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Engineering Notes

Flight in the Jovian Stratosphere: Engine Concept and Flight Altitude Determination

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I. Introduction

1 FROM the beginning of the space era, over 30 probes have been sent into the atmosphere or landed on the surface of the other planets in the solar system. However, most of them lacked the capability for sustained flight. So far, the only vehicles that have successfully reached and conducted continuous flight in the atmosphere of a planet other than the Earth have been the Vega-1 and Vega-2 balloons in 1985. They flew over 20,000 km into the atmosphere of Venus and collected valuable scientific information [1,2]. NASA's Perseverance rover or the planned Dragonfly mission [3] will use rotary aircraft. However, these will rely on electric motors and will have very limited range and speed.

When it comes to remote observation, satellites remain the most widespread means of exploring extraterrestrial atmospheres. However, the satellites fly high above the dense atmosphere. The most efficient way to probe and assess important physical parameters like pressure and temperature distribution, as well as altitude gradients, is to perform flight through different layers of the atmosphere. This can be achieved by flying the satellite into the thick lower layers toward the end of its mission or by designing an atmospheric entry spacecraft to collect data during its descent. Two notable examples for such spacecraft are the ESA's Huygens lander on Titan and NASA's Galileo probe, which uploaded valuable atmospheric data from Jupiter for almost an hour during its descent [4,5].

Continuous, powered flight can be conducted by a flyer (aircraft for other planets' atmospheres) that relies on classical airplane principles. To produce sufficient lift, relatively dense atmosphere and/or high airspeed are required. The engine will need to produce enough thrust to enable high-speed flight. The flight duration will obviously be limited to the amount of propellant carried on board or the mission will need to rely on external energy source, like solar power. However, the usage of electric engines supplied by solar panels is meaningful only near bodies close to the sun. For instance, the energy reaching the Jovian atmosphere consists of about 5% of the solar irradiation intensity at Earth's orbit [6]. Furthermore, only fraction of the solar energy reaches the lower layers, due to atmospheric scattering and other effects. For this reason, electric propulsion relying on solar panels is an unfeasible choice for high-speed flight on bodies with dense atmospheres or beyond the orbit of Mars.

An alternative approach would be to use heat from a nuclear fission reactor to produce thrust. The nuclear fuel has extremely high energy density that allows for months, if not years, of sustainable flight before the fuel is depleted. Unlike chemical combustion, the nuclear reaction does not rely on oxygen to produce heat. This enables flight in anaerobic atmospheres and without the need of carrying oxidizer. The engine can be designed as a ramjet, which relies on supersonic gas compression instead of turbo compressor to produce thrust, which has a number of advantages: it has few moving parts, which minimizes the risk of mechanical failure, and it is light. The latter is of paramount importance, given the capabilities of the launch vehicles and the cost to deliver every kilogram into orbit of other planets.§ Such design is called nuclear-powered ramjet engine (NPRE).

A nuclear heat engine has been tested on Earth within the U.S Military Project Pluto and has shown very promising results. The engine achieved 156,000 N thrust during tests [7]. The project envisaged the creation of the supersonic low-altitude missile (SLAM) nuclear-powered supersonic missile, capable of performing long-duration flight on complex trajectories. The project was closed in 1964, because the military favored the intercontinental ballistic missile approach. However, the results show that the technical challenges are manageable and NPRE is a viable flight propulsion option.

Research on nuclear-powered planetary flight was conducted by Miller [8]. The work suggests that a flight could be performed with a 3-tonne flyer at subsonic speed. The proposed flyer featured an engine with a turbo compressor, nuclear heat chamber, and a turbine. It relied on a classical turbo-jet principle and nuclear heat chamber to produce thrust. Considering the gas composition of Jupiter, a very high rotational speed and multiple compressor stages will be required to achieve sufficient compression. Such compressor will need to withstand higher mechanical loads and will have higher mass, compared with a turbo machine of similar performance on Earth. This will increase the total mission mass, cost, and the risk of mechanical failure. For these reasons, ramjet engines relying on supersonic compression can be more practical compared with subsonic turbo designs, especially for flight in hydrogen-rich atmospheres with high local speed of sound.

Another in-depth research on nuclear-powered flight was conducted by Maise et al. [9]. In their research, an atmospheric variation of miniature reactor engine (MITEE) nuclear rocket engine was proposed [9,10]. Some physical and geometric parameters of the engine were calculated, without conducting detailed design or evaluating possible flight altitudes. The proposed design features small payload and very high engine temperatures (1500 K), which would require a cooling system and very powerful reactor.

