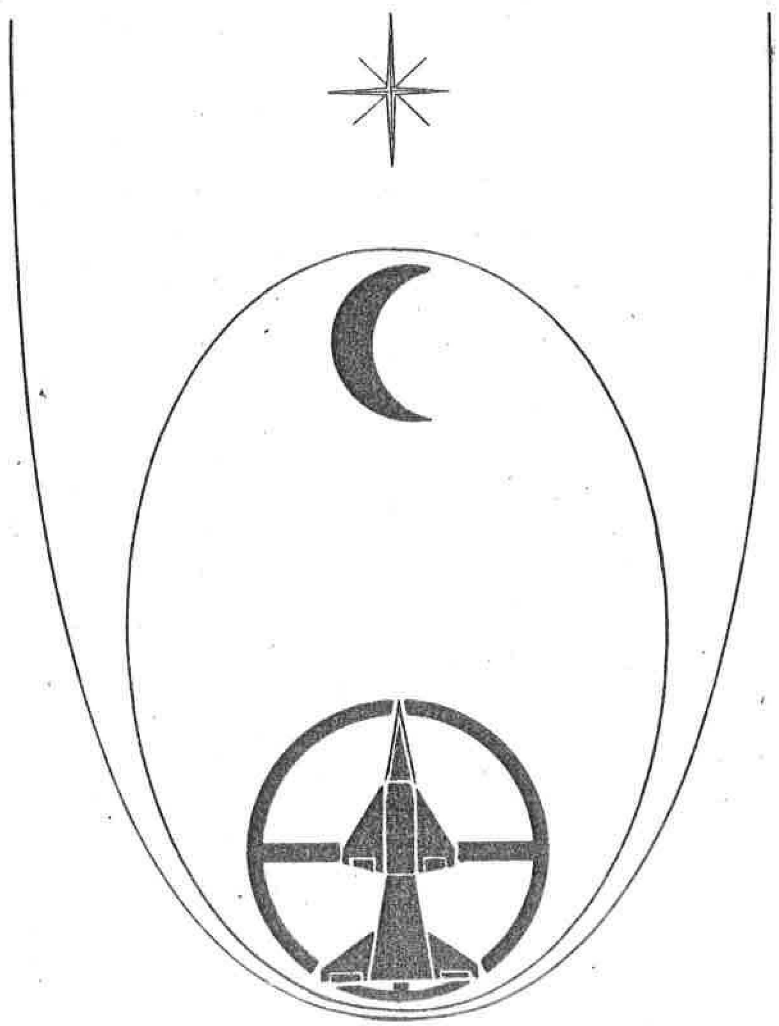
ANALYSIS OF ORBITAL SYSTEMS

by

Krafft A. Ehricke



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(With 21 Figures)

Abstract. The paper presents an analysis of orbital systems, consisting of the orbital establishment, its supply vehicles and their technique of operation. Satellite orbits are classified as permanent (stationary for more than 10 years) and as temporary. The lower altitude limit for permanent satellite orbits appears to be 450 to 500 miles. These orbits are occupied by observational satellites. An altitude range between 500 and 700 miles is found to be relatively most desirable for observational satellites. Three types of temporary orbits are defined, namely auxiliary orbits (120 to 150 miles altitude; duration of occupation a few hours at the most; purpose is payload transfer), orbits of departure of astronautical expeditions, particularly into the translunar space (350 to 400 miles altitude; duration of occupation about one year or less; purpose is assembly of inter-orbital vehicles), and orbits of arrival of astronautical expeditions (30,000 to 40,000 miles altitude for Venus or Mars expeditions; duration of occupation for a few days by returning expedition until being picked up by orbital vehicle from the earth). It is shown that the optimum satellite orbit of departure is as close to the earth as feasible and that the optimum orbit of return as well as the optimum satellite orbit at the target planet vary with the target planet. Therefore, with the exception of the orbit of arrival, all satellite orbits preferably lie below 700 miles altitude, that is, within the physical atmosphere of the earth.

Orbital supply systems are discussed, distinguishing between passenger-carrying and load-carrying orbital vehicles. The latter type is fully automatic. A large supply vehicle will be needed for periods of establishing orbital installations. For their maintenance a small version is anticipated. Desirable features and configurations for observational satellites and for orbital vehicles are discussed.

Nomenclature

C_L	lift coefficient
F	thrust
F_c	apparent centrifugal force
i	orbital inclination with respect to the ecliptic
P	precession constant
R	distance from the sun
r	distance from the earth
$r_{s,opt}$	distance of optimum satellite orbit of departure with respect to a given interplanetary transfer ellipse
S	lifting area
T_{pr}	period of regression of the orbital nodes due to polar-precession
T_{pr}'	period of regression with respect to the half of the globe being exposed to the sun
t_r^*	sidereal period of revolution in an orbit
v	velocity
W	vehicle weight
y	altitude
α	angle of attack
$\gamma_\oplus$	or γ parameter of the terrestrial gravity field $(6.254 \cdot 10^4 \, n \cdot \mathrm{mi^3/sec^2} = 3.98 \cdot 10^5 \, \mathrm{km^3/sec^2})$
$\gamma_\odot$	parameter of the solar gravity field $(2.067 \cdot 10^{10} \, n \cdot \mathrm{mi^3/sec^2} = 1.36 \cdot 10^{11} \, \mathrm{km^3/sec^2})$

[1] Bell Aircraft Corporation, Buffalo, N.Y., U.S.A. Preliminary Design Department.

δ	orbital inclination with respect to the equator	$\dot{\psi}_p$	rate of regression
δ	displacement thickness of the boundary layer	ω	angular velocity
			Subscripts:
θ	trajectory angle	A	apogee or aphelion
λ	mean free molecular path	c	circular
ϱ	air density	P	perigee or perihelion
σ	ϱ_y/ϱ_{00}	s	satellite
$\dot{\varphi}$	angular velocity of satellite in orbit	y	referring to altitude y
		00	surface value

I. Foreword

While the present paper was under way, the author learned about the paper "Fabrication of the Orbital Vehicle" by Messrs. GATLAND, KUNESCH and DIXON [11] which, in part covers the same subject as presented independently in [4] and in the present paper, namely the use of automatic supply ships. The British study concentrates on the fabrication of a space ship while [4] and this study stress, in addition, the use of automatic supply ships for the establishment and maintenance of satellites. However, the important factor is the principal agreement as to the general philosophy regarding automatic supply ships. This, the author feels, should be valued as a welcome and encouraging sign that the concepts regarding the avenues of approach toward space flight begin to converge in the light of careful analyses and strictly functional vehicle layout.

II. Introduction

Evaluation of the present trends in the technology of rocket propulsion indicates a growing emphasis not only on one of the main development lines — the guided missile — but likewise on the manned, rocket-powered plane which represents the other main development line. While details of the work in all leading countries are classified, the known activities and problems[1] in the guided missile field, as well as published data on recent successful flights of Bell Aircraft and Douglas Aircraft piloted rocket planes bear witness to this fact.

Such development trends, although presently based on military considerations, could not, for all practical purposes, agree more properly with what might be called for at the present time in a planned development program leading toward space flight. In a sense, therefore, a coordinated space flight program is already in existence. Although, figuratively speaking, the final destination of the space flight enthusiast is different from the terminal of the military planner, as far as rockets are concerned, both stations nevertheless do lie on the same route of development progress. Obviously, the military terminal is so close to the goal of the space flight enthusiast, at least in regard to orbital systems, that from thereon he may very well be capable of being on his own for the rest of the way.

Fig. 1 is a graphical presentation of such a planned space flight program. Originally published several years ago [1][2] it applies unchanged. In fact, a quite similar line of thought has been presented recently [2]. The program anticipates 4 phases:

[1] An example for the order of magnitude of the problems facing the guided missile development in leading countries is presented in the Air Force study entitled "Almost is Not Enough", Air Force **36**, No. 3 (1953).

[2] Numbers in brackets refer to References on pages 57 and 58.

Phase *A*. Development of a manned orbital glider (satellite rocket plane) with very limited stay time in the outer atmosphere; perhaps only as few as 5 to 10 revolutions.

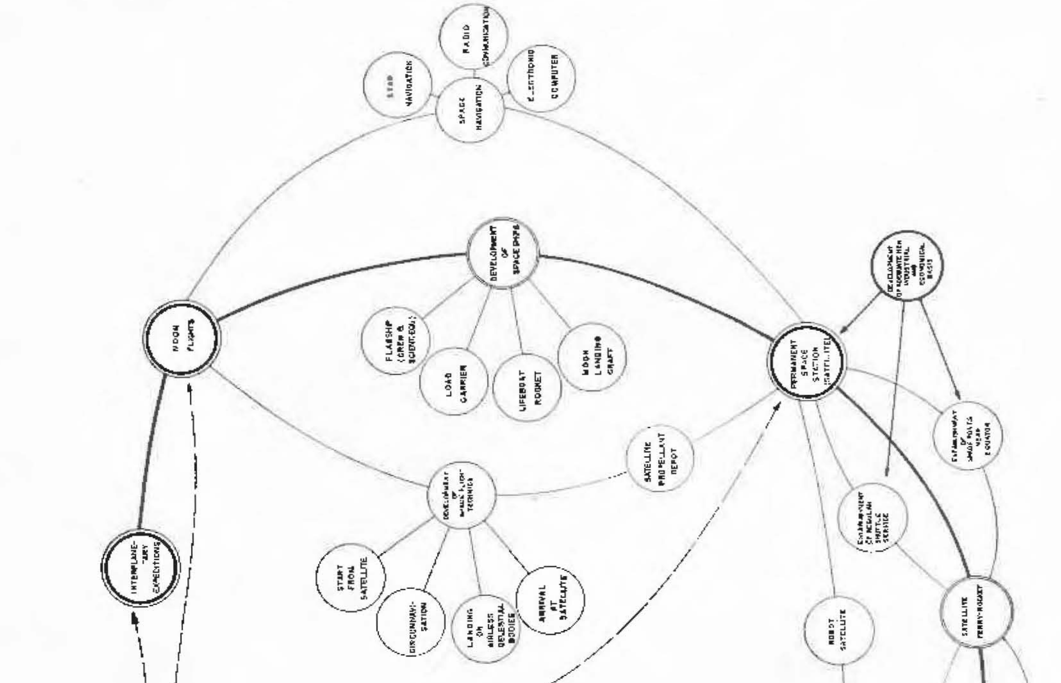

Fig. 1. Phases of development leading toward interplanetary flight.

Phase *B*. Development of a permanent space station.

Phase *C*. Development of space ships for astronautical expeditions (this is meant to include the complete technical and biological system required for such an endeavor).

Phase *D*. Interplanetary flight.

This whole program up to and including the first interplanetary expedition can take about as little time as you wish to assume (within technical reason), provided you succeed in making it a "hang-the-expenses" program, similar to the Manhattan Project. The likelihood that you succeed is extremely small, at least beyond phase *A* and certainly beyond phase *B*, becanse thereafter a military incentive can no longer be expected with a reasonable degree of probability. Of course, it is impossible now to accurately predict where the military "terminal" will be, but chances are that one can look for it somewhere in phase *B*; the limit presumably is reached with a low altitude temporary, and eventnally permanent observation station (manned reconnaissance satellite) which, as far as it is effective, represents the non plus ultra of military reconnaissance.

Military interest is a necessary, but not sufficient condition for a "hang-the-expenses" program. To get this under way seriously, it will take a state of acute emergency. Excluding this possibility, phase *A* could, within the next 10 to 15 years, be pushed up to the point where a multi-stage satellite rocket plane is available, and the art of flying between 26,000 and zero feet per second has adequately been mastered. At the same time, the use of automatic (instrument carrying) robot satellites in temporary orbits can be anticipated. Aside from reconnaissance, these instrumental satellites can be useful as test carriers and closely connected with the development of long range rockets of the ballistic type. These rockets are called guided space missiles in Fig. 1, since, for greater ranges, they reach far beyond the realm of temporary satellites into the regiou where the permanent satellite will be located. Fig. 2 shows the various altitude regions pertaining to air-breathing flight, rocket powered intercontinental flight, temporary, and permanent satellites. In the latter case, satellites whose orbit is stationary for a decade or more, have, somewhat arbitrarily, been classified as permanent. With reference to "space missiles" (intercontinental ballistic rockets), Fig. 3 presents a plot of summit altitude and range versus cut-off speed; where the summit altitude is given as ratio of snmmit point distance to surface distance from the center of the earth (snrface distance $r_{00} = 3{,}436$ n. mi. or 6,378 km), the range is expressed as half center angle (radians) of the trajectory, and the cut-off speed is presented in terms of the circular velocity at the cut-off point. This plot is based on a family of ellipses which yield the greatest range for a

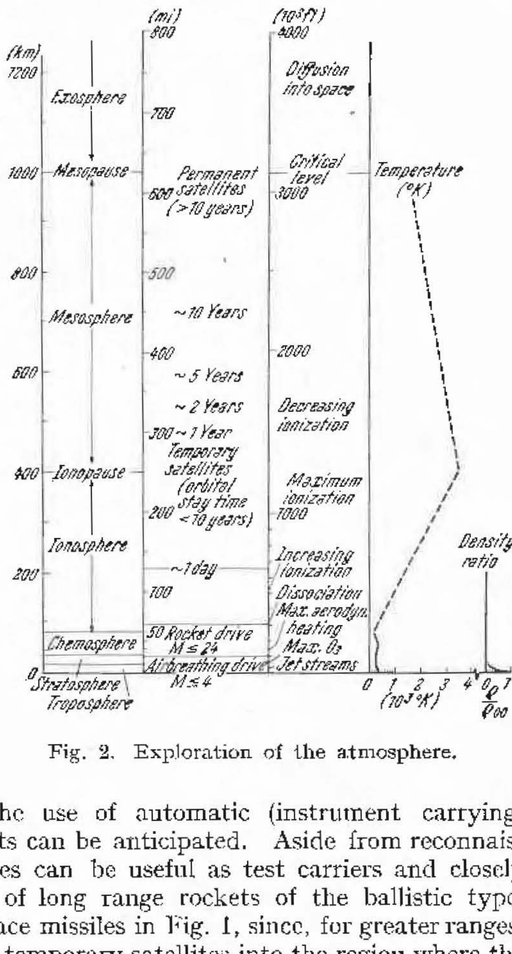

Fig. 2. Exploration of the atmosphere.

given cut-off velocity. Finally, the development of an instrumental satellite will be interrelated with the development of very-long-range or orbital rocket planes, inasmuch as it ties in with atmospheric research and investigation of hypersonic flight phenomena. Phase A thus can be termed, popularly "Conquest of the Atmosphere". As such it will, in its entirety, find military interest. On these premises it can be expected that any military development program will, in all major points, coincide with a coordinated space program during phase A.

