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Planetary Flight

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Aircraft and powerplant are proposed for flight missions in the atmospheres of several planets and moons. For flight missions on Venus, Earth, Mars, Jupiter, Saturn, and Titan, fairly complete conceptual designs are presented for vehicles ranging from 3 to 30 tons gross mass. The main virtues of these vehicles are that they are able to carry substantial payloads, provide long duration global coverage, loiter near or within phenomena of interest, and obtain in situ measurements. An important application would be planetology. A software model, Nuclear JET, was constructed to facilitate the conceptual design and optimization of each vehicle. The optimum designs were found to consist of subsonic airframes propelled by turbojet engines, using transonic blading in the compressor and impulse blading in the turbine, with heat provided by a direct cycle nuclear reactor. In the case of Mars, the optimum designs were found to consist of supersonic airframes, propelled by ramjet engines, with heat provided by a direct cycle nuclear reactor. A 3000-kg vehicle designed for the Jupiter atmosphere would have a 200-kg airframe, 200-kg gas turbine, 1200-kg reactor, and a 1400-kg payload. The most important results are 1) low mass vehicles with useful payloads can be built as light as about 1500 kg and 2) such vehicles can be designed without encroaching on even 1970s technical limits. Low mass planetary flight missions are feasible, affordable, and promise to be a rich source of new knowledge.

Nomenclature

D	=	diameter, mm
k_{eff}	=	effective neutron multiplication factor
k_{inf}	=	neutron multiplication factor for infinite reactor
L	=	length, mm
M_T	=	blade tip Mach number
P_f	=	fast neutron nonleakage probability
P_{th}	=	thermal neutron nonleakage probability
p	=	resonance escape probability
R	=	reactor radius, mm
r	=	degree of reaction for turbine rotor
r_{core}	=	reactor core radius, mm
r_H	=	rotor hub radius, mm
r_T	=	rotor tip radius, mm
T_{bulk}	=	fluid temperature in heat exchanger, K
T_{max}	=	highest material temperature in heat exchanger, K
T_{plate}	=	temperature in center of heat exchanger plates, K
T_{wall}	=	channel wall temperature in heat exchanger, K
Tt	=	stagnation temperature, K
t_{refl}	=	reflector thickness, mm
β_b	=	angle through which stator tip turns the flow, deg
β_c	=	angle through which rotor tip turns the flow, deg
ε	=	fast fission fraction
η	=	heat exchanger pressure ratio
π	=	pressure ratio
σ_a	=	allowable short-time stress, MPa
σ_{th}	=	thermal stress, MPa
τ	=	temperature ratio

Subscripts

0	=	ambient
1	=	diffusor inlet
2	=	compressor inlet
3	=	reactor inlet
4	=	turbine inlet
5	=	nozzle inlet
6	=	nozzle exit

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Introduction

THE weather and climate of other planets have fascinated researchers and writers for centuries. Recent planetary missions have greatly improved our knowledge and have led to new questions. Perhaps the most intriguing questions have been posed by the gas giants, as Jupiter, Saturn, Uranus, and Neptune are collectively known. Everyone is familiar with Jupiter's beautiful stripes and its mysterious red spot. There are even pictures of lightning storms on Jupiter. Similar phenomena are found on other gas giants. From the researcher's perspective, these colorful phenomena raise questions in the fields of chemistry, fluid dynamics, heat transfer, atmospherics, and climatology.

Although occultation, spectroscopy, and photography have helped to profile these atmospheres, some questions cannot be very well addressed by the fly-by missions performed so far. What is needed is in situ measurements, which include temperature, pressure, density, chemistry, particulates, and video images. Some such measurements were provided for Venus and Mars by packages that descended through the atmosphere and made soft landings. The upcoming Galileo mission will, among other things, drop a probe into Jupiter's atmosphere. This package will descend by parachute and for the first time yield a directly measured vertical profile of a gas giant's atmosphere.

Such probes are ambitious, and yield treasures of data and new questions. Most likely these new questions will only be answerable by obtaining more lateral coverage. This could be provided by dropping more probes. An alternative would be to design some kind of airborne research platform that could fly around to areas of interest, and collect, store, and transmit large quantities of data. For example, it might be useful to have a video tape of a flight from one Jovian stripe to another, or through the great red spot, or through a lightning storm, complemented by chemical, electrical, and atmospheric measurements.

This article presents conceptual designs for a class of vehicles capable of performing planetary flight missions.

General Reasoning

Because of the great expense of interplanetary transportation (e.g., $1 million U.S./kg), mass reduction naturally dominates the thinking of interplanetary mission designers. Due to the lack of oxygen in planetary atmospheres (Earth's excepted), it would seem infeasible to use combustion engines. For this reason, the author investigated nuclear flight. However, before launching into the details of modeling planetary flight, let us see what factors might give a designer cause for optimism or concern.

Reasons for Optimism

Planetary flight should be achievable for the following reasons:

1) Environment: Venus, Earth, Titan, and the gas giants have atmospheres that are dense enough to enable flight. (Mercury, Pluto, and most of the moons do not.) Mars is a marginal case.

2) Fuel: unlike air-breathing terrestrial aircraft where the fuel constitutes about one-third of the gross mass, nuclear fuel would be less than 1% of gross mass.

3) Structure: absence of the landing requirement obviates the landing gear, which constitutes about 7% of gross mass for terrestrial aircraft. Wing and fuselage structure can be significantly lightened, because the high cost of space transport justifies milling the wing and fuselage members out of slab stock, thereby eliminating most of the "nonoptimal mass" (e.g., joints, doublers, nontapered sheet, standard gauges, minimum gauges, fuel tanks, and cutouts), which together can more than double the structural mass. Cutouts are largely absent because there is no requirement for retractable landing gear, doors, windows, bomb bays, and field maintenance access hatches. Furthermore, there is no requirement for flooring and floor supports or for pressure bulkheads.

4) Gas turbine: since noise restrictions do not apply, all sound absorbing materials can be eliminated; and inlet guide vanes may be considered to ease the design of transonic compressor blading. The bird strike test can be dropped to lighten the compressor. Because the engines only need to function for about 500–1500 hr, less turbine temperature margin is needed. Since reactor power can be greatly increased without significantly increasing reactor mass, compressor and turbine efficiency won't dominate the design. For example, a lower turbine inlet temperature can be used to obviate turbine cooling.

5) Heat exchanger: because terrestrial radiation safety criteria do not apply, the atmosphere can be used directly as the reactor's coolant, thereby obviating weighty heat exchangers and pumps. While rocket reactors such as KIWI-A were designed for 5000 MW, and ramjet reactors such as TORY II were designed for 300 MW, planetary flight reactors only need to be designed for 3–30 MW. Also, since exit temperatures are far from material limits, the reactor can be designed for rapid power-up without exceeding allowable short-time thermal stress limits. Mass flow rates are low, and so the reactor inlet Mach numbers are also quite low. Thus, the pressure drops and structural loads are quite manageable.

6) Reactor neutronics: since the mass flow is small, the reactor void fraction can be kept quite small, which in turn avoids high fast neutron leakage. If BeO is chosen for the core material, and Be for the reflector, the reflector savings will nearly equal the reflector thickness. Therefore, the reflector can be made fairly thick, e.g., 33% of reactor radius, without significantly increasing the total reactor mass. This provides a fairly flat fission profile, which permits uniform radial heat exchanger performance with only modest radial variations in fuel loading density. Finally, it is possible to design these reactors so that most of the fissions result from thermal neutrons (i.e., fast fission factor less than e.g., 1.3). This gives the reactor an inherent stability as reactor criticality decreases with increasing core temperature, and since the void fraction is small, the reactor criticality will not vary much with density variations in the working fluid. Finally, if one chooses ${}_{92}U^{235}$, the reactor controls need no more than a 10-Hz response for the fine control of criticality.