An NPRE-based flyer has its technical limitations. It requires dense atmosphere for sufficient thrust and lift to be generated. This makes the concept unsuitable for flight on Mars, for instance. In case of rocky bodies with dense atmospheres like Venus and Titan, there is the moral obstacle of using nuclear power for aerodynamic flight, because the flyer will ultimately crash into the surface and contaminate the local ecosystem with radioactive material. These considerations make gas giants a viable option for such a mission. They feature thick atmospheres with no hard surface and are particularly interesting for exploration, due to the presence of weather and different atmospheric phenomena.

Jupiter has several advantages making it suitable for such a mission over the other gas giants in the solar system: it is closer to Earth and

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§The mission cost of the Galileo mission was $1.5 billion [4]. Given the mass of the atmospheric probe and the satellite of about 3000 kg, the per-kg mission cost was approx. $500,000.

easier to reach; its atmospheric and wind conditions are less aggressive compared with Saturn, Uranus, and Neptune; and its atmospheric composition has been studied by the Galileo probe, which facilitates the flyer design.

In this paper, a mathematical model for the determination of possible flight altitudes in the Jovian atmosphere is introduced. The required thrust for steady flight at different altitudes of an idealized ramjet engine is calculated as a function of the temperature in the heat chamber. The maximum possible flyer mass is derived. The model provides the necessary input for the initial design process. After the initial design, the detailed construction of engines and flyers can be carried out, which is subject of future work.

In the second part of this paper, the atmospheric conditions in the flight area are described, and the boundary conditions for the calculation are defined. In the third part, the calculation method is described. The fourth part offers a simplified thermodynamic calculation of the required nuclear reactor power and heat chamber pressure. The aim is to show that the pressure levels are sufficient for sustained flight and that the required power levels are achievable by a compact reactor. The fifth part includes the conclusion and offers an insight into possible future work.

II. Gas Composition and Environment

An altitude or flight level on Jupiter can be defined as the elevation above the Jovian sea level (JSL) (the JSL is the isobaric surface of static pressure equal to the Earth's mean sea level pressure or 1013.25 hPa). Most suitable for aerodynamic flight are the lower part of the stratosphere and the upper part of the troposphere up to 60 km above JSL. The pressure and density in this range match the atmospheric conditions on Earth at 20–30 km altitude, which is within the service ceiling of typical supersonic aircraft. However, compared with Earth, a flyer will experience about 2.5 times higher gravitational acceleration, meaning that it will need to either produce 2.5 higher lift or be 2.5 less massive to achieve sustainable flight (gravitational acceleration at JSL is $g_{\mathrm{Jup}} = 24.79\ \mathrm{m/s^2} = 2.53 g_{\mathrm{Earth}}$) [5].

The absolute upper limit for the calculations was set at the altitude of 90 km above the JSL, because the density on that level ($\rho_{90\ \mathrm{km}} = 0.0018\ \mathrm{kg/m^3}$) is equivalent to the air density on Earth at 50 km
2 AMSL; 23.5 km was the chosen lower limit, because this is the lowest altitude for which the Galileo probe returned full set of data.

The gas in the range of interest consists of approximately 86.1% hydrogen and 13.6% helium by mass fraction. The composition does not change considerably between 23.5 and 90 km: the molecular weight and specific gas constant are constant, whereas the heat capacity ratio varies by $\pm 2\%$ from an average of 1.54. The speed of sound gradually decreases at lower altitudes, meaning that the flyer will produce more thrust at lower flight levels.

The haze layer upper limit is at around 50 km above JSL, and there is an ammonia cloud layer at up to around 30 km that contains ammonia ice particles. The lower edge of the ammonia cloud layer is at around 0 km. The flight must be conducted preferably above the cloud layers, to allow unobstructed visual observation [11].

III. Calculation Approach

The flight performance calculation is based on an idealized engine with 0.5 $\mathrm{m^2}$ cross section. The producible thrust and maximum allowable flyer mass are calculated as a function of the desired altitude and for different heat chamber temperatures. A schematic of an NPRE is shown in Fig. 1. The operating principle of the engine relies on supersonic compression. During flight, the atmospheric gas enters the engine through the supersonic inlet 2, where it is slowed down through series of supersonic shocks. The gas continues to diffuse in the diffusor 3 before entering the heat chamber, where it is heated up by the reactor to the design temperature 4. The hot, highly compressed gas passes the nozzle throat and is subsequently accelerated to high supersonic speed, producing thrust.