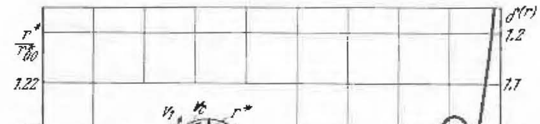

Fig. 3. Summit distance and range for optimum elliptic paths. v_c = circular velocity at cut-off point.

The same holds true for phase B only as far as the value of a satellite in an overall defense system is concerned.

Beyond the reconnaissance station, then, the conditions for astronautical progress will change radically, because the main customer for the products of this development will drop out. This implies the possibility that on the very verge of true astronautics may lie its most critical period. However, it is fully recognized that, since this period certainly lies 20 to 30 years ahead, new conditions may arise which either offer new incentives or make a more liberal devotion of civilized mankind to its truly noble aspirations possible.

The effectiveness and usefulness of any inhabited satellite, and the chances for realization of astronautical expeditions decisively depend on the development of a highly functional supply system as well as on a realistic attitude toward orbital establishments. In the subsequent discussions we will, for reasons of brevity, designate both, the supply system from earth to orbit, as well as the orbital establishment, as orbital system. Briefly, then, the most challenging task of phase B will be the development of orbital systems. Some of them may still lie within the realm of military interest, others just beyond that limit.

The work will logically be a continuation of the preceding phase inasmuch as then previous development efforts are utilized to the greatest possible extent. This point has been stressed before [3], particularly with reference to the fact that a continuation of the astronautical development program beyond military requirements will be immediately dependent upon the existence of an economic supply system which operates efficiently with what is available, rather than requiring "The Great Iron Ship" of space for proceeding.

At the end of phase A we can expect, at the most, to find the following status of rocket technology:

1. Three-stage vehicles of 600,000 to 1 million pounds take-off weight — perhaps even as high as 1.5 million pounds — will either be available, or within reach of short-term development efforts, because more is not needed for any terrestrial vehicle up to and including instrumental satellites and temporary orbital reconnaissance planes. In fact, most of the temporary instrumental satellites will weigh much less. At ideal velocities of the order of 31,000 ft/sec, as required for orbital supply ships, such initial weight will allow for a payload capacity of 9,000 to 12,000 lb in automatically controlled vehicles, and of 1,000 to 1,500 lb in manned vehicles with winged upper stage.

2. Man will have gained his first experience regarding short-time existence in space, sheltered in a cockpit. Conditions will be more severe in phase B when he has to stay in space for extended periods of time and, during this time, has to work outside, protected only by a space suit. As a result of the first trials during phase A it will be possible to determine more accurately the maximum safe stay time of a crew in space, at least during the initial phase of establishing and operating a satellite, because during this time outside activity will be particularly frequent. This probably will result in the requirement for a certain cycle of rotation of personnel; a very expensive proposition, because it cannot be done without winged upper stages which have a particularly low transport efficiency [4]. For this reason it will be found necessary to restrict the number of personnel in space to the bare minimum.

These apparently plausible premises pretty much determine the fundamental design criteria for orbital systems, namely,

(a) orbital establishments must be small, because of the limited payload capacity of available vehicles, or their souped-up versions, and because, for operational and logistic reasons, the establishment of a given unit cannot take any arbitrary length of time and, for that matter, number of supply flights,

(b) the number of personnel, permanently accomodated by the satellite must be very limited. It is, perhaps, not an unreasonable guess to assume 4 persons. This is about as much as can be carried — aside from the pilot — in one passenger vehicle within the range of available take-off weights (point 1 above). Thus not more than one passenger ship would be involved in a normal personnel rotation.

(c) From (a) and (b) it follows quite naturally that, for maintenance of an established satellite, only comparatively small quantities of supply are needed, probably not more than 2,000 lb per month [4]. Making this weight the standard payload for a small maintenance ship will provide a reasonable safety margin inasmuch as several ships can be sent up in rapid sequence, should need arise.

Before continuing, it may be pointed out here that the above reasoning applies to thermo-chemical rockets, because they are believed to be the only type of rocket propulsion available for phase A and for a large portion, if not all, of phase B. As far as we can see today, the technical application of nuclear energy to rocket propulsion will be restricted, for some time to come, to the transfer of heat to working fluids. Even under extremely favorable heat transfer conditions of high pressure and high temperature, thermo-nuclear power plants appear by no means superior — in many cases they are even inferior — to thermo-chemical systems [5], except under space flight conditions. This also is, in essence, the conclusion which must be drawn from the excellent paper [6] which points out means of greatly increasing the specific impulse of working fluids by radically lowering the chamber pressure (although this makes the heat transfer — and pile cooling — process very difficult). It can, therefore, be expected that the development of thermo-nuclear propulsion systems for rockets will not be sufficiently attractive until true space flight has become imminent, that is, in phase C and D (Fig. 1). At present, and in the decades to come, the atomic development program will spend much of its efforts to carrying this energy to the industrial and household consumer. The result of this effort doubtlessly will help building the "atomic space ship"; but, by then, I hope, will we have proceeded already well into phase *B*. Thus it is unlikely that we will encounter thermonuclear propulsion in phase *A* or *B*; on the other hand, it may not be altogether unreasonable to assume that phase *C* and *D* rapidly will do away with thermo-chemical propulsion for inter-orbital space ships. This, in fact, is part of the space ship development program which constitutes phase *C*.

On the basis of the general design criteria outlined above, it follows that the supply system must be different for "building periods" and for "maintenance periods". Inhabited satellites generally pass through both of these periods. Establishments in orbits of departure of astronautical expeditions are of temporary nature and, therefore, experience only the first period.

Based on these premises, a new satellite supply system, involving automatic supply ships for material transport, and restricting the use of winged upper stages to passenger service only, has been analyzed in [4]. The relative weight and complexity of a purely automatic supply ship has been compared to a supply ship having a crew or passenger carrying capability. The result was overwhelmingly in favor of the wingless automatic supply ship which, for a take-off weight of 13. to 1.4 million pounds (11,000 lb payload) was found to yield a 9 times greater payload capacity per mission (if not even more) than a ship of equal size with winged upper stage. It has also been shown that an automatic supply ship system with an 11,000 lb "heavy-duty" ship for the "building period" is reasonably adequate for the establishment of small observational (reconnaissance) satellites, as well as for the establishment of orbits of assembly and departure for atronautical missions up to and including our two neighboring planets.

The present study is an extension of the analysis presented in [4], covering complete orbital systems.

III. Requirements

The primary objective of an orbital supply system is to maintain connection between the earth and any orbital installation in the terrestrial gravity field, for the purpose of transporting personnel and equipment into and out of the respective orbit.

In order to fulfill the requirements connected with these objectives, it is necessary to

1. study the probable range of distances and inclinations (with respect to the equator plane) of future orbits,

2. analyze the probable material and transportation demands for the various types or orbital establishments.

Orbital establishments, involving human activity in space are

(a) observational satellites (permanent)
(b) orbits of departure of astronautical expeditions (temporary)
(c) orbits of arrival of astronautical expeditions (temporary)
(d) auxiliary orbits (temporary).

1. Observational Satellites

As observational satellites we define orbital installations which are concerned exclusively with terrestrial functions, pertaining to the earth as the abode of man, and to scientific activity.

Such satellites, being inhabited, will stay up permanently. Orbital inclinations of between 45 and 75 degrees are desirable, in order to pass, at one time or another, through the zenith of as many points on the surface as possible. Orbital altitudes should be as low as possible, in order to keep maintenance costs down and permit best utilization of earth-scanning devices (such as telescopes, camera, radar, other electromagnetic equipment). Low altitude is especially important for obtaining reasonable resolution with non-optical devices (e. g. radar; see, for instance [7]).

Fig. 2 indicates an altitude of 450 miles or more for permanent satellites. However, 400 miles, or perhaps even 350 miles, appear feasible if provisions are made for correction forces, applied to the satellite by means of an attached propulsion system, or by pushing or towing (to the limited extent necessary) with a rocket ship, provided the satellite is a small, single-body affair. In any case, there is no reason why the orbit of permanent satellites should be located beyond 600 to 700 miles altitude.

On the other hand, two factors may be mentioned which caution against placing orbits, particularly permanent orbits, at too low an altitude:

1. Orbital perturbation.
2. Accuracy of flight, particularly of automatic supply ships.

The second factor will be subject of a separate discussion which, in its details, is beyond the scope of the present paper. Some remarks pertaining to the first factor are presented subsequently.

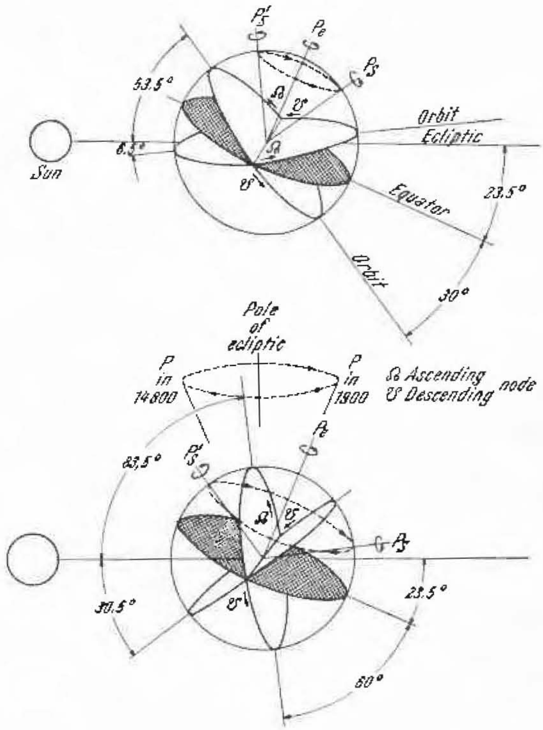

Fig. 4. Regression of the nodes due to polar-precession.

Any orbit around the earth is perturbed by the sun, moon, and by the oblateness of the earth. Its polar radius is about 12 miles smaller than its equatorial radius. As SPITZER has shown [8], these perturbations affect a satellite orbit in two ways:

(a) they cause periodic oscillations in altitude (tidal effect),

(b) they produce a precession of the orbital plane.

The tidal effect causes the orbit to deviate from a perfect circle (if it ever was one, for reasons of accuracy of the original establishment). SPITZER has shown that the effect of sun and moon is negligibly small. The sun causes a deviation of less than one foot in altitude, and the effect of the moon is about twice as strong. Perturbations due to the oblateness of the earth can be considerably larger. They are zero when the orbital plane coincides with the equator plane. With increasing inclination the oblateness becomes progressively more effective, reaching a maximum for $\delta = 90^0$ (polar orbit) where δ designates the angle of the orbital plane with respect to the equator plane of the earth. In a polar orbit the amplitude of the oscillation in altitude is found to be about 1 mile, if the orbital altitude is about 60 miles. At 625 miles altitude, the deviation still is about 0.94 miles, gradually decreasing to 0.75 miles at about 1875 miles altitude. The amplitude, therefore, changes very little within the range of relevaut orbital altitudes. Since the above values apply to a polar orbit, any orbit with less inclination will show a lesser altitude oscillation.

The tidal effects are therefore small in any case, except in regions of atmospheric density where even this altitude variation augments materially the effect of air forces in causing a decay of the orbit into a spiral of descent. This, however, is the region, roughly below 100 miles, the empty zone in Fig. 2, above the region of hypersonic rocket flight.

The second effect mentioned above is orbital precession owing to a nonspherical distribution of the gravitational potential in the surrounding space. The inhomogeneity is such that it produces a gravitational pull, directed normal to and toward the equatorial plane, on every body which moves outside of this plane. The result of this normal force is of considerable practical significance for observational activity. It causes a regression of the nodes, that is, a retrograde or "westerly" motion of the two points of intersection of satellite orbit and equator (nodes). In other words, the north pole of the satellite orbit is not at rest, but revolves about the north pole of the equatorial plane. These conditions are explained in Fig. 4 [3] for the cases of an orbital inclination of 30° and of 60°. In analogy to the lunisolar-precession of the equator with respect to the ecliptic plane owing to the perturbing effects of moon and sun, one may designate the cause for the nodal regression of the satellite orbit as polar-precession, because it is caused, for all practical purposes, by the polar oblateness of the earth. SPITZER [8] has shown that again the effect of sun, moon, and the stars is negligible for any practical length of time. Under the influence of polar oblateness the pole of the satellite orbit (Fig. 4) revolves about the pole P_e of the equatorial plane from P_s to P_s' and back to P_s. When the pole, coming from P_s arrives in P_s', the ascending and descending node have changed their places. The period of regression is the time required for P_s to complete one revolution about P_e along the circle indicated by the dashed line. Actually, as mentioned before, the lunisolar-precession causes P_e to revolve about the pole of the ecliptic plane, as indicated in the lower part of Fig. 4; but the motion is so slow that its influence on the orbital position with respect to the ecliptic plane can be neglected for all practical purposes.

The orbital period of regression due to polar-precession is given by

$$T_{pr} = \frac{360}{\dot{\psi}_p} \text{ (sec)} = \frac{0.00417}{\dot{\psi}_p} \text{ (d)} \tag{1}$$

where the rate of regression $\dot{\psi}_p$, in degree per second, depends, in analogy to the lunisolar-precession of the equator, on a precession constant P, multiplied by the cosine of the angle of inclination δ,

$$\dot{\psi}_p = P \cos \delta. \tag{2}$$

The precession constant, derived in [8] and [9], can be expressed in the first approximation, using present notation, by

$$P = \chi \frac{360}{t_r^*} \tag{3}$$

where

$$t_r^* = \frac{2\pi r_s}{v_c} = \frac{360}{\dot{\varphi}_s} \tag{4}$$

is the period of revolution, measured against a fixed point in space (r_s and $\dot{\varphi}_s$ represent the distance and angular velocity of the satellite) and

$$\chi = 1.64 \cdot 10^{-3} \left(\frac{r_{00}}{r}\right)^2 . \tag{5}$$

r_{00} and r being the radius of the earth and the mean distance between orbit and the center of the earth, respectively.

The period of regression, therefore, depends on orbital distance and inclination. It increases with both parameters. The preceding equations show that T_{pr} becomes infinite in the case of an polar orbit. At very small inclinations the period of regression reaches its smallest value and depends on the distance only.

The period of regression defined in Eq. (1) refers to a fixed coordinate system in space rather than to a point on the earth's surface. Of greater importance for observational purposes is the effect of regression on the visibility of certain parts of the surface. Only those regions are accessible to optical observation which lie in daylight. Due to the revolution of the earth about the sun, the distribution of light and darkness on the globe varies during the year. If the sun would be infinitely far away so that the motion of the earth could be regarded as being rectilinear for any practical length of time, then a given point on the surface could be observed in daylight at alternate time intervals equal to T_{pr}. However, since the earth moves along a circle, reaching the original position in regard to the sun after 365 days, and since the regression is directed retrograde, that is, opposite to the motion of the shadow of the earth, the period of regression in regard to the daylight side must be shorter. It can be expressed in the form

$$\frac{1}{T_{pr}'} = \frac{1}{T_{pr}} + \frac{1}{365} . \tag{6}$$

A given point on the earth can thus be observed in daylight at alternate intervals of T_{pr}', independent of the orbital distance as far as the additive term 1/365 is concerned.