7) Shielding: radiation shielding requirements make terrestrial nuclear flight technically too difficult. For planetary flight, the shielding could be dispensed with, provided that the payloads were designed with radiation hard materials.

Reasons for Concern

On the other hand, planetary flight may be difficult for the following reasons:

1) Environment: in the case of Jupiter, the gravity in the cloud-bearing atmosphere is three times that of the Earth's. This imposes a requirement for substantially more wing lift, airframe mass, and engine power, which cuts into the payload mass. In the case of Venus, the atmosphere consists mostly of CO_2, which being a rather nonideal gas, would increase the difficulty of compressor design. In the case of Mars, the atmosphere is so thin that flight with a subsonic airframe seems impossible.

2) Gas turbine: in the case of the gas giants, the atmosphere is mostly hydrogen and has a higher speed of sound than that found on Earth. Jupiter is the worst case with a speed factor of 3.5. The temperature of Saturn is lower, and so it has a speed factor of 2.5, whereas Uranus and Neptune are lower still. Therefore, the gas turbine must rotate faster by this factor to obtain roughly the same compression ratio per stage. This greatly increases the centrifugal forces and thus imposes a challenging strength of materials problem, particularly for the turbine. In fact, it seems too challenging, and so the gas turbine must be rotated at lower blade tip Mach numbers. Unfortunately, this means that many more stages are required to obtain the desired compression, which means more mass and complexity. Worse yet, due to the lower rotational Mach numbers, the loading per turbine stage must be reduced, otherwise the blading will have a very negative degree of reaction. This in turn means that the ratio of the number of compressor stages to turbine stages will decrease from, e.g., 4:1 to 2:1, hence, meaning more stages in the turbine as well as the compressor. In the case of Mars, it seems impossible to design a gas turbine at all. One problem is that the inlet diameter would have to be quite large in order to take in the required mass flow. But there is a much worse problem. A minimum Reynolds number is required in the compressor and turbine in order to turn the flow. But since the Mars atmosphere is so thin, the first stage of the compressor would need to have blading with chords about 100 times that of similar gas turbines on Earth, thereby making the turbomachinery very large and heavy.

Nuclear JET Model

Normally, an aircraft designer can make quick estimates based on experience with previous aircraft designed for a similar application. However, because planets have differing gravity and atmosphere, and as planetology is a completely new application for aircraft, the usual estimating techniques did not produce reasonable results. Therefore, the author went back to the fundamentals and set about writing a software that would support fairly complete conceptual design calculations and design tradeoff studies. In order to improve the reliability of the model, extensive use was made of previous work. Atmospheric data were obtained for Earth, Venus, Mars, Jupiter, Saturn, and Titan.[1-3] Data for Uranus and Neptune were not available. To be conservative, airframe mass estimations follow Shanley.[4] The gas turbine and ramjet designs draw primarily on Kerrebrock.[5] Materials data and flight reactor design principles are those of Bussard and DeLauer.[6] In order to validate the Nuclear JET (NJET) code, planetary flight vehicles were also designed for Earth, and compared with available textbook design problems and commercial engine data.

The NJET code consists of several thousand lines of C++ language code organized into five main parts. Each part contains several modules as described in Table 1. Although the

Table 1 NJET modules

1) Mission description
    - Materials data (gases, structural materials, heat exchange materials, neutronic materials)
    - Planetary data (atmosphere profiles, composition)
    - Mission description (planet, altitude, speed, gross mass, range)

2) Airframe design
    - Wing profile data (e.g., NACA 4415)
    - Wing geometry design (area, span, sweep angle, tapers, Reynolds number, lift, drag)
    - Wing mass estimate (material, load factor, nonoptimum mass factor, tension flange, sheer webs, ribs, secondary structure)
    - Fuselage geometry design (diameter, length, Reynolds number, drag)
    - Fuselage mass estimate (material, load factor, nonoptimum mass factor, skin, stiffeners)
    - Airframe mass

3a) Gas turbine design
    - Turbojet cycle
        - Cross section data (temperature, pressure, density, enthalpy, entropy, Mach number, channel flow parameters, channel area)
        - Component data (temperature ratio, compression ratio, efficiency)
    - Diffuser design (material, diameters, length, mass)
    - Compressor cascade design
        - Stage design (hub-to-tip ratio, flow swirl factors, velocity triangles, Mach triangles, stage temperature ratio, stage compression ratio, stage efficiency)
        - Blade design (diffusion factor, wake, shock, blade angles, blade chord, blade number, Reynolds number, temperature ratio, pressure ratio, efficiency)
        - Stage mass estimate (material, blades, outer rim, disk, inner rim, pressure hull, tension, bending)
    - Turbine cascade design (stage, blade, and mass calculations similar to those for compressor cascade
    - Nozzle (material, diameters, length, mass)
    - Turbojet mass estimation (component masses and lengths)

3b) Ramjet design
    - Ramjet cycle (cross section and component data similar to turbojet)
    - Diffuser design (material, diameters, length, mass)
    - Nozzle (material, diameters, length, mass)
    - Turbojet mass estimation (component masses and lengths)

4) Nuclear reactor design
    - Heat exchanger
        - Channel data (material, mass flow, thermal power, void fraction, length, channel gaps, plate thicknesses, total pressure drop)
        - Profile data (bulk, wall, and material temperatures; pressure; Mach, Reynolds, Prandtl, and recovery numbers; thermal stress, short-time allowable stress)
    - Neutronics
        - Core and reflector design (materials, dimensions, void fractions, geometric buckling, reflector savings, and masses)
        - Criticality design ($_{92}U^{235}$ loading, materials buckling, resonance escape factor, fast and thermal nonleakage factors, criticality, fast fission factor)

5) Payload
    - Payload mass estimate
    - Radiation dose rates (gamma, fast neutron)

computations are largely automated, there are several top level input parameters that the user can adjust to optimize the design: 1) mission: planet, altitude, Mach, range, vehicle gross mass; 2) airframe: wing profile, material, ratio of wing, and wing content mass to gross mass; 3a) gas turbine: material, compressor temperature ratio, ratio of blade hub-to-tip radius of first compressor stage, rotor blade tip Mach number of first compressor stage, turbine inlet temperature, turbine inlet vane swirl factor, nozzle type; 3b) ramjet: material, inlet pressure recovery, reactor outlet temperature, nozzle type; and 4) reactor: materials, reactor pressure ratio, inlet Mach number, void fraction.

The NJET code has a few built-in constraints to ease the search for good designs:

1) Gas turbine: the user chooses the compressor inlet's hub-to-tip ratio and blade tip Mach number, but then NJET will automatically reduce the blade tip Mach (and, hence, the rotation speed) if the tensile and bending stresses in the compressor blades are too great. In this case compressor stages will be added to make up the reduced stage compression ratio. The user chooses the turbine inlet swirl angle β_b, but the turbine radius will be automatically adjusted to give a degree of reaction around zero (impulse blading). The blade chords are chosen to insure a Reynolds number adequate to turn the flow (3×10^5 for compressor, 2×10^5 for turbine). However, to avoid unduly high stresses at rotor blade roots, rotor blade chords are not allowed to decrease as one moves from tip to hub.

2) Reactor: because the mission requires a simple quick power-up, gap and plate thicknesses are chosen so that the thermal stress is always kept below the allowable short-time stress ($\sigma_{th} < \sigma_a$). Because the code uses Fermi age theory to calculate the reactor criticality, the fast fission factor is limited to a range for which the theory is valid ($\varepsilon < 1.30$). The ratio of reflector thickness over core radius is given a lower bound for fission flattening reasons, and an upper bound to keep the mass down ($0.25 < t_{refl}/R < 0.5$). The user chooses the reactor inlet Mach number and a target pressure ratio, then NJET automatically hunts for fuel loading, reflector thicknesses, and void fractions that will obtain criticality within the above mentioned constraints. One can then make the tradeoff between reactor pressure drop (a strong function of inlet Mach) and reactor mass (a rather flat function of Mach except at higher power).

Results

Mission

The choice of altitude is of great interest for research. The altitudes shown below were chosen as follows. As the Earth vehicles serve primarily to validate NJET, the 11-km altitude was chosen since most commercial aircraft are optimized for that altitude. In the cases of Venus, Jupiter, and Saturn, cloud bearing altitudes were chosen, subject to the restriction that the atmospheric density and dynamic pressure would lead to reasonably straightforward searches for optimal vehicles. For Mars, the 4-km level was chosen because the atmosphere is thin. Note that this altitude only gives the vehicle access to about two-thirds of the Martian surface. In the case of Titan, there is not much atmospheric information, so that density and dynamic pressure considerations were used exclusively. Uranus and Neptune atmospheric data were unavailable to the author.

The Mars flight vehicle is different from the others in that the author chose a supersonic airframe with ramjet propulsion. This is because at supersonic speeds, there is sufficient dynamic pressure so that the wing area can be designed to customary proportions, and the turbomachinery can be dispensed with, thereby eliminating weight as well as an intractable design problem.

For subsonic airframes, the critical Mach number (e.g., 0.7) was chosen. For the supersonic vehicles, the highest possible Mach number used was limited only to the choice of structural materials. The alloy Ti-6242 is commonly used for high-temperature aerospace applications, and at about 540°C has material properties more or less similar to AL24S-T at room temperature (aside from the higher specific gravity), and can endure 1000 h before creep becomes significant (0.2% plastic strain). For the Mars application, this temperature corresponds roughly to Mach 4.0.

A load factor of 5.0 was chosen to permit maneuvers and flights through storms. A range of 1,000,000 km was chosen to give global coverage of Jupiter. Vehicles of 3, 10, and 30 tonne were designed for each of the missions exhibited in Table 2.

Airframe

For all the vehicles, a straight, tapered wing was chosen. Popular wing profiles and structural materials were chosen, as displayed in Table 3. For the subsonic airframes, NACA 4415 was used for the wing, and AL24S-T for the structure. For supersonic airframes a symmetrical biconvex wing was chosen (similar to that of the F-104 Starfighter), and Ti-6242 for the structure.

The area of the wing was calculated as that required to provide the lift equal to vehicle weight, for the angle of attack resulting in maximum lift-to-drag ratio. For NACA 4415, an angle of attack of about 2 deg gives a lift:drag better than 20:1. For the symmetrical biconvex 0006, an angle of attack of about 4 deg gives a lift:drag of about 6.5:1. Good streamline forms were chosen for both the subsonic and supersonic vehicles. The minimum allowed fuselage diameter was 1.2 m (1.5 for Mars), because the reactors have about 1 m diam. If the wingspan exceeded the minimum fuselage length, the length of the fuselage was arbitrarily increased to this value. The mass estimation summed up wing components (bending flanges, sheer webs, ribs, and secondary structure) and fuselage components (sheet, frames, and secondary structure). The results were multiplied by a nonoptimum mass factor of 1.5 in order to be conservative. The airframe size and mass estimates can be seen in Table 4.

Wing and fuselage mass depend strongly on the wing sweep angle. For a large aircraft, a sweep angle of, e.g., 35 deg would increase the airframe mass by about 50%, so there is an incentive to design a straight wing. However, for the designer who is trying to package the aircraft into a booster payload bay or into a suitable re-entry vehicle, the choices of wing sweep angle, wing aspect ratio, and the wing chord taper would be of great interest. Since the airframes shown above contribute about 10–20% of vehicle gross mass, the designer may be quite willing to budget more mass in order to use other geometries. An alternative would be to fly at lower altitudes, which would result in a shorter wingspan, much like a cruise missile.

Gas Turbine and Ramjet

Although the gas turbine turns out to be the lightest subsystem in our vehicle, it is also the most technically complex. In aircraft design, it usually takes longer to design the gas turbine than the rest of the aircraft, and so it must be given special attention. For the Mars flight vehicle we will use a ramjet.

Cycle Analysis—Gas Turbine

The gas turbine cycle analysis is dominated by two numbers: 1) the choice of turbine inlet temperature Tt_4 and 2) the choice of compressor temperature ratio Tt_3/Tt_2 (and, hence, compression ratio).

1) In order to obviate the need for cooled turbine technology, the turbine inlet temperature was chosen at a low value of 1300 K. Although cooled turbines can operate at, e.g., 1600 K and provide higher thermodynamic efficiency, this alternative was discarded in favor of technical simplicity because fuel economy is not the major concern for a flight reactor.

2) In order to provide a dense working fluid to the reactor (thereby obtaining low reactor inlet Mach numbers and low pressure losses while avoiding large core void fraction), the

Table 2 Mission parameters

Planet	Gravity, N/kg	Altitude, km	Mach	Velocity, m/s	Dynamic pressure, Pa	Load factor	Range, km
Earth	9.81	11	0.70	204	6,880	5.0	1,000,000
Venus	8.45	55	0.70	194	20,085	5.0	1,000,000
Mars	3.92	4	4.00	903	4,365	5.0	1,000,000
Titan	2.12	30	0.70	117	12,557	5.0	1,000,000
Jupiter	29.98	0[a]	0.70	702	35,143	5.0	1,000,000
Saturn	12.03	−30[b]	0.70	523	7,965	5.0	1,000,000

[a] h = 0 is defined as the p = 100,000 Pa level. [b] h = 0 is defined as the p = 10,000 Pa level.

Table 3 Airframe geometry and structure parameters

Structural material	Wing profile	Wing sweep angle, deg	Wing aspect ratio	Wing chord taper	Wing thickness taper	Fuselage streamline form	Nonoptimum mass factor
AL24S-T	NACA 4415	0	6.0	0.4	0.4	5:1	1.5
Ti-6242	CONV 0006	0	6.0	0.4	0.4	10:1	1.5