Fig. 5 shows the period of regression T_{pr} and T_{pr}' as function of orbital altitude and inclination. The period increases with both parameters, but it grows more rapidly with the inclination than with the distance. Because of Eq. (6) the period of regression should not be too long. Hence, for large orbital inclination, low orbital altitude again is desirable. Inasmuch as there is a lower altitude limit, several observational satellites may become necessary.

Summarizing, then, it can be stated that all known circumstances point toward *high* orbital inclination, but *low* orbital altitude as the most desirable position of observational satellites. A maximum of about 700 miles orbital altitude, therefore, appears acceptable.

2. Orbits of Departure

Orbits of departure are occupied by temporary satellites with astronautical functions. In these orbits are the astronautical (inter-orbital) vehicles assembled, equipped, and fueled. From these orbits they depart to their destination in the cislunar or translunar space.

It has originally been explained in [3] that, due to the polar-precession of inclined orbits, observational satellites can not be used as bases for astronautical endeavors, at least not into interplanetary space. For that matter, no orbit of departure can be used for the return. The reason for this is that orbits of departure whose plane is inclined with respect to the plane of the transfer ellipse which leads to the target planet, require additional energy, since the velocity vector not only has to be increased from circular to hyperbolic, but also its direction must be changed. If v_1 is the velocity in the old orbit and $v_2 \gtreqless v_1$ the velocity in the new orbit, and if i is the angle between the two orbital planes, then the total velocity increment, required for the scalar and the directional change is given by

$$\Delta v = \sqrt{v_1^2 + v_2^2 - 2\, v_1 v_2 \cos i}. \tag{7}$$

For the directional change alone $(v_1 = v_2 = v_c)$ the energy penalty amounts to

$$\frac{\Delta v}{v_c} = \sqrt{2\,(1 - \cos i)} = 2 \sin \frac{i}{2}\,. \tag{8}$$

This relation is plotted in Fig. 6 and shows the fallacy of the assumption that an observational satellite in an inclined orbit could be used as base for interplanetary flights. This does not apply to the moon or to any operation in the cislunar space, because in this case the transfer paths to the apogee are ellipses or nearly parabolas with the earth remaining in one focus.

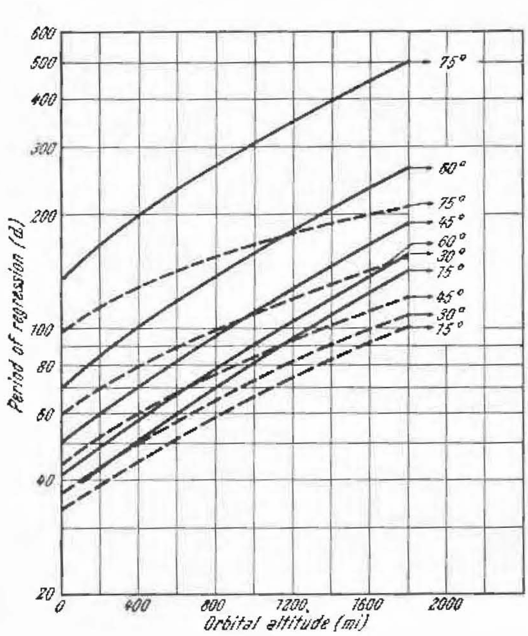

Fig. 5. Period of regression of orbits. Absolute ———, daylight side - - - - -

An astronautical expedition, therefore, will return into an orbit which is different from the orbit of departure. As far as lunar circumnavigations and cislunar operations (training and development flights) are concerned, this is the only thing to observe.

In regard to interplanetary expeditions the conditions are different. It has been shown in [4] that there exist certain optimum satellite orbits of departure or arrival for which the transfer energy to another distance from the sun becomes a minimum. This optimum orbit is defined by the relation

$$r_{s,opt} = \frac{2\,\gamma_{\oplus}}{\dfrac{\gamma_{\ominus}}{R_P}\left[\sqrt{\dfrac{2\,R_A}{R_P + R_A}} - 1\right]^2} \tag{9}$$

where $\gamma_{\oplus}$ and $\gamma_{\ominus}$ are parameters of the terrestrial and solar gravity field, respectively (cf. nomenclature) and R_P and R_A are the perihelion and aphelion distance of the interplanetary transfer ellipse (normally assumed to contact the orbits of the planet of departure and of the target planet, respectively). The optimum satellite orbit distance is shown in Fig. 7, plotted against the distance of one of the apsides, while the other, of course, is assumed to lie in the earth's orbit. It will be noted that just for Venus and Mars the optimum satellite orbits are very far out.

These orbits are optimum inasmuch as they yield the smallest transfer energy from the satellite orbit to the *planetary* orbit of the target planet. For the optimum satellite orbit near the target planet Eq. (9) must be applied to the gravity field of the respective planet; that is, $\gamma_{target\,planet}$ must be substituted for $\gamma_{\ominus}$. The transfer energy, in velocity equivalents, is plotted in Fig. 8 for all planets in the solar system, as function of the distance of the satellite orbit of departure (or

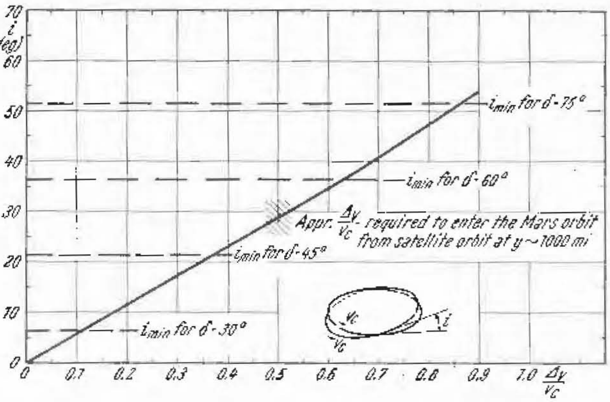

Fig. 6. Velocity increment required for change of orbital plane. δ = orbital inclination with respect to the equator, i = orbital inclination with respect to the ecliptic.

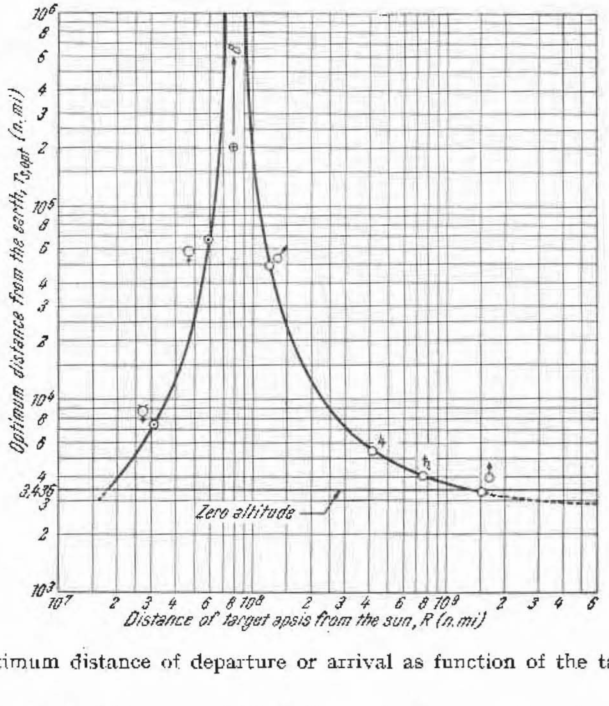

Fig. 7. Optimum distance of departure or arrival as function of the target planet.

arrival). In the case of Venus and Mars the savings amount to about 4,000 ft/sec. However, it has also been shown in [4] that the optimum orbit of *departure from the earth* lies as close to the surface as feasible, if the orbital supply system is taken into account. An example, illustrating this fact is presented in Fig. 9, showing the conditions for an orbit of departure for a Mars expedition [4]. The figure shows various velocities involved as function of the distance from the earth, where r_{00} is the earth's radius, r_I is the distance at which the ascending supply ship has attained local circular velocity, v_c is the circular velocity, Δv_{II} is the velocity increment which throws the vehicle into the transfer ellipse to the satellite orbit, and Δv_{III} is the short burst of power required at the apogee point to enter the satellite orbit; finally, v_A is the velocity at the apogee point of the transfer ellipse and ΔV_{st} is the velocity increment required for the Mars ships, leaving the orbit, to attain hyperbolic velocity with respect to the earth and to enter an interplanetary transfer ellipse whose aphelion lies in the Mars orbit. The circular velocity at r_I must be attained independently of the distance of the orbit of departure. Therefore what counts for a minimization of the overall effort — supply and departure — is the sun $\Delta v_{II} + \Delta v_{III} + \Delta V_{st}$. It can be seen that the curve representing this sum of velocities increases with the distance from the earth.

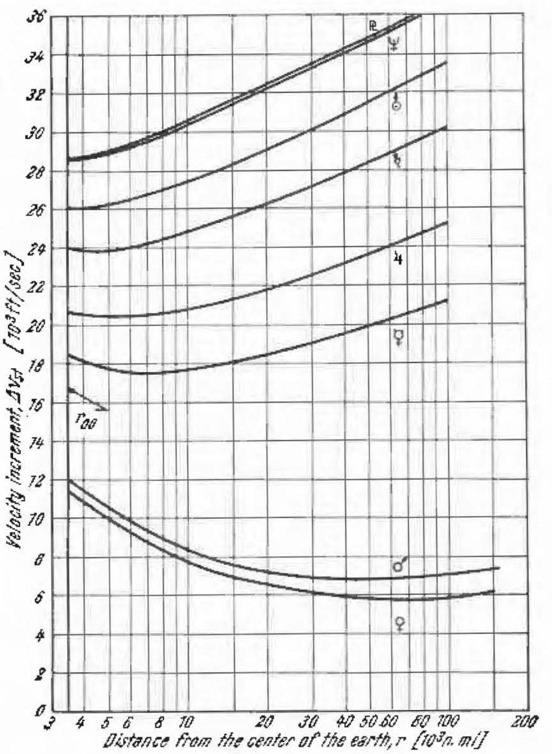

Fig. 8. Velocity of departure vs. distance from earth.

As a matter of economy, the altitude of the orbit of departure should therefore be as low as feasible. This holds true also for all lunar and cislunar operations.

The orbital inclination has no significance for lunar and cislunar operations with one exception which will be mentioned subsequently. For circumnavigations the moon orbit can be intersected from any direction and there is no principal preference as to the inclination of the satellite orbit around the moon. The exception is that, for observation of certain areas and particularly if, at a later stage, landings on a predetermined landing site, are planned, then the resulting requirements of course determine the inclination of the satellite orbit of departure. Under these conditions it would be more or less accidental if an observational satellite could be used as base for departure or could be aimed at during a return manoeuvre.

For interplanetary expeditions the orbital inclination is determined by the plane of the transfer ellipse to the target planet. Thus the inclination of the orbit of departure at the beginning of the assembly operations (the initial inclination) follows from the period of regression T_{pr} and from the time required to get the expedition under way. Since the time of departure is fixed uniquely by the correct constellation of earth and target planet for a given transfer ellipse, it is the time of beginning of the assembly that must be selected and which, for the given altitude, yields the desired initial inclination.

Summarizing then, it can be stated that orbits of departure should be as close to the surface as possible for maximum economy. Since they are of temporary nature, orbital altitudes 350 miles, perhaps down to 300 miles, appear feasible. In a sense (Fig. 2), the best orbits of departure lie thus "deep" in the atmosphere, namely in the lower portion of the mesosphere. The orbital inclination is determined by orbital altitude, plane of the transfer ellipse, and assembly

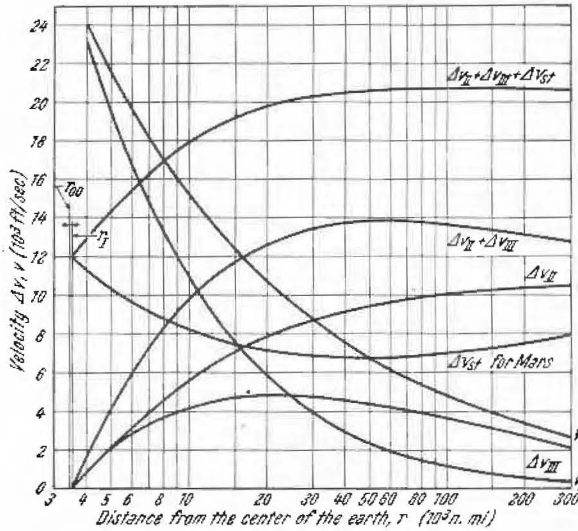

Fig. 9. Effect of supply system on selection of assembly orbit of interplanetary expedition to Mars.

time. Usually, the orientation of the interplanetary transfer ellipse will be similar to that of the ecliptic plane ($\delta = 23.5^0$); so, at one time, the orbit of departure must also lie in this plane. For this inclination and for an altitude of 300 to 400 miles the period of regression (Fig. 5) T_{pr} is approximately 55 days. In [4] an example has been presented of a small Mars expedition, involving 8 persons and 3 ships. Its assembly requires about 600 flights with the 11,000 lb payload automatic supply ship. The assembly time, utilizing the advantages of prefabrication on the earth to the utmost extent would be not more than 4 to 5 months. During this time the assembly orbit completes roughly 2 to 3 periods of regression. The orbital plane oscillates between -23.5^0 and $+23.5^0$ inclination with respect to the equator. The trajectories of the supply ships must follow this change; that is, they must be varied from ascent to ascent.

3. *Orbits of Arrival*

Althongh the supply system requirements supersede the recommendation of Eq. (9) concerning the optimum orbit of departure for planet-bound space ships, the above analysis nevertheless applies unchanged to the orbits of arrival. Arriving in the optimum orbit saves mnch propellant which must be taken as payload through all preceding propulsion periods and which, therefore, is particnlarly expensive.

The crew is picked up by an auxiliary ship, or the space ship is refueled from the earth. This energy, freshly supplied from the earth after return, is applied much more economically, since it has not been carried through part of the solar system prior to its use.

Since Venus and Mars are the only planets likely to be visited in the foreseeable future, it follows, therefore, that orbits of arrival from interplanetary expeditions can be expected to lie very far out, namely at about 40,000 miles distance. Consideration of the energy requirements for picking up crew aud equipment, however, may lead to a compromise orbit at a somewhat reduced distance of perhaps 25,000 to 30,000 miles.

4. Auxiliary Orbits

Standard supply ships will not be capable of reaching orbits thus far out, even without payload. In this case it is necessary to introduce an auxiliary orbit which is occupied temporarily for the purpose of transfer of propellant, material, or passengers. It is taken over from the ascending supply ships by a space ferry which carries the load into the far-out orbit. This has been proposed in [3] and also in [10] as a method of restricting the size of supply ships without unduly reducing the transport capacity.

It appears that the auxiliary orbit can be located as close as 120 miles above the surface for a few revolutions (meaning several hours). In the present case, when personnel and equipment is to be picked up in an orbit of arrival, the winged passenger ship(s) must be refueled and provided with additional propellant tanks or boosters in the auxiliary orbit.

From the discussions in Sub-Sections 31 through 34 it can be concluded that nearly all orbital establishments preferably should be located as close to the surface as feasible. For observational satellites, altitudes between 400 and 700 miles (650 to 1,100 km) are indicated. Assembly orbits can lie at even lower altitude, in the range between 350 and 400 miles (560 to 650 km), depending on the assembly time required. Extreme cases are the auxiliary orbits and the orbits of arrival from interplanetary space, the former being located around 120 miles (200 km), the latter somewhere between 25,000 and 40,000 miles (for Venus and Mars expeditions), depending on the recovery effort regarding the residual hardware in the orbit of arrival. Probably not much, if any, of this will be recovered. The smaller these efforts are — being, in the extreme case, restricted to picking up the personnel and scientific material — the closer can the orbit of arrival be to its theoretical optimum, roughly at 49,000 miles. These extreme orbits are occupied only occasionally and then for very short periods of time.

Two discrete ranges of orbital inclinations can be anticipated: Large angles of inclination between 60^0 and 80^0 for observational satellites and small angles around 23^0 for orbits of departure and arrival. Auxiliary orbits needed for flights into orbits of arrival will, therefore, also have inclinations around 23^0. For lunar expeditions any orbital inclination may occur.

5. Supply Requirements

The supply requirements depend very much on the level of effort which the contractor is willing to buy.

For the purpose of the present discussion, a permanent crew of 4 persons has been assumed tentatively for the observational satellite. This is not meant to imply that such a satellite could not be smaller; it only means that, in the author's opinion, it does not have to be larger in order to fulfill its observational and some scientific functions reasonably well.

It has been stated in [4] that an adequate satellite for 4 persons would have a volnme of about 20,000 cu ft (566 m^3) and a surface weight of not more than about 500,000 lb (abont 225 metric tons), reqniring a total of about 50 successful flights with the large automatic supply ship. More details are presented in Section 4.

The supply demands for this satellite would be of the order of 2,000 lb (about 1 t) per month.

For lunar circumnavigations, [11] and [4] independently arrived at a weight of about 500,000 lb in the orbit of departure, reqniring ronghly 50 successful flights of an 11,000 lb payload supply ship. In other words, this endeavor is of the same level of effort as the observational satellite.

Regarding the minimum permissible limit of effort for successful interplanetary expeditions, there exists yet a considerable divergence of opinion, with [12] anticipating a particularly high minimum level of effort. Whether or not this is correct will be born out by the experience gained in the process of establishing a terrestrial (observational) satellite and conducting circumnavigations of the moon. For a Mars expedition with 8 persons and 3 ships the weight in the orbit of departure is abont 6.8 million pounds [4] for the case of return into the optimnm orbit of arrival. This reqnires a total of about 600 supply flights with the large supply ship; a level of effort which is by a factor of ten larger than for observational satellites and lunar circumnavigations.

Of primary interest for the layout of orbital supply systems are observational satellites, the development of space ships, circumnavigation of the moon and perhaps a small-scale landing on its snrface; in other words phase *B* and *C* in Fig. 1.

IV. Missions

The primary objective of an orbital supply system is the accomplishment of the two principal missions:

1. Material supply.
2. Transportation of personnel.

For highest economy and efficiency, both missions should not be combined in one vehicle.

The required capacity of the material supply system is very different for establishing and for maintaining a satellite. Therefore, a large and a small automatic supply ship has been proposed.

The large ship thus will be used for a limited time only. Since the upper stage is wingless and does not return, it should be laid out in such a manner that most of its components can be utilized as constrnction elements in space. This, to a certain extent, determines the layout of large snpply ships of which more details are presented below.

The components which are most promising as construction elements are the payload section and the propellant tanks.

1. Observational Satellite

Of particular interest for the satellite are the propellant containers of stage 3 of the large supply ship. In order to utilize them to best advantage, the layout of the satellite must take these units into account as construction elements.

Since Noordung [13] has proposed the wheel- or doughnut-concept, this type of satellite design has been very much in vogue, because it allows for rotation of the system, thereby producing apparent gravity or weight in the tube of the wheel. This concept, however, has two important shortcomings: it requires a comparatively large satellite body and it requires a very delicate balancing system.

The centrifugal acceleration is given by $v^2/\varrho = \omega^2 \varrho$ where ϱ is the radius of curvature and ω the angular velocity of a point. Hence, for a given acceleration, either ω or ϱ must be large. Of these two parameters the angular velocity is subject to more stringent limitations for obvious physiological reasons and in order to keep the Coriolis acceleration, another effect resulting from the body's inertia, at reasonably low level. This acceleration is given by the relation $2\,\omega\, d\varrho/dt$. For a given angular velocity the Coriolis acceleration is proportional to the radial velocity of a body, because it represents the inertial resistance of the body mass to the change in angular velocity with changing radial distance (Fig. 10). In the course of normal activities on the satellite such changes in radial distance will occur frequently. Now, if ω is very large, even small values of $d\varrho/dt$ will produce a considerable Coriolis force, thereby making working and living conditions in the satellite difficult or even intolerable. Consequently, the wheel must have a large radius ϱ and this requirement automatically leads to unnecessarily sizeable satellites.

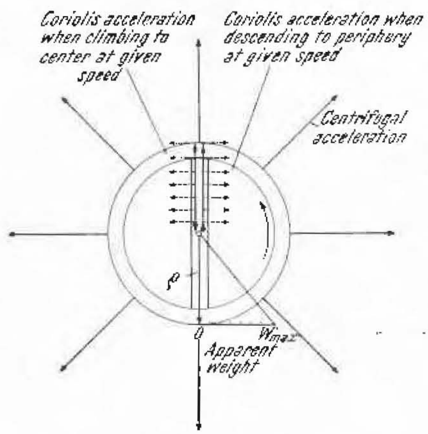

Fig. 10. Inertial conditions in a satellite.

In a wheel-type satellite most of the mass is located in the peripheral tube, that is, at the greatest possible distance from the common center of gravity. Consequently, a very delicate mass balance must be maintained, because every irregularity in the peripheral mass distribution will produce the maximum possible moment. This makes activities on the satellite a rather clumsy affair and much additional weight in the form of liquids for balancing purposes is required.

In order to produce a functional satellite which combines the advantage of apparent gravity with smaller size and a less precarious balancing situation, it is proposed to concentrate most of the mass in a center body from which two extensions, oppositely directed, lead to the crew space. This design permits the use of comparatively long arms, hence of large apparent gravity at small angular velocity. Fig. 11 presents a schematic sketch of a 4-man observational satellite, designed according to this principle. Some characteristic data are presented in Table I.

The radial mass distribution diagram on the right hand side of Fig. 11 indicates that most of the satellite's mass is concentrated in the center which contains all stowage, reserve parts, emergency equipment, radioactive power supply with shielding material, earth scanning equipment, purifiers, attitude control and, of course the entrance tubes on the hub with the double air locks. In the peripheral sections are located the living and working quarters (for 2 persons on each side), control motors and actuators.

As pointed out before, the large concentration of mass in the center permits more freedom of action in the satellite, particularly in the peripheral parts with less danger of upsetting the balance of the system. For this reason only two extensions, the smallest possible number for reasons of symmetry, have been assumed. Another advantage of central mass concentration in this type of design is the minimization of radial stress in the extension tubes and in the peripheral sections.

Table 1. *Characteristic Data of a 4-Person Observational Satellite*

Overall weight	500,000 lb
Volume	approx. 20,000 cu. ft (283 cu. m.)
Mean specific weight	approx. 25 lb/cu. ft (0.4 tons/cu. m.)
Angular velocity	0.2295 rad/sec
	13.15 deg/sec
Period of revolution	27.4 sec
Number of revolutions	2.185 rpm
Centrifugal acceleration (190 ft level)	10 ft/sec^2
	0.31 g
Coriolis acceleration (at $d\rho/dt = 5$ ft/sec)	2.29 ft/sec^2
	0.071 g

The data in Table I show that reasonable conditions with respect to available space, centrifugal acceleration (which should be high) and Coriolis acceleration (which should be low) are obtained in this comparatively small orbital establishment. At a radial speed of 5 ft/sec it takes a man only about 40 seconds to cover the radius in its entire length. During this time he would be subject to a Coriolis force of only 7 per cent of his terrestrial weight (about 12 lb), or to about 21 per cent of his peripheral weight (that is, as if the load were about 36 lb on earth). As he reduces the distance from the center, the Coriolis force is progressively felt more strongly, since the man loses apparent weight. The radial apparent "g" distribution on the left hand side of Fig. 11 indicates that at a distance of about 50 ft from the center, the Coriolis force equals the apparent weight. In satellites with higher angular velocity this point lies at a greater distance from the center; which is less favorable inasmuch as it raises the disturbance level of the Coriolis force.

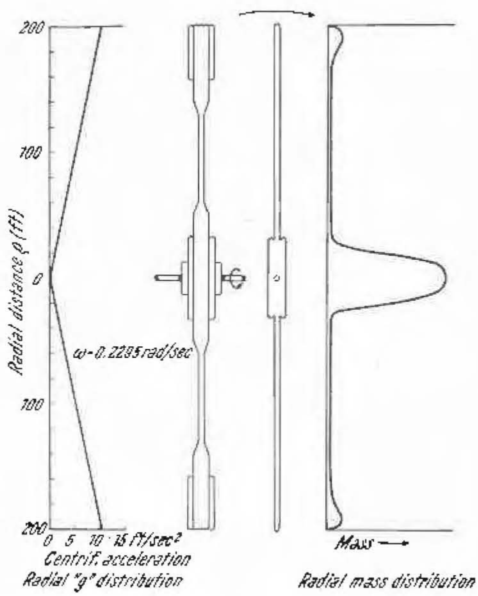

Fig. 11. Small observational satellite.

For a satellite concept as indicated in Fig. 11 the propellant containers of stage 3 can be utilized to a considerable extent. Details of this design are beyond the scope of the present paper.

2. *Other Orbital Installations*

Except for the observational satellite, all other orbital installations are of temporary nature, permitting (within reason) somewhat less elaborate provisions for living space.

Assembly crews in orbits of departure must operate under conditions of complete weightlessness. At present it is not known whether or not such a crew, in its leisure time, should be subjected to apparent gravity as in the satellite where, however, the crew is practically all the time subject to apparent gravity. The change from weight to weightlessness and vice versa within intervals of few hours possibly is much less attractive then continuous weightlessness for a period of days or weeks in a row. Pending a clarification of this point which is part of the space medical research conducted during phase *B*, the living quarters of assembly crews will or will not rotate.

In any case, however, the necessity can be anticipated of establishing special living quarters as initial construction phase in the orbit of departure. The cabin space in passenger ships is by far too limited to accomodate persons for days or weeks. They may only be used for the first one or two days, until the first living quarters are established. Consequently, it will again be necessary to use parts of stage 3 as construction elements for living quarters.

V. Automatic Supply Ships

The automatic supply ships are characterized by the absence of all provisions required for a human crew. This fact implies a greatly increased freedom of functional design.

1. The Large Automatic Supply Ship

Primary fnnction of this vehicle is to establish rather than maintain orbital installations. The third stage which enters the target orbit does not return to the earth. Therefore, the design of this stage should take possible utilization, at least in part, as construction element for the orbital installation into account; in other words, the design criteria of stage 3 are determined by space conditions rather than by the short period of atmospheric flight.

The vehicle (Fig. 12) consists of
a large booster-type first stage
a second stage of annnlar cross-section
a third stage, partly contained in the hollow center of stage 2.

Stage 1 is designed for parachute recovery. It supplies a momentum thrust of about 1.74 million pounds (about 800 metric tons) from 5 main motors and 4 tiltable twin-motors.

Stage 2 is considered expendable and, therefore, can be designed for a one-way mission. The reason for making this stage expendable is that its empty weight (structure, power plant and equipment) is less then 20 per cent of the overall empty weight of the vehicle. Moreover, its impact point is considerably more distant from the launching site thau that of the first stage. This results in a more time consuming and expensive recovery operation. Finally, the second stage is exposed to considerably greater thermal and mechanical stress during the descent than the first stage, so that it appears doubtful whether a reconditioning would be possible at all; and if so, the job will be correspondingly more difficult aud time-consuming and perhaps ultimately lack the required reliability. In the light of these considerations, parachute recovery of stage 2 does not appear practical.

Stage 2 has 4 main motors and 4 tiltable (hinged) control motors. Together they supply a thrust of about 279,600 lb. The operational momentum thrust, however, is 346,000 lb (about 158 t), because the thrust of stage 2 is augmented by the motors of stage 3 which add another 66,400 lb. This joint operation during which the motors of stage 3 are fed by the propellant supply of stage 2, has the advantages of reducing the upper stage power plant weight in general and that of the completely expendable second stage in particular, and of increasing the reliability of the final stage separation, since the motors of stage 3 are already ignited.

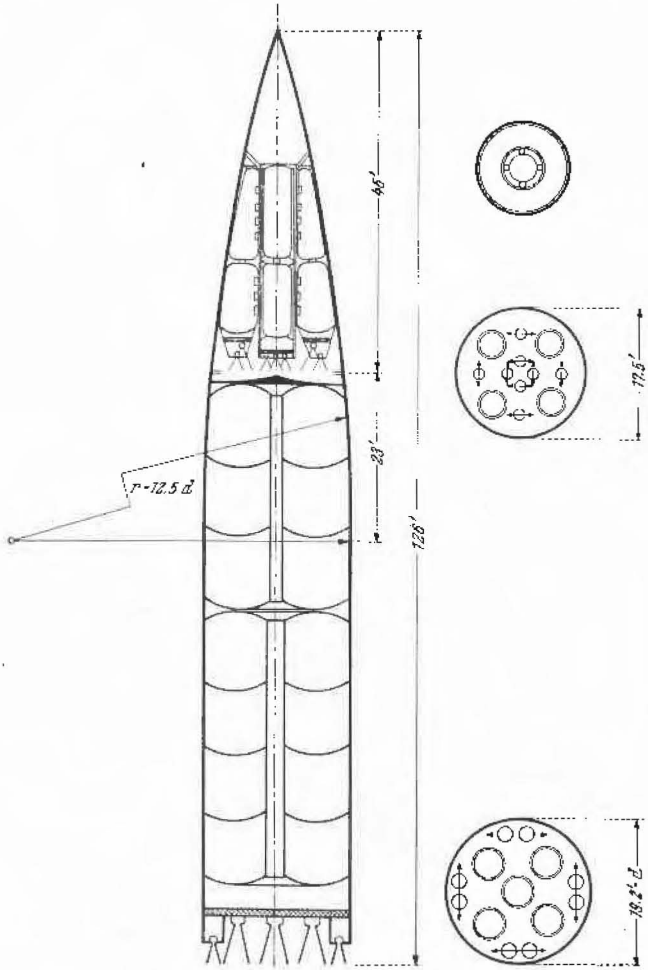

Fig. 12. Schematic sketch of 11,000-lb supply ship.