Table 4 Airframe size and mass estimates

Planet	Gross mass, tonne	Wing area, m^2	Wing span, m	Wing mass, kg	Fuselage D × L, m × m	Fuselage mass, kg	Airframe mass, kg	Airframe drag, N
Earth	3	9.5	7.6	120	1.5 × 7.6	40	160	2,580
	10	31.8	13.8	512	2.8 × 13.8	244	756	8,833
	30	95.5	23.9	2,085	4.8 × 23.9	1,267	3,352	27,289
Venus	3	2.8	4.1	40	1.2 × 6.0	25	65	3,562
	10	9.4	7.5	180	1.5 × 7.5	94	274	7,544
	30	28.2	13.0	764	2.6 × 13.0	491	1,255	23,274
Mars	3	3.0	4.2	42	1.5 × 15.0	63	105	2,607
	10	9.9	7.7	185	1.5 × 15.0	168	353	6,771
	30	29.7	13.3	784	1.5 × 15.0	435	1,219	18,666
Titan	3	1.1	2.6	12	1.2 × 6.0	9	21	1,950
	10	3.8	4.8	48	1.2 × 6.0	22	70	2,565
	30	11.3	8.2	181	1.6 × 8.2	84	265	5,891
Jupiter	3	5.7	5.9	130	1.2 × 6.0	75	205	8,421
	10	19.0	10.7	658	2.1 × 10.7	445	1,103	28,223
	30	57.0	18.5	3,074	3.7 × 18.5	2,314	5,388	87,754
Saturn	3	10.1	7.8	134	1.6 × 7.8	49	183	3,230
	10	33.7	14.2	586	2.8 × 14.2	299	884	11,106
	30	101.2	24.6	2,431	4.9 × 24.6	1,553	3,983	34,442

Table 5 Gas turbine thermodynamic cycle analysis

Planet	Units section	Ambient 0	Inlet 1	Compressor inlet 2	Reactor inlet 3	Turbine inlet 4	Nozzle inlet 5	Nozzle exit 6
Earth	Tt, K	236	236	236	591	1,300	986	986
	Pt, kPa	28	28	28	608	584	155	155
	Ht, kJ/kg	255	255	255	637	1,453	1,070	1,070
	S, J/kg·K	2,958	2,958	2,967	3,069	4,936	4,980	4,980
	M, –	0.70	0.70	0.57	<0.11[a]	0.50	0.50	1.97
Venus	Tt, K	335	335	335	837	1,300	985	985
	Pt, kPa	86	83	86	2,614	2,561	314	314
	Ht, kJ/kg	266	266	266	664	1,140	741	741
	S, J/kg·K	2,465	2,465	2,471	2,545	6,277	6,323	6,323
	M, –	0.70	0.70	0.57	<0.10[a]	0.50	0.50	1.76
Mars[b]	Tt, K	759	759	n.a.	759	n.a.	2,600	2,600
	Pt, kPa	71	71	n.a.	39	n.a.	32	32
	Ht, kJ/kg	554	554	n.a.	554	n.a.	2,556	2,556
	S, J/kg·K	2,728	2,728	n.a.	2,841	n.a.	9,412	9,412
	M, –	4.0	4.0	n.a.	0.10	n.a.	0.50	3.19
Titan	Tt, K	82	82	82	204	1,300	1,192	1,192
	Pt, kPa	52	52	50	1,056	1,045	695	695
	Ht, kJ/kg	83	83	83	208	1,400	1,275	1,275
	S, J/kg·K	1,504	1,504	1,512	1,608	1,406	4,518	4,518
	M, –	0.70	0.70	0.57	<0.06[a]	0.50	0.50	2.45
Jupiter	Tt, K	207	207	207	517	1,300	1,042	1,042
	Pt, kPa	140	140	135	2,044	1,942	722	722
	Ht, kJ/kg	2,558	2,558	2,558	6,396	17,048	13,210	13,210
	S, J/kg·K	21,570	21,570	21,684	22,850	52,257	52,673	52,673
	M, –	0.70	0.70	0.57	<0.08[a]	0.50	0.50	1.96
Saturn	Tt, K	107	107	107	268	1,300	1,169	1,169
	Pt, kPa	31	31	30	407	390	246	246
	Ht, kJ/kg	1,341	1,341	1,341	3,352	17,742	15,731	15,731
	S, J/kg·K	17,520	17,520	17,640	18,820	59,245	59,437	59,437
	M, –	0.70	0.70	0.57	<0.07[a]	0.50	0.50	2.22

[a]Adjusted for each vehicle to tradeoff reactor pressure loss against reactor core void fraction and reactor mass (see Table 9).
[b]Mars flight vehicles use ramjet engines, so that there is neither compressor nor turbine.

compressor temperature ratio is set at a slightly high value of 2.50. This is not always the best value from the thermodynamic point of view, however, fuel economy is not the major concern.

3) Efficiency parameters were chosen to be inlet 0.97, compressor 0.85, turbine 0.90, and nozzle 1.00. Detailed stage-by-stage analysis of compressors and turbines confirms the compressor and turbine estimates. The reactor efficiency (pressure ratio) was obtained from optimization iterations and was always at least 0.95.

These assumptions lead to the thermodynamic cycles shown in Table 5.

Cycle Analysis—Ramjet

The ramjet cycle analysis is dominated by two numbers: 1) the diffusor pressure recovery, and 2) the choice of reactor outlet temperature Tt_4.

1) In order to achieve good diffusor pressure recovery, a three-oblique shock diffusor was used to achieve a pressure recovery of about 0.55 at Mach 4.0.

2) Since there is no turbine we are free to choose a much higher reactor outlet temperature. In this case we are limited only by the heat exchanger performance and reactor material considerations.

Inlet and Nozzle

The mass flow was chosen so that the engine thrust would equal the airframe drag. For the subsonic vehicles, the inlet diameter was chosen so that the Mach number at the narrowest point would equal the flight Mach number, which is to say there is no external diffusion. For the supersonic vehicles, the diameter of the incoming stream tube was used. The nozzle diameter was chosen to match the exhaust pressure to the ambient pressure. The material chosen was steel with a thickness sufficient to contain the maximum engine pressure (with safety factor 3.0), as in the event of a compressor stall, the hot high-pressure gases in the reactor would escape forward through the compressor. The results of this approach are displayed in Table 6.

Compressor

The design of the compressor is dominated by the choices for the ratio of rotor hub to rotor tip r_H/r_T, M_T, β_c, the use of inlet guide vanes, and sometimes the strength of materials.

1) A high value for r_H/r_T results in a larger compressor diameter, higher mass, lower rate of rotation, lower blade stresses, and slightly higher efficiency. While stationary compressors tend to have large r_H/r_T as mass is not a consideration, flight compressors will have low values. For Earth, Venus, and Titan, 0.5 was chosen as a reasonable value. In the case of Jupiter, stress problems forced the value up.

2) The temperature rise per stage τ increases as the square of M_T. Hence, the largest possible value of M_T should be chosen. However, if the compressor blades move at supersonic speeds, there will be some shock losses. Generally, if the tips of the rotor blades are supersonic and the hubs are subsonic (transonic compressor), very good compression π with little shock loss is achievable. For Earth, Venus, and Titan, transonic compressors with M_T = 1.2 were proposed. Unfortunately, for Jupiter and Saturn, the speed-of-sound for the hydrogen atmosphere is about three times higher than for air, so that the same M_T would cause blade stresses nine times higher. This is impossible for the materials chosen (steel), and so lower values of M_T were chosen.

3) A high value for β_c results in highly loaded compressor stages (the blades have a high angle of attack), and consequently, fewer stages are required. The maximum possible value should be chosen without risking a compressor stall. A value of β_c = 35 deg was chosen as reasonable.