Stage 3 consists of 4 engines, a set of tanks and a guidance and payload section, shortly referred to as nose section. Propulsion system and tanks are inside stage 2. The propulsion systems of both, stage 2 and 3 are on an equal level, hence are accessible through the same door. The containers are completely protected throughout the entire atmospheric ascent. The combined oxidizer-fuel set measures about 5.3 ft in diameter and has a length of 20 ft, yielding a volume of abont 450 cu. ft. Table II presents some characteristic data of the vehicle (for a more detailed tabulation cf. [4]).

The nose section contains the guidance and control system and the payload proper. Different nose sections can be employed to suit the particular mission.

In order to make the supply ship flexible in this respect, the nose section has been kept free from protection by the second stage. Within limits, the shape and volume of this section can be varied.

a) Power Plant

The total number of rocket motors in all stages is 25 [1]. The number of motor types (i. e. different combustion chambers and injection systems) is 3, namely

Type 1	285,000 lb thrust	low-altitude nozzle
Type 2	40,000 lb thrust	low-altitude nozzle
	53,500 lb thrust	high-altitude nozzle
Type 3	16,600 lb thrust	high-altitude nozzle

Table II. *Large Automatic Supply Ship*

Stage	1	2	3
Payload (lb)	232,000	46,000	11,000
Dead weight (lb)	137,000	31,300	6,650
Effective propellant weight[2] (lb)	931,000	148,800	27,300
Auxiliary fluids (lb)	40,000	5,900	1,050
Gross weight (lb)	1,340,000	232,000	46,000
Momentum thrust per stage (lb)	1,741,000	340,000	66,400
Overall mass ratio		28.5	

One propellant is used in all stages, for reasons of efficiency and simplicity: Liquid oxygen and hydrazine (or, as far as weights, volumes, and stage performance are concerned, any propellant of comparable density and specific impulse). Using O_2—N_2H_4, the propellant data are presented in Table III.

Table III. *Propellant Data*

Stage	1	2	3
Propellant	oxygen - - hydrazine		
Mixture ratio (O/F)	0.9	→	→
Chamber pressure (psia)	590	→	→
Theoretical specific impulse (shift. equil.) (sec)	305	350	350
Velocity correction factor	0.94	0.93	0.93
Used specific impulse	295	325	325

It is realized that the mixture ratio selected will make conventional (unrestricted) regenerative cooling difficult. The theoretical combustion chamber temperature is approximately 5,590 °F (3,350 °K). Assuming a combustion efficiency of 96 per cent, the actual chamber temperature becomes of the order of 5,300 °F (3,200 °K). At this temperature, and at the high pressure employed, the heat flux density becomes very high, about 10 Btu/in^{2}sec at the throat of the 16,000 lb thrust motor, that is, approximately 2 to 3 times as much as present day values. It cannot safely be predicted whether hydrazine, a metastable compound, is capable of operating under such severe conditions without decom-

[1] Not counting a very small motor for final velocity adjustment of stage 3.

[2] Defined as the propellant weight required for accelerating the vehicle.

position. One principal alternative is, of course, to use liquid oxygen, at least for cooling part of the motor. In this case the cooling jacket pressure would have to be increased slightly above the critical pressure of oxygen which is about 51 atmospheres. Aside from this, there are other methods considered, combining the use of highly heat resistant materials with an appropiate design, which gives reason for the assumption that such motors can be operated 10 years from today. Details are beyond the scope of this paper. The importance of developing new types of motor cooling, in order to take advantage of high performance propellants for orbital flight, is obvious.

The propulsion system of the first stage consists of 5 main motors of the type 1 mentioned before, and of 4 sets of twin motors of type 2 with low-altitude nozzle, that is, exit pressure 0.7 atm. The main motors are mounted rigidly, arranged as shown in Fig. 12. The 4 twin motors are tiltable, serving as control motors: 2 for pitch, two for yaw and roll. Twin motors have been taken, in order to utilize the type 2 motors developed for stage 2.

Each main motor and each control twin motor is equipped with a complete feed system, consisting of an O_2—N_2H_4-operated gas generator, gas turbine, and one pump each for oxygen and fuel. The fluid pressure resulting from the pressure due to gravity head and from the gas pressure in the tanks, provides ample pressure on the suction side of the pumps.

The tankage — and, hence, the whole body of stage 1 — is subdivided into two sections, one above the other. This has been done for the following reasons:

1. The large body of stage 1 can be transported and handled, as well as checked, more conveniently if it consists of 2 separate sections.

2. If, during pressurization tests prior to assembly and firing, a tank leakage is discovered, then only the section concerned, not the whole stage has to be exchanged. This means a special simplification if leakage or damage occurs in the upper section.

3. Misalignments can more readily be corrected.

4. The lower section is a straight cylinder which experiences little heating during the ascent. Therefore, it probably can consist of a light metal alloy. For the upper section which already contains the transition to the ogive this may not be the case. If, for reasons of temperature and pressure, this section must have a steel wall, then the separation into two sections has the advantage of making the use of different construction materials possible.

5. For reasons of low weight and better handling and transportation no tail fins are provided. In order to counteract the effect of their absence on the stability conditions during the ascent — that is, to reduce the instability to a level where it can be handled by the control motors as far as control moments and response time to commandos is concerned — the center of gravity (CG) must be kept as far forward as possible, particularly during the first portion of the flight, when the dynamic pressure is still high. One effective way to accomplish this is the method of programmed emptying of the tanks. In this case the lower section is emptied first. This section is slightly larger (containing about 60 per cent of the propellant), in order to place the division line between the two sections as far forward as feasible.

Stage 2 has 4 rigidly mounted main motors of type 2 with high-altitude nozzles ($p_e = 0.1$ atm), and 4 tiltable control motors of type 3 for pitch, yaw, and roll trimming. These motors ignite at the moment of separation from stage 1. One feed system of the same type as used in stage 1 serves one main motor plus one control motor together so that a total of 4 feed systems for about 70 000 lb of thrust each is required.

At the same time stage 3 is ignited. This system consists of 4 tiltable motors which at the same time serve as control motors in pitch, yaw and roll after stage 3 is on its own. There is one feed system for all 4 motors.

From the low pressure (suction) side of one of the stage 2 feed systems with correspondingly dimensioned feed lines one fuel and one oxygen line is branched off, leading into the respective suction lines of the stage 3 pumps, upstream of the point where fuel and oxidizer is tapped for the gas generator. With this arrangement the stage 3 feed system can operate normally with the only difference that the propellants come from another tank system.

Immediately preceding the point where the supply lines leave stage 2 there is a guillotine cutting device and just inside stage 3 is a plug valve. Where the supply lines enter into the suction lines of the stage 3 pumps, a two-way valve is located. During the operation of stage 2, this valve blocks the suction lines coming from the stage 3 tanks. About 2 or three seconds prior to separation, the guillotine device, electromagnetically actuated by a timing device, cuts the supply lines which thereupon are sealed promptly by the plug valves due to the overpressure from the stage 3 side, since simultaneously the 2-way valve has opened the suction lines to its own tanks. Thus stage 3 already operates independently when the motors of stage 2 are turned off.

b) Structure

Stage 1, subdivided into two sections as explained in the preceding section, consists of integral tanks which are equipped with splash plates, in order to prevent damage to the containers by the impact of residual fluid after cut-off, caused by sudden deceleration due to drag and pull of the parachute. The forward end of the tanks is protected from the gas jets of stage 2 as well as from the air jet after separation by means of a jet deflector.

A yoke above the main motors with outriggers to the control motors collects the thrust force and, through a thrust frame consisting of 16 beams (about 110,000 lb or 50 tons per beam) induces the force into a collector at the base of the main structure below the tanks. This section below the tanks is a re-enforced semi-monocoque construction. From there the thrust force is equally distributed into the integral tank body structure and into the wall of the second stage.

The length of the cylindrical portion is 57 ft at a diameter of 19.2 ft, yielding a total volume of 16,400 cu. ft. The ogive meridian is a circular arc section with a radius equal to 12.5 calibers. Its total length is 69 feet. Of the total volume of 11,200 cu. ft, approximately 0.58 belong to stage 1, 388 cu. ft (11 m^3) or 0.034 are nose section volume, and the rest is stage 2 plus the shielded portion of stage 3. The semi-vertex angle of the ogive is 16.1 degrees. The overall surface area of the ogive section is 2,800 sq. ft (262 m^2), and the surface area of the cylindrical section is 3,450 sq. ft (322 m^2). Hence, the total vehicle surface is 6,250 sq. ft or 584 m^2.

In stage 2 the thrust is evenly distributed to the inner and outer wall of the annulus, in order to prevent shear forces on the propellant tanks. Normally, with gas pressure in the propellant tanks there would be the danger of buckling of the inner cylinder, causing the structure to be fairly heavy in this portion. Actually, stage 2 is never required to operate with a hollow inner cylinder. Stage 3 automatically supports the walls. If, in addition, light re-enforcements are provided on the inside of the tanks, as indicated in Fig. 12, sufficient structural strength is obtained at little weight penalty.

In stage 3 the thrust frame leads straight up from the motors to the nose section. Four tubes are provided which are hollow and which contact the inner cylinder wall of stage 2 where they are guided so that stage 3 cannot rotate either before or during the expulsion process.

The containers are suspended betweeu the columns of the thrust frame. Therefore they do not have to carry any load, except their own. They can be made of very thin aluminum sheet, double-walled for protection against meteoritic dust. Top and bottom are removable so that both coutainers can be mounted together in space as one construction element. Details as to its use are beyond the scope of the present paper.

The joint thrust force of both stages 2 and 3 is so distributed that stage 3 always has a certain weight with respect to stage 2. This weight increases during the joint burning, as stage 2 is emptied while stage 3 remains fully loaded. The payload nose of stage 3 rests on a conical bearing on top of stage 2 so that lateral displacements dne to inertial forces are also avoided. Because of this relative weight of stage 3, premature separation normally cannot occur. At burn-out of the second stage the thrust of stage 3 produces separation.

c) Gnidance and Control

Automatic guidance and control for all three stages originates from the nose section. The first and second stage contain only the control motors and their actuators.

The vehicle is launched vertically and, by a 3-step program device (1 tilting program for each stage) is directed into a circular orbit, and from there into a transfer ellipse which contacts the target orbit. The path deflection for the first stage is of the order of 90 to 15 degrees, for the second stage 15 to 4 degrees, and for the third stage 4 to 0 degrees. Now, it is not so very important at which altitude exactly the third stage reaches zero degrees trajectory angle. More important is that, upon completion of the program of deflection, the vehicle has local circular velocity. Transition into the transfer ellipse requires only a small additional velocity increment which preferably is not produced by the 4 main motors but by a small auxiliary motor of about 1,000 lb thrust. When the powered flight is completed, the vehicle must possess an initial velocity which will carry it along the unpowered elliptic path as shown in Fig. 13.

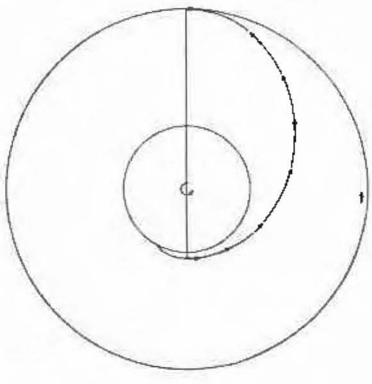

Fig. 13. Ascent into the target orbit.

While the vehicle moves along this transfer ellipse, its axis must be turned by 180 degrees in order to have correct attitude for the third and final propulsion period at the target orbit. For a target orbit altitude of 600 mi the cruising flight time is about 47 minutes (depending somewhat on the cut-off altitude) or, roughly, 2,800 seconds. In the case of continuous turning, the angular velocity must therefore be of the order of 0.064 deg/sec, a rate which can easily be maintained by a comparatively small attitude device of about 100 lb weight. The weight of stage 3 during the cruising period is approximately 18,500 lb. Aside from the attitude control in pitch which maintains the tangential direction of the vehicle nose with respect to the flight path, attitude control in yaw and in roll is required, in order to ascertain correct positiou of the vehicle in all three axes at the time of the final propulsion period. This period again is very short, involving velocity increments of the order of a few hundred feet per second only, for the orbital altitudes under consideration.

2. *The Small Automatic Supply Ship*

The small automatic supply ship serves to maintain an established orbital installation. Therefore, it is distinguished from the large supply ship by a considerably smaller payload capacity and by the fact that it has to operate over a much greater length of time. Consequently, the recovery problem becomes particularly important for small snpply ships where every improvement in economy of operation, however slight, is augmented significantly by the length of operation. For these reasons, the question as to which configuration should be selected for small snpply ships is of primary interest. In general, the functional operation of the vehicle will be very similar to that described above in connection with the large supply ship so that this does not have to be repeated here.

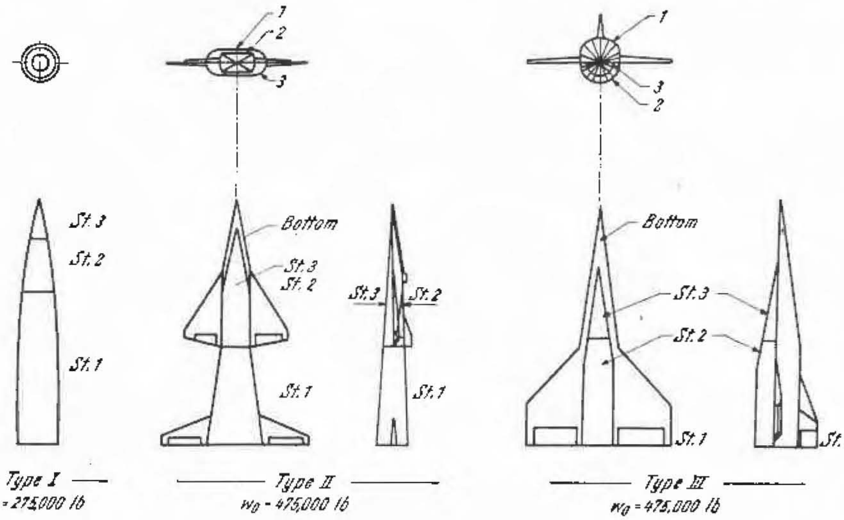

Fig. 14. Three configurations for 2,000-lb supply ship.

Fig. 14 illnstrates schematically 3 basic types of design:

Type I conventional concept of multi-stage vehicle with series arrangement of stages.

Type II provides parallel arrangement of second and third stage, mounted on a booster rocket. This is actually a modification of the telescope arrangement used for the large supply ship.