Table 6 Turbojet inlet and nozzle size and mass estimates

Planet	Gross mass, tonne	Mass flow, kg/s	Inlet[a,b] D × L, mm × mm	Nozzle[c] D × L, mm × mm	Mass, kg
Earth	3	3.44	254 × 92	198 × 396	1
	10	11.79	472 × 177	366 × 734	3
	30	36.43	828 × 312	644 × 1289	20
Venus	3	6.66	202 × 77	164 × 329	1
	10	14.21	296 × 112	242 × 483	4
	30	43.66	518 × 196	422 × 845	24
Mars	3	2.89	610 × 1782	1174 × 2347	4
	10	7.53	986 × 2875	1896 × 3791	15
	30	20.80	1638 × 4779	3156 × 6310	69
Titan	3	1.85	104 × 39	94 × 187	<1
	10	2.43	120 × 45	108 × 215	<1
	30	5.60	182 × 69	164 × 328	<1
Jupiter	3	3.03	196 × 73	162 × 324	11
	10	10.11	358 × 133	296 × 590	6
	30	31.45	632 × 235	520 × 1041	32
Saturn	3	0.91	194 × 72	178 × 357	<1
	10	3.13	362 × 134	330 × 662	1
	30	9.71	638 × 236	584 × 1169	8

[a]For subsonic vehicles, inlet diameter refers to the narrowest point near the leading edge.
[b]For Mars vehicles, inlet diameter refers to the incoming stream tube.
[c]Nozzle diameter refers to the exit diameter.

4) Inlet guide vanes (IGVs) are used to give the incoming air a usually solid body rotation, so that higher values of M_T can be used. In the compressors shown here, no IGVs were used. In the cases of Earth, Venus, and Titan, satisfactory results could be obtained without them. In the cases of Jupiter and Saturn, they were of no use because the compressors were entirely subsonic.

5) The blade and rotor stresses were always calculated and limited to 60% of yield stress. Only in the cases of Jupiter and Saturn compressors did the stress limit influence the basic design.

The above design choices yielded the compressor geometry and mass estimates shown in Table 7.

The lengths were derived from the blade chords, which in turn were chosen to meet the minimum Reynolds number required for turning the flow. The masses were calculated by summing mass estimates for blades, outer rims, disks, inner rims, and pressure hulls for rotors; and summing mass estimates for blades and pressure hulls for stators. To this a factor of 1.5 was multiplied for nonoptimal mass and engine accessories. All materials were assumed to be cobalt steel.

Turbine

The design of the turbine is dominated by the choices for turbine inlet Mach number, the rate of rotation (determined by the compressor design), the angle through which the tip of the first stator turns the flow β_b, the degree of reaction r, and the strength of materials.

1) The turbine inlet Mach number should be as high as possible without causing sonic flow blockage in the first stator, and 0.50 is reasonable here.

2) A high β_b results in highly loaded turbine stages, and consequently fewer turbine stages, thus 60–65 deg were chosen. However, for Jupiter and Saturn, the rotation speed of the turbine is much slower than the axial flow speed, and so the high loading would result in large, heavy, overstressed turbines. Consequently, β_b was reduced, and the lighter stage loading resulted in more stages being used to extract the required work from the flow.

3) A high degree of reaction results in a larger turbine radius, greater centrifugal forces, greater mass, higher blade temperatures, and higher turbine efficiencies. The degree of reaction chosen for commercial turbofans is usually about 0.50, since fuel efficiency is the major consideration. For planetary flight, there are incentives to achieve lower mass, stresses, and blade temperatures, therefore, impulse blading (r = 0.0) was chosen. Negative degree of reaction was not chosen because the stage efficiency falls significantly for degrees of reaction less than −0.50.

4) Tensile and bending stresses were calculated for the rotor blades, and tensile stresses were calculated for the disks and rims. The maximum allowed stress was 60% of the yield stress, and this stress limit dominated the rotor design for Jupiter and Saturn.

Using this approach, we arrive at the turbine geometry and mass estimates shown in Table 8.

Nuclear Reactor

Heat Exchanger

The reactor's thermal power is used to heat the working fluid. The design of the heat exchanger for flight reactors would normally be quite challenging, but the thermal power requirements listed in Table 9 are modest. The design of the heat exchanger is dominated by the following parameters: the thermal power (which is determined by the gas turbine cycle), the inlet Mach number, the reactor void fraction, channel gap dimension, and overall geometry.

1) The choice of reactor inlet Mach number has two important effects: the pressure loss in the heat exchanger increases with the square of the inlet Mach number, while the frontal area (hence, mass) of the heat exchanger varies in-

Table 7 Turbojet compressor geometry and mass estimates

Planet	Gross mass, tonne	r_H/r_T[a,b]	M_T[a,b]	r_T,[b] mm	β_c,[a] deg	τ[b]	π[b]	Stages[c]	Compressor D[b] × L,[c] mm × mm	Mass,[c] kg
Earth	3	0.50	1.20	158	35	1.16	1.65	11	316 × 603	25
	10	0.50	1.20	293	35	1.16	1.65	11	586 × 613	77
	30	0.50	1.20	514	35	1.16	1.65	11	1028 × 615	217
Venus	3	0.50	1.20	126	35	1.14	1.64	13	252 × 377	19
	10	0.50	1.20	183	35	1.14	1.64	13	366 × 378	33
	30	0.50	1.20	321	35	1.14	1.64	13	642 × 380	80
Titan	3	0.50	1.20	65	35	1.16	1.66	11	130 × 207	2
	10	0.50	1.20	75	35	1.16	1.66	11	150 × 207	3
	30	0.50	1.20	113	35	1.16	1.66	11	226 × 209	6
Jupiter	3	0.75	0.50	159	35	1.08	1.25	21	318 × 413	94
	10	0.75	0.45	291	35	1.07	1.23	23	582 × 465	137
	30	0.77	0.37	532	35	1.06	1.18	29	1064 × 604	209
Saturn	3	0.50	0.52	121	35	1.09	1.26	21	242 × 946	41
	10	0.50	0.50	224	35	1.08	1.24	22	448 × 1004	96
	30	0.50	0.50	395	35	1.08	1.24	22	790 × 1006	257

[a]Choices for r_H/r_T, M_T, and β_c largely determine the compressor design.
[b]r_H/r_T, M_T, r_T, τ, π, D values refer to the first stage of the compressor cascade.
[c]Stages, L and mass values refer to the entire compressor cascade.

Table 8 Turbojet turbine geometry and mass estimates

Planet	Gross mass, tonne	r_H/r_T[b]	M_T[b]	r_T,[a,b] mm	β_b,[a] deg	τ[b]	π[b]	r[b]	Stages[c]	Turbine D[b] × L,[c,d] mm × mm	Mass,[c,d] kg
Earth	3	0.93	0.42	128	60	0.90	0.67	0	3	256 × 113	3
	10	0.93	0.42	237	60	0.90	0.67	0	3	474 × 115	6
	30	0.93	0.42	416	60	0.90	0.67	0	3	832 × 115	16
Venus	3	0.93	0.39	78	60	0.95	0.72	0	6	156 × 141	5
	10	0.95	0.48	140	65	0.92	0.61	0	4	280 × 74	4
	30	0.95	0.48	245	65	0.92	0.61	0	4	490 × 73	8
Titan	3	0.95	0.40	86	60	0.91	0.68	0	2	172 × 72	1
	10	0.95	0.40	98	60	0.91	0.68	0	2	196 × 75	1
	30	0.95	0.40	149	60	0.91	0.68	0	2	298 × 76	2
Jupiter	3	0.94	0.17	134	35	0.98	0.94	0	14	268 × 379	93
	10	0.94	0.16	258	35	0.98	0.94	0	14	516 × 371	93
	30	0.98	0.23	814	45	0.97	0.88	0	7	1628 × 176	220
Saturn	3	0.76	0.11	89	25	0.99	0.97	0	15	178 × 1565	24
	10	0.79	0.11	174	25	0.99	0.97	0	15	348 × 1638	69
	30	0.79	0.11	307	25	0.99	0.97	0	16	614 × 1803	186

[a]Choices for β_b and r largely determine the turbine design.
[b]r_H/r_T, M_T, r_T, τ, π, r, and D values refer to the first stage of the turbine cascade.
[c]Stages, L, and mass values refer to the entire turbine cascade.
[d]The lengths and masses were derived in the same way as for the compressor cascade.

Table 9 Nuclear reactor geometry and performance estimates

Planet	Gross mass, tonne	Thermal power, MW	Inlet Mach	Length, mm	Void fraction	Gap, mm	Plate, mm	T_{max}, K	η	Load, kN
Earth	3	2.8	0.09	688	0.060	2.3	35.4	1555	0.96	12
	10	9.6	0.10	743	0.159	3.1	16.3	1817	0.96	13
	30	29.7	0.11	892	0.310	4.0	8.9	2276	0.96	18
Venus	3	3.2	0.05	680	0.049	1.7	32.3	1495	0.99	14
	10	6.8	0.09	686	0.057	1.4	22.4	1470	0.96	52
	30	20.8	0.10	738	0.136	2.0	12.9	1659	0.97	49
Mars	3	5.8	0.10	926	0.359	1.6	2.8	2768	0.81	5
	10	15.1	0.10	1217	0.542	1.4	1.2	2767	0.80	10
	30	41.7	0.10	1779	0.700	1.3	0.6	2765	0.78	23
Titan	3	2.2	0.02	697	0.046	1.7	35.9	1659	0.99	4
	10	2.9	0.03	687	0.042	1.4	32.6	1563	0.98	10
	30	6.7	0.05	693	0.057	1.4	22.4	1611	0.94	28
Jupiter	3	32.2	0.06	687	0.077	0.9	11.2	1380	0.95	42
	10	107.7	0.07	772	0.176	1.4	6.7	1476	0.96	47
	30	335.0	0.08	913	0.343	1.9	3.7	1682	0.96	61
Saturn	3	13.1	0.05	714	0.095	1.8	17.5	1494	0.96	8
	10	45.0	0.06	803	0.216	2.5	9.1	1748	0.96	11
	30	139.8	0.07	988	0.380	2.9	4.8	2113	0.95	19

versely with the inlet Mach number. Therefore, the author searched for the highest inlet Mach number that would still give an acceptable pressure ratio ($\eta = 0.95$).

2) The choice of reactor void fraction has two important effects: 1) the frontal area (hence, mass) of the heat exchanger varies inversely with the void fraction, 2) while the fast neutron nonleakage factor decreases exponentially with void fraction. Therefore, NJET iterates between the heat exchanger and neutronics routines to search for the least void fraction that would give a critical reactor of reasonable mass.

3) Plate and gap construction is used to maximize hydraulic diameter of the voids. The choice of cooling channel gap width has three important effects: the pressure drop decreases somewhat with increased gap width; while both the thermal stresses and material temperatures T_{max} increase with the thickness of the plates. NJET iterates to find the widest gap that would not cause the plates to either exceed short-time thermal stress limits or to approach the material melting point.

4) The heat exchanger is assumed to have the overall geometry of a right cylinder, and so the mass flow together with the inlet Mach and void fraction yields the heat exchanger dimensions. In the case of Mars, the air is thin, therefore, the heat exchanger must have large channel area, which leads to heavy reactors.

The heat exchanger profile for the Jupiter 3-tonne vehicle is shown in Fig. 1. The five curves portrayed are 1) fluid temperature T_{bulk}, 2) channel wall temperature T_{wall}, 3) temperature in the center of the solid plates T_{plate}, 4) fluid pressure P_{bulk}, and 5) fluid Mach number. Temperatures are well within material limits, the pressure loss is small, and the exit Mach number is only 0.10.

Normally, the structural design of a flight reactor is dominated by the axial force caused by pressure loss as the working fluid passes through the heat exchanger. However, the worst case shown in Table 9 (Jupiter, 30 tonne) the force is 61 kN, which is less than that due to the 5-g load factor of flight maneuvers. For the high-power engines, the plate and gap construction offers the advantages of higher hydraulic diameters, which allows a lower pressure drop for a given void fraction. Furthermore, it eases the application of fission flattening techniques, such as designing radially variations in uranium loading. But this geometry has the disadvantage of some structural complexity, since all the plates must be supported on edge by a frame that will increase both mass and neutron absorption.

However, for the low mass vehicles (other than the Mars vehicles), another design suggests itself. Because the void fractions are all quite low (less than 0.095), and pressure drops very small, we can use round holes for cooling channels. Furthermore, because the reflectors are fairly thick, and the core temperatures are all far from the material limits, we can consider dispensing with fission flattening techniques. Such a geometry is mechanically self-supporting and quite easy to fabricate.

The reflector will need some forced convection cooling, especially for the higher mass vehicles. The heat from the reflector is only, e.g., 3% of the core thermal power due to gammas and neutrons leaking from the core. For the 3-tonne vehicles it is only 0.1–1.0 MW. Rather than use regenerative cooling, the ram air pressure spilled from the inlet or first compressor stages will be enough.

Neutronics

The design of a reactor is dominated by the choices of materials and geometry. The material chosen for the core was BeO that was loaded with highly enriched uranium (92% ${}_{92}U^{235}$, 8% ${}_{92}U^{238}$), whereas the material for the reflector was chosen to be Be with 15% void for cooling channels. BeO and Be are good moderators, and allow the design of lighter, smaller reactors than graphite would. The "reflector savings" of Be is about equal to the thickness of the reflector. Highly enriched uranium has little ${}_{92}U^{238}$ (which only captures neutrons without producing fissions). The reactors were designed to operate primarily on thermal neutrons. The reactor is a right cylinder.

1) Absorbing materials (such as ${}_{92}U^{238}$) were minimized, in order to obtain a fairly high resonance escape probability ($p = 0.77$), which is the chance that a fast neutron will slow down to thermal energy without being absorbed. Thus, for the case of an infinite reactor, which has no leakage, the neutron multiplication factor would still be high ($k_{inf} = 1.99$), which means that 0.99 neutrons still must be allowed to leak away (or be absorbed by control rods).

2) Void fraction and reflector size were varied to obtain modest values for the fast neutron nonleakage probability ($P_f = 0.58$).

3) By choosing good moderating materials, the thermal neutron nonleakage probability P_{th} was kept high (0.91–0.92).

4) The core is surrounded on top and sides by a reflector. The top reflector thickness is chosen to be 15% of core length. The side reflector thickness, core void fraction, and uranium loading are chosen so that k_{eff} would be 1.05 for the reactor at operating temperature, which means that 0.05 neutrons/fission must still be absorbed by control rods. The fraction of uranium in the reactor thus works out to about 1–2 parts per thousand.

5) Fissions can be induced by fast or thermal neutrons. The fast fission factor ε, which is defined as the total number of fissions (fast and thermal) divided by the number of thermal fissions, is kept below 1.30. This is to insure that the reactors operate primarily on thermal neutrons, and so that the NJET calculations (which use the Fermi age model), are accurate.