Type III represents an arrangement in series of second and third stage, both being mounted parallel to the first stage.

Type I could be realized more conveniently in the present case due to the smaller size of the maintenance ship. However, like in the case of the large supply ship, the second stage probably will have to be written off as a total loss which, for reasons mentioned above, does not make this solution very attractive. In addition it requires more power plant weight and has one more ignition period than the large supply ship. The main advantage of this type, as compared to the other two types depicted in Fig. 14 is its low weight, because it does not have wings. However, from the viewpoint of reducing weight by omitting wings, it might be more attractive to choose Type III with wingless first stage. One then gets about the same low take-off weight, but avoids the disadvantages of the first type. The wingless versions would weigh about 275,000 lb for a payload capacity of 2,000 lb.

Type III with winged first stage is based on a suggestion made by Dr. DORNBERGER[1]. The inherent advantage of this type is gained by equipping the first stage with wings. In the form of an airplane the first stage is most easily and conveniently recovered by directing it (either by pilot or by remote control) back to the launching site. Moreover, joint operation of the first and second stage has an even greater engine saving effect that the telescope arrangement or, for that matter, Type II, in spite of the fact that the second stage is equipped with high-altitude motors and therefore operates with flow separation during take-off.

Type II has been studied independently by the author in 1951. This configuration has originally been used for the passenger vehicle (see below) with the only difference that in the latter case the third stage is winged, while in its application to the small supply ship, the second stage has been provided with wings. This stage attains considerably higher speed and altitude than the winged stage in Type III. Its maximum flight velocity is of the order of $M = 20$, while that of the first stage is half as much. Consequently, a winged second stage principally faces more severe flight conditions, particularly with respect to aerodynamic heating.

It must be kept in mind, however, that these flight conditions must be explored and mastered anyhow, in order to make an orbital passenger ship — hence, human space flight — technically feasible; and a returning orbital passenger ship must be subjected to even more severe conditions than a winged second stage.

The justification in the present case for giving preference to a winged second rather than first stage lies, in the opinion of the author, in the fact that recovery of the first stage can be accomplished by means of parachute (at least this is the working assumption throughout the present report), while the second stage cannot be recovered in this manner. For maximum economy of the maintenance operation which requires recovery of both, first and second stage, it appears, therefore, that the second stage must be equipped with wings, while the first stage does not need wings. For this reason, Type II appears, for this particular purpose, most attractive.

With 2,000 lb payload and based on oxygen-hydrazine, Type II weighs about 475,000 lb. If emphasis is on low take-off weight (which, in this case however, does not mean good economy), then Type III with wingless first stage appears to be most promising, its take-off weight being of the order of 275,000 lb. For more data on the low-weight configuration cf. [4].

3. Flight Performance

The total energy of the supply ship's rest mass after completion of the ascent is given by the sum of kinetic energy in the orbit and of the potential energy difference between surface and orbital altitude. The velocity which is equivalent to this energy constitutes the minimum velocity required for transfer into the given orbit. The term minimum velocity implies that no losses occur while the energy is produced, that is, during the various propulsion periods. The minimum velocity is given by the relation

$$v_{min} = \sqrt{\frac{\gamma}{r_s}\left(1 + \frac{2\,y_s}{r_{00}}\right)} = v_c \sqrt{1 + \frac{2\,y_s}{r_{00}}} \tag{10}$$

[1] The author is greatly indebted to Dr. W. DORNBERGER, Guided Missile Consultant at the Bell Aircraft Corporation, for many discussions on this subject.

and is plotted in Fig. 15 as function of the altitude y, together with the circular velocity. The three propulsion periods of an ascending orbital vehicle have the purpose of attaining circular velocity at a low altitude, say, of 350,000 ft (Phase I), entering the transfer ellipse (Phase II), and finally of entering the target orbit (Phase III). Fig. 15 shows that at an altitude of 350,000 ft, the minimum velocity is about 26,200 ft/sec. At 700 miles (3.7 million ft) altitude, which we presently assume as limiting altitude for observational satellites and for orbits of departure, the minimum velocity is 27,800 ft/sec. The minimum velocity required for Phase I, therefore, is about 94 per cent of the minimum velocity required for the target orbit.

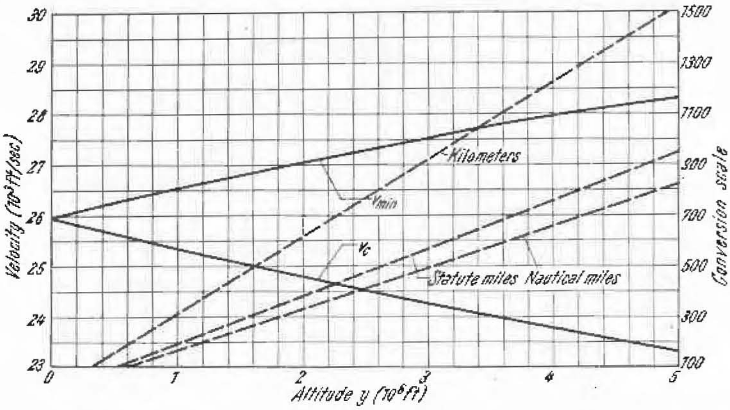

Fig. 15. Minimum velocity and circular velocity.

Losses during the propulsion periods are due to gravitational pull, aerodynamic drag, and steering forces which cause the resulting thrust axis to deviate from the flight direction. By proper selection of the flight program steering losses can be kept very small. Drag losses are overcompensated by additional propulsive force due to increasing pressure thrust as the vehicle gains altitude; this holds true for large, slowly ascending 3-stage orbital vehicles of which two stages are operating practically outside the atmosphere, that is, at technically negligible ambient pressure. By far the largest loss is caused by gravitational pull. In the case of gravity losses there are no countervailing gains, like for the drag losses. From numerous trajectory calculations it follows that the overall loss during the first propulsion phase is about 85 to 93 per cent of the circular velocity at 350,000 ft altitude, with 90 per cent being a fairly good average. Since nearly all of the minimum velocity is produced during the first propulsion phase, as stated above, it is simpler to take the minimum velocity of the target orbit and divide it by 0.9. This then corresponds to a loss during the first propulsion period of about 85 per cent. and is, therefore, on the conservative side. The resulting velocity of 31,000 ft/sec (9.5 km/sec) is the ideal velocity on which the layout must be based. No return propellant is required for automatic supply ships.

Vacuum conditions are reached prior to the burn-out of stage 2. This is important, because the interference effect of aerodynamic forces would make the expulsion of stage 3 extremely difficult, if not impossible. At the given dimensions and with an initial acceleration of 1.44 g (cf. tentative trajectory data presented in [4]), it takes stage 3 less than 2 seconds to emerge from the duct.

The expulsion process is characterized by the requirement that both vehicles must be in a condition of free fall. This is to avoid that stage 2 which then is no longer burning, has "weight" with respect to the third stage. This weight would produce an excentric load on that portion of stage 3 which is still inside the duct. This load would increase as the respective centers of gravity move apart, until separation is completed.

At the time of separation of stage 3, the trajectory angle is still different from zero, although its value is small, of the order of 4 degrees. Fig. 16 *a* shows that if, during separation, the thrust F_3 of stage 3 points in flight direction ($\alpha = 0^0$), then the apparent weight (which includes the centrifugal effect) $W_2' = \left(1 - \frac{v^2 r}{\gamma}\right) W_2$ of the second stage produces a load $F_3 \sin\theta$ on stage 3. At the velocity of about 18,000 ft/sec, existing at that time, $W_2' \simeq 0.5\, W_2$, if W_2 is the instantaneous zero-velocity surface weight of the second stage. Of the weight W_2' the fraction $(F_3/W_2') \sin\theta$ or $(2\, F_3/W_2) \sin\theta$ is, under the conditions of Fig. 16 *a*, supported by stage 3. As the center of gravity $(C.G.)_3$ moves forward and away from $(C.G.)_2$, this weight fraction produces a moment in pitch which tends to tilt the nose of stage 3 in upward direction. The residual weight of stage 2, contributing to the gravity deflection of the trajectory is then $W_2' - F_3 \sin\theta$, or, as far as the interesting weight component normal to the flight direction is concerned, $(W_2' - F_3 \sin\theta) \cos\theta$.

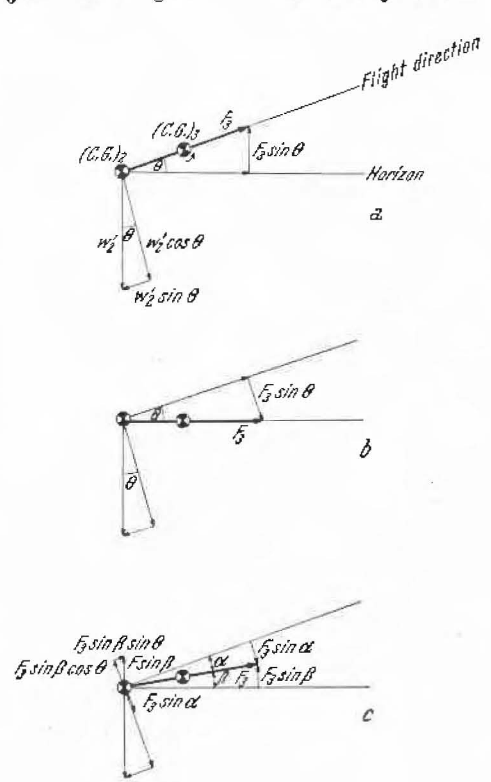

Fig. 16. Expulsion of stage 3. (*a*) Thrust inflight direction. (*b*) Thrust horizontal. (*c*) Correct thrust direction.

If the thrust is horizontal (Fig. 16 *b*), this weight component is increased beyond the value resulting from gravity alone, owing to a thrust component $F_3 \sin\theta$ which now is pointing in downward direction. This condition causes an apparent centrifugal force of the same magnitude, acting in the opposite direction, tilting downward the nose of stage 3.

Fig. 16 *c* shows the correct case where the thrust direction is at a negative angle of attack, somewhere between the two extremes. The correct negative angle of attack α is determined by the condition

$$F_3 \sin\alpha = F_3 \sin\beta \cos\theta, \tag{11}$$

that is, the centrifugal effect and the weight fraction to be supported by stage 3 must balance each other, eliminating any moment with respect to the C.G. of stage 3, thereby creating a condition which is equivalent to that of free fall of both centers of gravity as they glide apart. The free fall in this connection refers to the motion normal to the flight direction. The weight component $W_2' \sin \theta$, which now is no longer supported by thrust, expedites separation. The required angle of attack is given by

$$\sin \alpha = \sin \beta \cos \theta. \tag{12}$$

where $\beta = \theta - \alpha$. Therefore, the correct angle follows from trial and error. In the present case θ is very small, hence $\cos \theta \sim 1$ and $\sin \alpha \simeq \sin \beta \simeq \sin \theta/2$ or $\alpha - \theta/2$. With a trajectory angle of 4 degrees, it follows that the required angle of attack at the beginning of the separation process must be -2 degrees. During the short time of separation the trajectory angle changes very little, so that in the first approximation a constant angle of attack can be assumed. Stage 2 preferably is tilted, by a special program, into this angle of attack immediately before stage 3 ceases to obtain propellant from the second stage and switches to its own supply (cf. V, 1, a, p. 38).

The importance of complete absence of perceptible dynamic pressnre during the separation is apparent from the preceding discussion. For this reason the vehicle must clear the atmosphere, before a stage separation of this type becomes practical. This is one important reason — aside from several others — why this system cannot be applied to the first and second stage.

VI. Orbital Passenger Ship

Passenger vehicles are characterized by at least one winged stage, the upper stage, which provides accomodations for personnel, but not for a cargo-type payload. This restriction is of fundamental importance, as it eliminates design compromises which inevitably carry a severe penalty in terms of size, weight, aerodynamic quality and safety. It permits the functional design of the winged passenger stage as a hypersonic glider.

1. *Description*

The ascent of a passenger ship into the target orbit is not different, except in details, from that of an automatic supply ship. The important part is the return into the atmosphere, by far the most difficult and, presently, least understood portion of the flight path of an orbital vehicle. It will be seen that this fact alone — aside from all advantages inherently associated with the use of automatic supply ships — suggests strongly a functional separation of load-carrying and passenger carrying vehicles.

The return into the atmosphere determines the design criteria of the upper stage. These criteria call for

good aerodynamic (particularly hypersonic) qualities
structural reliability
transport safety.

This shows that for the upper stage of the passenger vehicle, emphasis lies on entirely different featnres than in the case of stage 3 of the large supply ship where highest transport efficiency and maximum utilization of the upper stage in space are of paramount importance. A compromise between these requirements whose effect on the design reveals strongly divergent and, in part, even opposite tendencies, cannot lead to a satisfactory solution, at least not until the technique of orbital systems is much further advanced than we justifiably may expect it to be within the next 15 or 20 years when the first orbital systems may be born.

A concept which emphasises a number of design features of orbital passenger ships is shown in Fig. 17. This orbital passenger ship consists of 3 stages:

a booster-type first stage
a flat-bottom second stage
a winged third stage.

The two last stages are arranged parallel to each other and are mounted on top of the first stage.

The first stage resembles that of the large supply ship, except for size and weight. It is smaller, because the second stage turned out to be considerably larger than its counterpart in the supply ship, owing to the particular dimeusions and shape of the passenger stage, which dominates the design of this vehicle. It was found, however, that the takeoff weight of the passenger ship can be kept about equal to that of the large supply ship so that the same first-stage rocket engine assembly could be used. Parachute recovery is assumed which should be simplified, compared to the large supply ship, due to the lower weight and lower cut-off velocity of this stage.

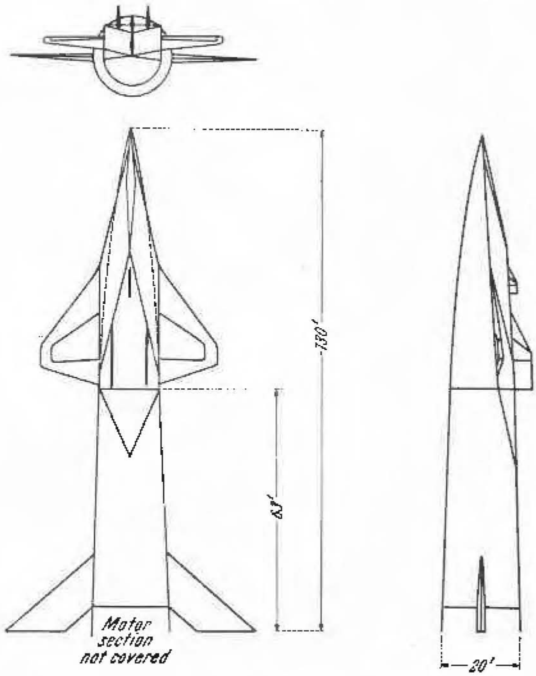

Fig. 17. Orbital vehicle for passenger transport.