6) The reactors are designed with $k_{eff} = 1.050$ for the hot reactor as the "all rods out" case. Control rods would be used

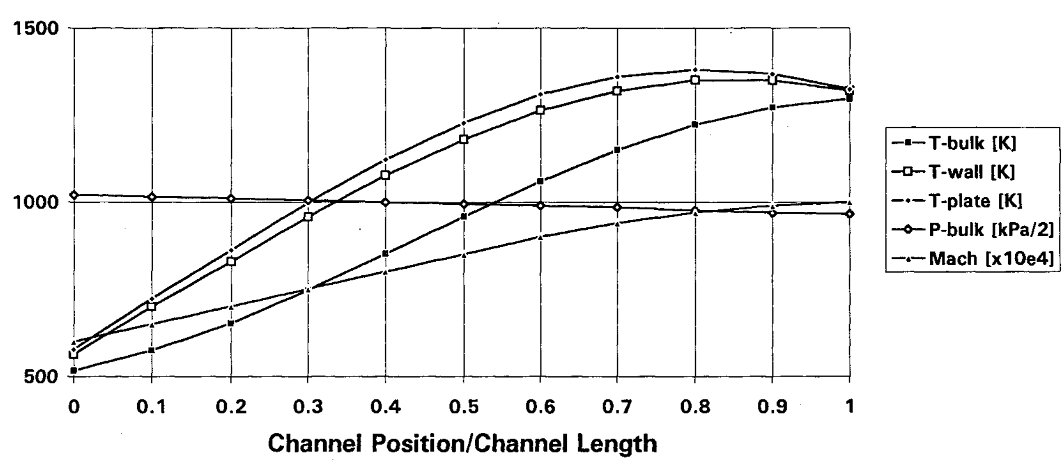

Fig. 1 Nuclear reactor heat exchanger profile for Jupiter 3-tonne vehicle.

to keep the reactor just critical during normal operation. Thus, we have more uranium than needed to go critical. As the uranium is fissioned away, the control rods would be gradually withdrawn to maintain criticality, until at the end of the fuel cycle, the reactor would be running with all rods out.

2) Many reactor properties are a function of temperature, so that k_{eff} (all rods out) is calculated for the cold reactor (T = 293 K), which is the reactor startup condition. As the criticality decreases with temperature, the reactor has an inherent stability against power fluctuations.

Following this method, the reactor criticality analysis will yield the values in Table 10.

The above core and reflector dimensions together with the void fractions result in the mass estimates shown in Table 11. Since $_{92}U^{235}$ has an energy density of 1.3 gm/MW day, the thermal power requirement and the range yields the mass of uranium fissioned.

If the fissioned $_{92}U^{235}$ is less than 10–20% of the uranium invested, then a fixed investment can be used, e.g., by dissolving uranium oxide in the BeO during reactor core fabrication. The 3-tonne vehicles have simple fuel cycles, each requires less than 7% of the fuel to be fissioned.

Payload

The payload is calculated as the gross vehicle mass less the airframe, gas turbine, and reactor masses and is shown in Table 12. A further figure of importance to the payload design is the radiation dose that it (and everything else) must be designed to withstand. The dose rates calculated in Table 12 assume that the payload is just 1 m from a point source, and that there is nothing to attenuate the radiation. For the low mass vehicles, the dose rates should not present very great problems.

Vehicle Layout

For each of the major subsystems, NJET computes the length and diameter. With this information, we can now lay out the vehicle. In Fig. 2 is a schematic for the Jupiter 3-tonne vehicle.

1) The reactor and gas turbine in the back together weigh 1400 kg, while the payload in the front weighs about 1400 kg. The wing is positioned midway down the fuselage. For lower mass vehicles, the wing would be smaller, and positioned further towards the rear.

2) For simplicity, the compressor is placed fore of the reactor, and the turbine aft. This means an extra meter of drive shaft is needed, but a lot of duct work is avoided. Some room is needed between the compressor and the reactor to match the flow areas, as well as between the reactor and the turbine.

3) The nozzle is placed aft most, with the inlet placed under the belly of the aircraft, F-16 style.

Table 10 Nuclear reactor criticality analysis

Planet	Gross mass, tonne	r_{core}, mm	t_{refl}, mm	U fraction, per mille	p	P_f	P_{th}	k_{inf}	k_{eff}, hot	k_{eff}, 293 K	ε
Earth	3	344	155	1.70	0.77	0.58	0.92	1.992	1.050	1.076	1.27
	10	371	184	1.41	0.78	0.58	0.91	1.996	1.050	1.075	1.25
	30	446	182	1.19	0.78	0.58	0.91	1.994	1.050	1.072	1.26
Venus	3	340	153	1.72	0.77	0.58	0.92	1.992	1.050	1.072	1.27
	10	343	154	1.59	0.78	0.58	0.91	1.996	1.050	1.075	1.25
	30	369	158	1.58	0.76	0.57	0.92	1.991	1.050	1.070	1.27
Mars	3	463	229	1.22	0.76	0.59	0.89	2.001	1.050	0.078	1.29
	10	608	273	0.81	0.77	0.59	0.89	2.006	1.050	1.069	1.27
	30	890	270	0.55	0.77	0.59	0.89	2.004	1.050	1.055	1.28
Titan	3	348	149	1.64	0.78	0.58	0.91	1.994	1.050	1.096	1.26
	10	343	154	1.77	0.76	0.57	0.92	1.990	1.050	1.091	1.28
	30	346	163	1.68	0.77	0.58	0.91	1.993	1.050	1.095	1.26
Jupiter	3	343	170	1.81	0.75	0.57	0.92	1.986	1.050	1.071	1.29
	10	386	165	1.46	0.77	0.58	0.91	1.993	1.050	1.077	1.26
	30	456	205	1.19	0.77	0.57	0.92	1.991	1.050	1.069	1.27
Saturn	3	357	168	1.56	0.78	0.58	0.91	1.994	1.050	1.091	1.26
	10	401	189	1.34	0.78	0.58	0.91	1.995	1.050	1.091	1.26
	30	494	201	1.08	0.77	0.58	0.91	1.994	1.050	1.090	1.26

Table 11 Nuclear reactor mass estimates

Planet	Gross mass, tonne	Uranium fissioned, kg	Uranium invested, kg	Core mass, kg	Reflector mass, kg	Mass, kg
Earth	3	0.21	8	658	510	1168
	10	0.71	8	739	709	1448
	30	2.19	12	1052	1006	2058
Venus	3	0.25	8	641	491	1131
	10	0.52	7	654	505	1159
	30	1.61	9	747	599	1346
Mars	3	0.10	14	1093	1375	2468
	10	0.25	21	1773	2818	4591
	30	0.69	45	3629	6031	9659
Titan	3	0.28	8	692	503	1196
	10	0.37	8	668	508	1176
	30	0.85	8	674	548	1222
Jupiter	3	0.69	9	643	562	1205
	10	2.31	10	813	684	1498
	30	7.18	13	1072	1189	2261
Saturn	3	0.38	8	708	600	1308
	10	1.30	10	870	850	1720
	30	4.02	15	1284	1367	2651

Table 12 Payload mass and radiation dose rate estimates

Planet	Gross mass, tonne	Airframe mass, kg	Turbojet mass, kg	Reactor mass, kg	Payload mass, kg	Gamma dose rate, Mrad/h	Fast-n dose rate, Mrad/h
Earth	3	160	29	1,168	1,643	23	6
	10	756	87	1,448	7,709	79	20
	30	3,352	253	2,058	24,336	243	62
Venus	3	65	25	1,131	1,778	26	7
	10	274	41	1,159	8,525	55	14
	30	1,255	112	1,346	27,287	170	43
Mars	3	105	4	2,468	423	47	12
	10	353	15	4,591	5,041	123	30
	30	1,219	69	9,659	19,052	340	84
Titan	3	21	3	1,196	1,780	18	5
	10	70	4	1,176	8,749	24	6
	30	265	8	1,222	28,505	54	14
Jupiter	3	205	188	1,205	1,403	263	68
	10	1,103	236	1,498	7,163	880	224
	30	5,388	461	2,261	21,890	2,736	699
Saturn	3	183	65	1,308	1,444	107	27
	10	884	166	1,720	7,229	367	94
	30	3,983	451	2,651	22,915	1,142	291