The second stage is tailored for high mass ratio and expendable design, consisting essentially of propellant containers and rocket engine assembly.

The third stage is a hypersonic orbital rocket glider which, in the present concept, accomodates one pilot and 4 passengers, a total of 5 persons, corresponding to a payload weight of 1,200 lb, approximately. Table IV, representing a condensed version of the relevant portion of Table IV in [4], summarizes the basic data pertaining to the passenger ship.

Table IV. *Orbital Passenger Ship*

Stage	1	2	3
Payload (lb)	274,000	46,000	1,200
Dead weight (lb)	157,000	33,000	16,100
Effective propellant weight[1] (lb)	875,000	186,130	26,900
Auxiliary fluids (lb)	34,000	8,870	1,800
Gross weight (lb)	1,340,000	274,000	46,000
Momentum thrust per stage (lb)	1,741,000	410,000	49,800
Overall mass ratio[2]		32.4	

[1] Cf. Table II.
[2] Larger than in Table II, because upper stage returns.

2. *The Orbital Glider*

The most important part of the passenger ship is its final stage which is destined to carry human beings into and out of space. For this reason it must be designed with particular concern for its "payload".

The three principal design criteria for the orbital glider have been mentioned before, namely good hypersonic qualities, structural reliability and transport safety.

The returning glider passes through 3 regimes of flow, known as free-molecule flow, slip flow and continuum flow [14]. The free-molecule flow is characterized by the condition that the mean free path of the molecules is much larger than the body dimensions of the vehicle, meaning that collisions between molecules and body surface are much more frequent than collisions among molecules themselves. In free-molecule flow a boundary layer does not exist. The gas molecules strike the surface, remain in contact with it for an unknown period of time and then are re-emitted in some random direction with respect to their

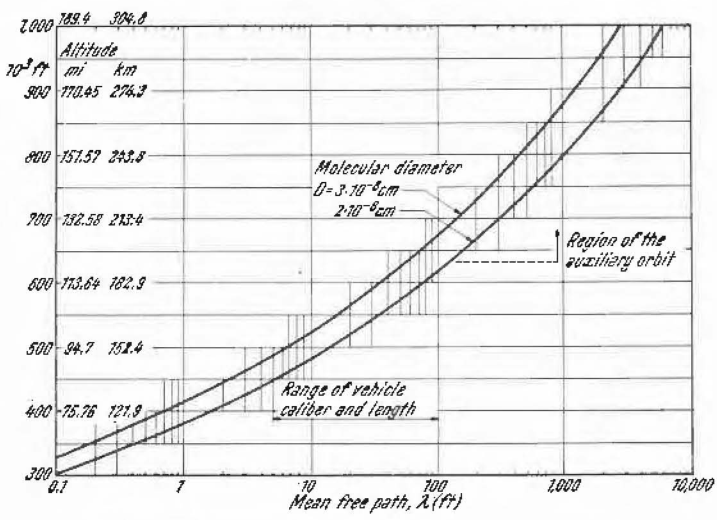

Fig. 18. Mean free path.

original direction of motion. As the air density is increased, a boundary layer begins to form; however this boundary layer is different from the conventional concept inasmuch as the air is not at rest with respect to the surface, as is usually the case at higher densities. This flow regime which is termed slip flow is encountered when $\lambda/\delta \sim 1$, i. e. when the mean free path of the molecules, λ, is of the order of the displacement thickness of the boundary layer δ. TSIEN [14] suggests that the slip flow may occur in the range $1 > \lambda/\delta > 0.01$. Between slip flow and free molecular flow $(1 < \lambda/\delta < 10)$ TSIEN assumes a transition region, which he calls realm of fluid mechanics, and in which intermolecular collisions are of equal importance to molecule-wall collisions. At values $\lambda/\delta < 0.01$ begins the rather well known regime of continuum flow.

Fig. 18 shows the mean free path as function of altitude for two molecular diameters (data taken from [15]). It can be seen that the region of free-molecule flow extends, approximately, from 450,000 ft upward. The slip flow domain lies roughly between 300,000 ft and perhaps 140,000 ft in local regions around the nose or at the leading edge of two-dimensional surfaces.

The regions of free-molecule flow, slip flow, and continuum flow separate two fundamental concepts: aerodynamics (continuum flow) and superaerodynamics (slip flow and free-molecule flow), the density (or mean free path) being the decisive parameter. On the other hand, air velocity breaks down the flow regimes into subsonic, supersonic, and hypersonic flow where hypersonic flow is defined as the velocity range in which the term $1/M^2$ (M = MACH number = ratio of flight velocity to sound velocity of the ambient air) becomes small enough to be neglected. Strictly, the linearized theory as a means for calculating the pressure distribution about a two-dimensional body fails already at very low supersonic MACH numbers ($M > 1.1$). BUSEMANN's third order theory is applicable up to MACH number 3 approximately. At higher MACH numbers the third order theory must be replaced by the hypersonic theory of the perfect gas which in turn, depending on the bonndary layer conditions, must be modified at very high MACH numbers to take gas imperfections (e. g. dissociation) into account. In the present discussion it is important to note that as far as the aerodynamic theory in general is concerned, the difference between MACH number 10 and 20 in, say, continuum flow, is less significant than the difference between MACH number 15 in continuum or in slip flow.

In the domain of free-molecule flow, the drag coefficient is very high [16], [17], [18], but the overall drag as well as the heat transfer by forced convection ([18] through [23]) are both negligible small owing to the extremely low density.

In general, it appears that the slip flow region can be broken down into two regions. One region, $0.01 < \lambda/\delta < 0.1$, directly following the continuum flow region, and a second region, $0.1 < \lambda/\delta < 1.0$, preceding the intermediate region. In the first mentioned region, shock-boundary layer interaction is dominant and may lead to a considerable increase in skin friction and heat transfer [24], while in the second region of even more rarified gas, the before-mentioned slip effect becomes dominant, causing a reduction in heat transfer. Thus, as the vehicle passes through the slip flow region, one may visualize the development of successive types of boundary layers. At the present time this makes theoretical analyses difficult and predictions uncertain. Little is known about the drag throughout the slip flow region and even less about the lift. Knowledge of the lift coefficient is important for computation of the sink velocity of the glider through the slip flow region.

Approximately, below a speed of 14,000 ft/sec and below 150,000 ft altitude most of the vehicle's surface is subject to continuum flow conditions. The shock wave drag becomes increasingly important. Surface conditions are now controlled by the conditions behind the shock wave and also by hypersonic shock wave — boundary layer interaction. The latter causes a thickening of the boundary layer at the leading portions, thereby changing significantly the geometrical shape of the leading contours. Qualitatively, the highest heat transfer rate, hence, the most critical region of aerodynamic heating can be expected in the upper region of continuum flow and the lower (denser) region of slip flow. Thereafter conditions gradually improve due to decreasing flight velocity. It is important to point out, that the above description is valid only for a given type of boundary layer, that is, laminar or turbulent. The necessity for maintaining laminar boundary layer conditions in continuum flow, whenever possible, is of great importance, because transition to turbulent boundary layer immediately increases the heat transfer coefficient considerably, especially when the flight speed in this region is still high.

These aerothermodynamic conditions, roughly outlined above for a better understanding of the problems, determine the design of the orbital glider.

The path of the glider is defined by the well-known condition of equality of aerodynamic lift and vehicle weight, W. At sufficiently high flight speeds the weight is reduced by an amount equal to the apparent centrifugal force F_c.

$$L = W - F_c = W - \frac{W v^2}{g r} = W - \frac{W v^2 r}{\gamma} \tag{13}$$

where $\gamma = g r^2$, g and r the gravitational acceleration and distance from the center of the earth, respectively, at flight altitude y, and the lift $L = C_L S \frac{\rho_{00}}{2} \sigma v^2$ with S being the lifting surface area and σ the density ratio ρ/ρ_{00}. Solving for σ which is a measnre for the instantaneous flight altitude, we obtain

$$\sigma = \frac{\rho}{\rho_{00}} = \frac{W}{S} \frac{2}{\gamma \rho_{00} C_L} \left(\frac{\gamma}{v^2} - r\right) = \frac{W}{S} \frac{2}{v_c^2 \rho_{00} C_L} \left[\left(\frac{v_c}{v}\right)^2 - 1\right] \tag{14}$$

where v_c is the local circular velocity and v the instantaneous flight velocity. This equation can be written in a more general form, thereby introducing an important glide parameter

$$\frac{\sigma C_L}{W/S} = \frac{2}{\gamma \rho_{00}} \left(\frac{\gamma}{v^2} - r\right) \approx \frac{2}{\gamma \rho_{00}} \left(\frac{\gamma}{v^2} - r_{00}\right). \tag{15}$$

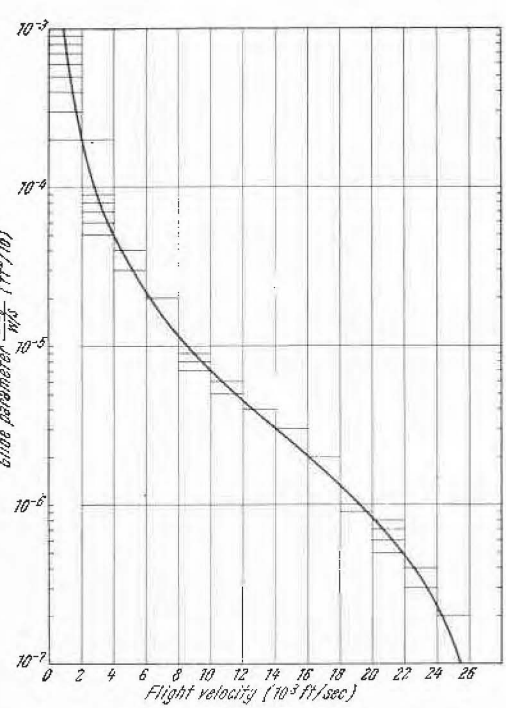

Fig. 19. Glide parameter as function of flight velocity.

The approximation $r = y \neq \neq r_{00} \approx r_{00}$ is justified, because, $y/r_{00} \ll 1$ in the altitude range where the aerodynamic lift becomes significant [say, at $(W - F_c)/W > 0.2$, i. e. $v < 23{,}000$ ft/sec, corresponding to a glide altitude of roughly $y < 250{,}000$ ft].

The generally valid glide parameter is plotted in Fig. 19 versus the flight speed (using $r \approx r_{00}$). Thus, at a given flight speed the glide altitude is proportional to the lift parameter C_L (W/S).

At the high flight MACH numbers in question, the stability of the boundary layer (i. e. not the transition, but the possibility of the development of a transition from laminar to turbulent flow) is no longer determined by the REYNOLDS number exclusively but, more important, by the MACH number just outside the bonndary layer and by the ratio of wall temperature to free stream temperature (i. e. the direction and intensity of heat flow). A high ratio of temperature just outside the boundary layer to wall temperature has a stabilizing effect on the boundary layer. However, the REYNOLDS number appears to remain the decisive parameter for the magnitude of friction and heat transfer. A general discussion of this situation is brought forward in a separate paper. With decreasing REYNOLDS number the heat transfer decreases in laminar as well as in turbulent boundary layer flow. Therefore, turbulence at very great altitude (low REYNOLDS number) is not necessarily more harmful than laminar heat transfer, if turbulence should occur at all. The laminarity requirement is most significant in the continuum flow regime.

The drag principally should be high, in order to limit the duration of descent, hence, of thermal stress on the structure. It is necessary, however, that the drag be produced in such a manner as to minimize the heat flux density into the vehicle in the conrse of the slow-down process. For instance, increasing the skin friction drag by producing a turbnlent boundary layer would definitely constitute a step in the wrong direction. By the same token, an increase in wave drag as form drag (not drag due to lift) is not necessarily beneficial. In continuum flow, the limit for the desirable wave drag is given by the maximum permissible rate of deceleration and by the fact that the higher pressure behind the shock wave produces a denser bonndary layer, thereby increasing the heat transfer coefficient. On the other hand the flow velocity behind the shock wave is lower and the temperature of the air is higher (by a considerable amount at hypersonic speeds), than before the shock wave. Higher temperature tends to increase the boundary stability, provided the wall temperature can be kept at the same valne in spite of higher heat inflnx due to higher pressure. If, in continnum flow, this increased stability is capable of preventing tnrbulence, the higher laminar heat transfer is the lesser evil. A more detailed discussion of all these counteracting effects is beyond the frame of this report and will be presented in a forthcoming paper.

As a result of the many considerations it follows that the orbital passenger glider preferably is small. It is characterized by very low lifting area load (the propellant containers are empty upon return into the atmosphere). Most of the large lifting area should be flat bottom body area. The advantages of flat-bottom bodies, namely reduced wing area and reduced heat influx into the wind-side (or lifting) body surface have been pointed out earlier by E. SAENGER [25]. Reduced wing area is possible, because at the high flight speeds, a flat plate has excellent lifting characteristics, even if it has a very low aspect ratio[1]. Thus the body is capable of carrying a major portion of its own weight, thereby reducing the bending moment on the wings and permitting a lower structural weight. With the body acting as an important lifting device, a comparatively smaller angle of attack becomes feasible for the same lifting force, the strength of the shock wave preceding the lifting surface is reduced, and consequently the heat flux density per unit surface area becomes lower for a given boundary layer condition (laminar or turbulent).

In the tentative concept of the orbital glider, shown in Fig. 20, a two-dimensional body is used which, in effect, represents a very low aspect ratio wing and which prodnces about 60 per cent of the hypersonic lift. The wing planform is of secondary importance in hypersonic flight and has no influence in free-molecule aerodynamics. A higher aspect ratio than the one offered by the body alone is desirable solely for reasons of low-speed flight and of landing (reduction of the angle of attack at touch-down). A high taper ratio[2] provides for high structural strength at the wing root and reduces the wing tip load. The comparatively large root chord at the same time reduces the sensitivity with respect to the excitation of wing oscillations by resisting wing twist, thereby cutting down angles of attack and wing load variations due to wing flutter. No attempts have been made to

[1] Aspect ratio is defined as the square of the wing span, divided by the wing area which is taken to include the part covered by the body.

[2] Taper ratio is defined as ratio of tip chord to root chord.

optimize the thickness ratio[1] of the wings of the prototype configuration shown in Fig. 20. The form drag (or wave drag at zero lift) increases strongly with increasing thickness ratio, however, the drag due to lift increases with decreasing thickness ratio. Restriction of form drag, therefore, limits the thickness ratio.

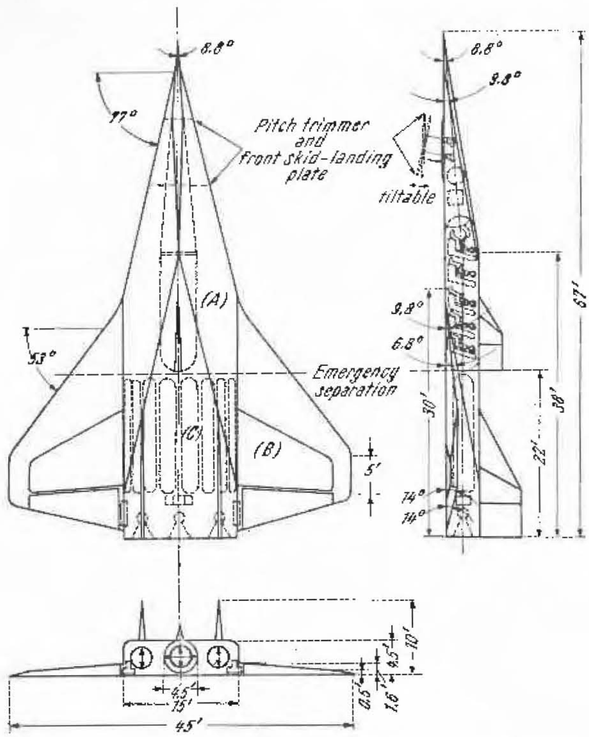

Fig. 20. Schematic sketch of 5-person orbital rocket glider.

Flat bottom area: Section	(A)	445 ft²
	(B)	540 ft²
	(C)	330 ft²
Total		1,315 ft²

Wing load: Full 35 lb/ft² (hypersonic)
Empty 11,1 lb/ft²

It is assumed that only 50% of the above area is effective as lifting surface at subsonic speed. Hence the wing load is doubled.

Moreover, since high lift force is required, and since high drag, as far as compatible with heat transfer considerations, is desirable also, the drag due to lift should be very high (i. e. the operational point on the polar of the orbital glider should be far beyond the point at which lift over drag is a maximum[2]). Thus a comparatively small thickness ratio can be anticipated offhand.

[1] Thickness ratio is defined as ratio of maximum thickness to length of chord of a given wing section in flight direction.

[2] Maximum lift over drag, a requirement for long range, is of no significance here. The angle of attack for maximum lift over drag is considerably smaller than the angle of attack for maximum lift. Given a certain minimum dynamic pressure (say, about 0,7 atm) and a suitable vehicle planform, aerodynamic controllability and stability can be maintained even at high angles of attack.

The three fundamental characteristic features of an orbital passenger ship are therefore:

low lifting area load

high ratio of body lift to wing lift

operation at angles of attack which exceed those required for maximum lift over drag.

Smallest possible vehicle size is called for by heat transfer considerations (reduction of surface area) and also by the second of the characteristic features enumerated above, since, as pointed out already by SÄNGER and BREDT [25], the ratio of body lift to wing lift decreases with increasing vehicle size.

In view of the flat bottom body configuration of the passenger stage, a parallel mounting of the two upper stages becomes an engineering necessity, because, otherwise, the lift of the flat bottom area which is large in comparison to the base area, would, during powered ascent and trajectory deflection, produce an intolerably high bending load on the joints between second and third stage. Another very important advantage of parallel mounting is, in this case, the rearward shifting of the center of pressure of the overall vehicle. This contributes greatly to improve the stability conditions and results in a saving of structural weight by requiring smaller fins on the first stage.

The resulting tentative glider configuration depicted in Fig. 20 is fairly slender, although, presumably, not slender enough for maximum range operation which, however, is of no concern in connection with the orbital passenger vehicle. Even so, it is not suitable for the stowage of bulky payload. The vehicle consists of a flat bottom body (*A*), (*C*) and a pair of wings of trapezoidal planform which is suggested by the desirability of high taper ratio and which also yields a comparatively high aspect ratio for a given wing area and taper ratio thus improving somewhat the low-speed flight characteristics of the glider.

The front portion (*A*) of the body contains the cone-cylinder unit for passenger accomodation. It includes all furnishings, power source, guidance and stability equipment, and is imbedded in the front portion of the body (forming so-to-say an obliquely oriented roof) for maximum heat protection of the passenger compartment. Most of the descent flight is a pure instrument operation with little need for visibility. As the vehicle approaches the surface of the earth, steel lids can be removed from windows and periscope lenses. The rear portion (*C*) of the body contains the power plant assembly with parallel arrangement of the units.

Safety can be taken into account by providing an emergency separation of the passenger-carrying section (*A*) from the rest of the vehicle. This appears to be the only method which permits a saving of the passengers under hypersonic, very-high-altitude flight conditions, and which, at the same time does not require an excessive increase in structural weight of this section. This separation which makes the front portion an independent airplane, provides protection primarily against failures in the power plant section (*B*). However, in this section the probability of failure is inherently largest, because it involves the greatest number of parts in operation during powered ascent. A serious structural failure in the front portion — during powered ascent or during the descent — would almost certainly result in death of the pilot and passengers. Complete safety is impossible in such a mission and utterly incompatible with the low-weight requirements dictated by our present energy sources. But then, it must again be emphasized that the smaller the vehicle, the less likely is a failure in the front sectional structure or surface construction. Thus safety considerations furnish another important argument in favor of small size of the orbital passenger vehicle. This condition can principally not be satisfied if the passenger carrying vehicle must, at the same time carry a substantial payload of cargo into the satellite orbit. Separation of cargo and passenger transport, therefore, is not only required from the viewpoint of cargo transport efficiency and economy, but also from the viewpoint of passenger transpost safety and reliability.

Another important design consideration is the pressure distribution over the upper surface of the glider. Strong, and above all, sudden, changes in static pressure are liable to produce violent crossflow, contaminating the boundary layer and producing premature transition from laminar to turbulent flow. The configuration Fig. 20 yields higher pressure over the wings than over the body. Inspection of the side view indicates that contamination of the body boundary layer is most likely to originate from the region where the leading edge of the wing is close to the body. In order to minimize the suddenness of the transition this portion has been smoothed out by a curved contour.

More details will be presented in a subsequent paper. For the present discussion a qualitative analysis may suffice.

3. *Flight Path*

Fundamentally, the powered ascent of the passenger vehicle is the same as that of the automatic supply ship. A certain increase in trajectory deflection is permissible within limits, because more lifting area is available to prevent excessive gravity deflection. The limits are given by the condition that the angle of attack thus required during powered ascent must not exceed a certain critical value, defined by the maximum permissible wing load of the passenger stage. In order to increase this angle as far as the complete vehicle is concerned, the third stage is assumed to be mounted at a negative angle with respect to the vehicle axis (Fig. 17).

Aside from this, no aerodynamic requirements exist for the winged stage during the powered ascent. For all practical purposes, in this period stage 3 is a rocket rather than a glider plane. This is emphasized by the fact that the vehicle has cleared the atmosphere before separation from stage 2 occurs. Although absence of a technically perceptible dynamic pressure is not as necessary a condition for separation as in the case of the large supply ship, it nevertheless simplifies the separation process of two parallel stages.

The passenger stage provides guidance and control for all stages during the ascent. These operations, as well as guidance and control of stage 3 proper (including attitude control in space) must be fully automatic. Therefore, only one pilot is needed who manages the supervision and monitoring of the servo system and maintains connection with surface station and space station. As to the return flight, the pilot will take over in the final portion of the descent when low supersonic speeds are reached. It then will require all the skill of a good pilot to bring the airplane safely to the ground, because he must accomplish a dead-stick landing with an airplane of poor low-speed qualities due to unavoidable compromises with hypersonic superaerodynamics.

Fig. 21 shows the velocity-altitude and velocity-temperature relation for a typical range of lift parameters $0.003 \leqq C_L/(W/S) \leqq 0.05$ ft²/lb.

The velocity altitude correlation is obtained easily from the generally valid glide parameter — defined in Eq. (15) and plotted in Fig. 19 — and from a density-altitude table, in the present case [15].

The correlation between velocity and skin equilibrium temperature follows directly from the equality of heat influx and radiation.

$$h_c(T_i - T_w) = \sigma\,\varepsilon\left[\left(\frac{T_w}{100}\right)^4 - \left(\frac{T_0}{100}\right)^4\right] \tag{16}$$

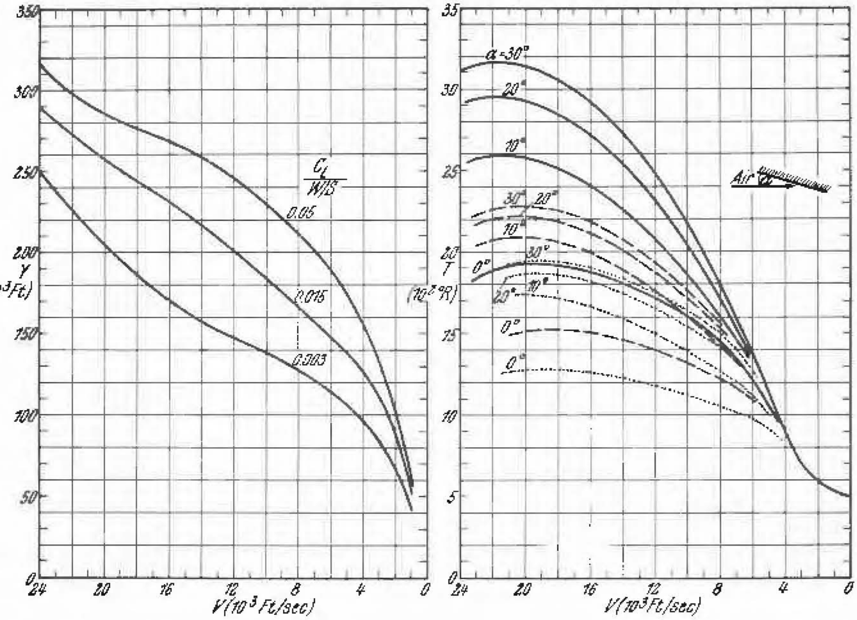

Fig. 21. Range in variation of altitude and equilibrium temperature of descending glider. Lift parameter: ——— 0,003, ... 0,015, — — — — 0,05.

where h_c is the convective heat transfer coefficient (Btu/ft²sec °R), T_i is the insulated wall (or boundary layer) temperature (°R), T_w is the actual wall temperature (°R), σ is the STEPHAN-BOLTZMANN constant (0.173 Btu/ft²hr (°R/100)⁴), ε is the emissivity of the wall (here taken as 0.9), and T_0 is the ambient temperature. Of these quantities, h_c is computed from

$$N_u = \frac{h_c\,x}{k} = 0.332\,(\mathrm{Re})^{1/2}\,(\mathrm{Pr})^{1/3} \tag{17}$$

where N_u is the NUSSELT number, k is the heat conductivity of air (Btu/ft²sec °R/ft), $\mathrm{Re} = \varrho\,x\,v/\mu$ is the REYNOLDS number with ϱ = density, x = distance from leading edge, v = velocity outside the boundary layer, μ = coefficient of viscosity, and $\mathrm{Pr} = \mu c_p/k$ is the PRANDTL number, c_p being the specific heat of the air at constant pressure. Eq. (17) is strictly correct only for incompressible flow, but it can be applied to compressible flow provided the air properties contained in N_u, Re, and Pr are calculated at a reference temperature which lies between T_i and T_w and is given in [26] as

$$T' = T_0\left(0.70 + 0.023\,M^2 + 0.58\,\frac{T_w}{T_0}\right). \tag{18}$$

The insulated wall temperature has been computed as isentropic stagnation temperature with variable specific heat and a recovery factor of 0.9, but without considering dissociation,

$$T_i = \eta \frac{v^2}{2\,g\,J \left[\bar{c}_p\right]_{T_0}^{T_i}} \tag{19}$$

where $J = 778.3$ ft-lb/Btu, η is the recovery factor and $\bar{c}_p$ is the mean specific heat between T_0 and T_i. The equilibrium temperatures are shown in Fig. 21 for $\alpha = 0$ degrees angle of attack and for $\alpha = 10^0$, 20^0 and 30^0. In calculating the latter values, a two-dimensional shock wave has been assumed and variation of the ratio of specific heats taken into account. In assuming a two-dimensional shock wave, the conditions outside the boundary layer are identical with those behind the shock. The temperature data represent those of an uncooled wall, 1 ft behind the leading edge, being in radiation equilibrium at the given inclination with respect to free stream direction, and at the given velocity-altitude correlation as specified by the lift parameter on the left side of Fig. 21. Particularly at altitudes above 200,000 ft the heating values become inaccurate because they fail to account for the slip effect. Moreover, it is not known whether the reference temperature method is reliable at MACH numbers above 10. However, the temperature curves show a trend which is probably correct.

It can be seen that a temperature maximum is reached roughly between 16,000 and 22,000 ft at the lift parameters considered, but that this maximum is lowered significantly with increasing lift parameter. It can also be seen that the skin temperature increases, in all cases, significantly with the angle of attitude. Now if a high angle of attitude is due not to high angle of attack but to bluntness of the wedge (high form drag), Fig. 21 shows that high temperature is obtained without the relief of higher glide altitude. On the other hand, if the high attitude angle is due to large angle of attack then the same drag may be obtained, not as form drag, but as drag due to lift. In this case, due to the higher lift coefficient inherently connected with a larger angle of attack in these flow regions, one obtains the benefit of higher glide altitude, hence, reduced heat influx in spite of higher attitude angle. It, therefore, is not irrelevant how the drag is produced (if the vehicle is the drag producer), and it should be understood that form drag is less desirable than drag due to lift.

Actually, a descending glider would not follow a given lift parameter curve because the lift coefficient C_L varies with altitude, velocity and angle of attack. It is reasonable to assume, however, that the actual lift parameter curve of the descending glider will lie between 0.003 and 0.015. Thus, in a small atmospheric layer, roughly between 250,000 and 150,000 ft, the flight velocity is cut down from about 20,000 ft/sec to 7,000 ft/sec, that is, about 13,000 ft/sec within 100,000 ft. Above and below this region, the deceleration is much less. For this range of lift parameters, therefore, the "critical region of descent" can be expected to coincide with, or lie inside of, the atmospheric layer between 220,000 and 150,000 ft (roughly 70 and 45 km) altitude.

VII. Concluding Remarks

The concept of orbital systems has been introduced in order to describe the integrated complex of orbital establishment and orbital supply. The orbital establishment is defined by the position and distance of its orbit as well as by its design and its function. Orbital supply — which may be regarded as a system in itself — is described by the supply vehicles and their functions.

In this paper an overall appraisal of orbital systems has been given, regarding the aspects of their development, the criteria governing the selection of their respective orbits, the orbital establishments, and the technique of supply.

Terrestrial and, partly, also orbital vehicle development is, in all countries concerned, directly or indirectly governed by military considerations on which the present accomplishments in aerodynamics and rocket flight are based. In view of this fact it is important to recognize that progress beyond the level of military intesest into the realm of astronautical endeavors depends decisively upon continuity of vehicle development and of level of effort in the field of orbital and inter-orbital projects.

This report indicates that, by a systematic analysis and by translation of the resulting conclusions into concepts of functional orbital systems, this required continuity can be preserved without sacrificing any of the bold aspirations of lunar and interplanetary research.

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