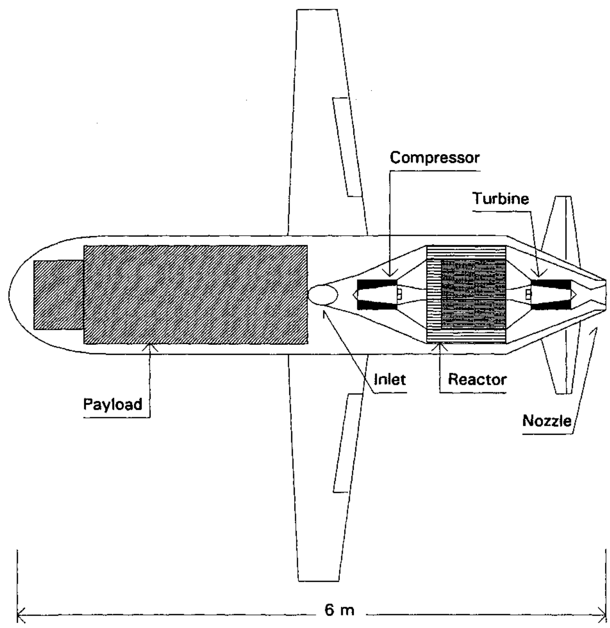

Fig. 2 Vehicle layout for Jupiter 3-tonne planetary flight mission.

4) A secondary flow of air is provided for cooling the reflector.

5) The payload occupies the entire fore of the fuselage. The most radiation sensitive payload should be placed in the nose. As the foremost payload is about 4 m distant from the reactor, the dose rate will be a factor 16 below the values shown in Table 12. The remaining interposed payload attenuates the radiation perhaps as much as another factor of 10.

A planetary flight vehicle must be packaged into an entry vehicle. It would seem reasonable to package it into an entry vehicle of geometry similar to the Space Shuttle, and assume a belly-first attitude during the hypersonic part of its entry path. Once the entry vehicle reached subsonic speeds, a drogue parashoot would be deployed to stabilize the vehicle, and the entry vehicle discarded using the usual method of exploding bolts and separation boosters, as is the case with planetary soft landing missions. The planetary flight vehicle would now be ready to perform a reactor and engine startup, discard the parashoot, and begin its flight. In the case of Mars, the situation is more complex, because the vehicle is intended for supersonic flight. Separation of the entry vehicle while still supersonic has its hazards, but should be achievable. Because there is not a lot of altitude to play with, these events should take place over the region of Hellas, which is a geological depression with a maximum depth of -6 km (10 mbar) compared to the 6.1-mbar level that is taken to be 0 km.

Conclusions

Technical Challenges

By constructing a fairly complete conceptual design, we can show that planetary flight missions are possible even on Jupiter, and furthermore, that the airframe, gas turbine, and reactor designs all fall within known 1960–70s engineering limits, and are conceptually simple.

1) The main challenge in airframe design would be the manufacture and assembly of very thin tapered sheet metal for the wings and fuselage of the lighter aircraft. This is especially so for the Mars airframes that are made of titanium, a somewhat difficult metal to work with. For flight in the cloud bearing regions of the colder planets and moons, deicing should be considered. For Mars flight, erosion of the leading edges and thin metal surfaces must be considered, due to the 900 m/s impact of dust particles.

2) The major challenge in the gas turbine design would be to push the material stress limits for Jupiter and Saturn applications. For Venus flight, the corrosive sulfuric atmosphere must be addressed.

3) The reactor design seems to be well within known material, thermal, and neutronic limits.

4) And finally, the main challenge for the payload designers would be to ensure that all materials and components used are radiation hard.

Low Mass Vehicles

The most encouraging results of this study come from looking at the low mass vehicles. The left side of Fig. 3 shows that the minimum vehicle masses all turn out to be under 1500 kg, with the exception of the Mars vehicles which have a minimum vehicle mass of about 2500 kg. As one increases the payload mass, the gross mass will increase faster than the payload mass because the vehicles must use heavier airframes. In the case of Mars vehicles, the reactor also gets larger and heavier. For this reason, the slope of the Mars curve is steepest. If a designer takes great care in minimizing the airframe drag of the Mars vehicle and improving the inlet pressure recovery, or chooses materials suitable for flight at higher Mach (say hypersonic), then the reactor can be lightened considerably. Alternatively, one may aspire to higher altitudes than the

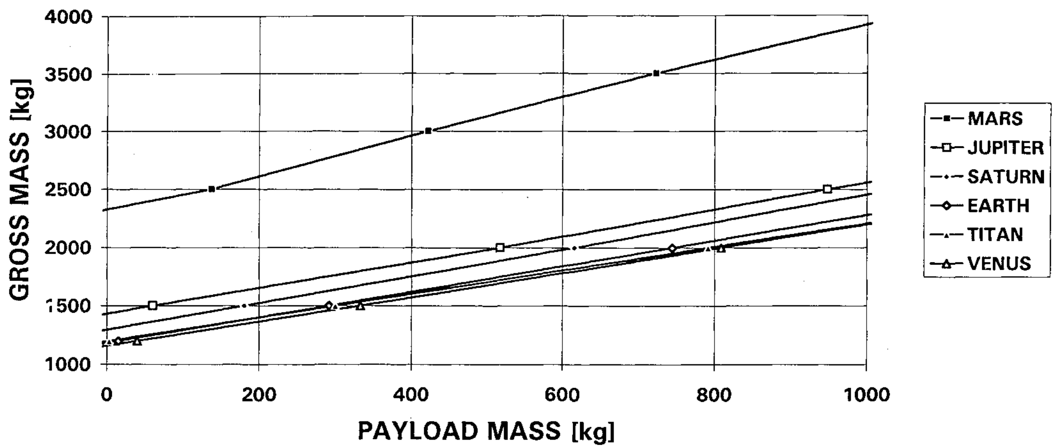

Fig. 3 Gross mass vs payload mass for low mass planetary flight missions.

modest 4 km chosen in this study, and thereby obtain wider coverage of the Martian surface. This would be worth considering, even though it may never be possible to design a vehicle capable of flight over the region of Tharsis, which contains peaks and ridges as high as 29 km.

In each case (except Mars), it seems that if the mission calls for a modest payload of about 300 kg, the vehicle gross mass would be about 1500–1800 kg. This vehicle must be packaged into a re-entry vehicle, which should not be too large or massive. One may wish to include in the mission a planetary orbiter that would be sent along both for scientific and communication relay purposes. All of this together would weigh in at about 3000 kg (4000 kg for Mars). We can see that this total mass does not greatly exceed that of some previous interplanetary mission vehicles.

Exhortation

This conceptual design study shows that planetary flight is achievable and affordable. Missions should be considered that take advantage of the vehicle's abilities to carry a substantial payload, provide long duration global coverage, fly to any point of interest, loiter near or within localized phenomena, and obtain diverse in situ measurements. The planetary flight vehicle is the key to greatly enhancing our knowledge of many planets and moons in our solar system.

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