The thrust R produced by the NPRE can be calculated from the following relation [12]:

$$R = G_e v_e - G_1 v_1 + (p_e - p_1) A_e - X_\alpha \tag{1}$$

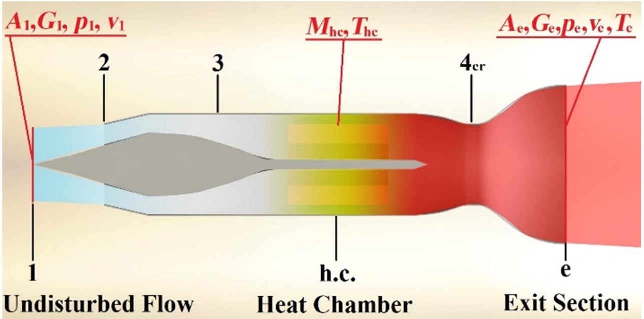

Fig. 1 NPRE sections: ~~1, undisturbed flow; 2, gas inlet; 3, diffusor; h.c., heat chamber; 4cr, critical nozzle section; e, nozzle outlet (exit section).~~ The notation for the relevant physical parameters for each section is ~~also~~ given. 3

You can delete the indicated text. It is already incorporated in chaper III Calculation Approach.

where G_1, p_1, v_1 and G_e, p_e, v_e are atmospheric gas mass flow rate, pressure, and velocity of the undisturbed flow (index "1") and at the nozzle outlet (index "e"), respectively, A_e is the nozzle outlet surface area and X_α is an additional drag term.

In an ideally designed inlet, the geometry is optimized in a way to allow the oblique shock waves originate from a common point on the front edge of the inlet. The ideal diffusion angles that lead to such shock structure can be determined experimentally in a wind tunnel or by performing computational fluid dynamics (CFD) simulations. Under these conditions, the X_α term will be close to zero.

The second term $(p_e - p_1) A_e$ is the pressure thrust term. For a nozzle with finite size, the jet flow will not be ideally expanded; i.e., it will leave the engine at higher than the ambient pressure, producing thrust. In an ideal engine the gases leave the domain fully expanded. In this case the pressure thrust term is fully converted into momentum thrust and the engine efficiency is maximized.

The first $G_e v_e - G_1 v_1$ term is the momentum thrust term. This term can be maximized by introducing a nozzle with suitable expansion ratio to allow for full expansion of the gases leaving the engine.

For this calculation, an idealized highly efficient NPRE is assumed and both X_α and the pressure thrust term are considered negligible. For a noncombusting engine in stable operation the gas mass flow rate leaving the engine equals the inlet flow rate, i.e., $G_e = G_1$. The thrust relation will then take the following form:

$$R = G_1 (v_e - v_1) \tag{2}$$

The undisturbed flow speed is determined by the flight Mach number M_1 and the speed of sound of the environment (c), which can be calculated from the Galileo probe data:

$$v_1 = c M_1 \tag{3}$$

High supersonic speed is required for good compression. At flight Mach number $M_1 = 3.0$ the free flow speed at an altitude of 60 km is 2469 m/s.

The mass flow rate depends on the engine inlet area A_1:

$$G_1 = A_1 \rho_1 v_1 \tag{4}$$

The required cross-sectional area will depend on the shock wave structure and gas compression efficiency. For the current idealized estimate, A_1 is assumed to be 0.5 $\mathrm{m^2}$. The density ρ_1 is available from the Galileo probe data.

The speed of the exhaust flow v_e can be determined by the following expression:

$$v_e = M_e \sqrt{\gamma R_j T_e} \tag{5}$$

The exhaust flow speed needs to be higher than the flight speed. However, higher M_e will require larger and heavier nozzle. The exhaust Mach number is assumed $M_e = 3.2$, which provides good balance between thrust and resulting mass. The heat capacity ratio γ and specific gas constant R_j are available from the Galileo data.

The exhaust temperature T_e is a function of the heat chamber temperature T_{hc} and the ratio of heat chamber to exhaust flow Mach number, i.e., $T_e \sim f(M_{hc}/M_e)$. Because the NPRE is a breathing engine, the gas flows through the heat chamber at nonzero speed M_{hc}. The heat chamber Mach number will depend on the geometry and resulting shock structure and needs to be estimated. Obviously, for a ramjet, M_{hc} will be between 0 and 1. In this analysis, M_{hc} is assumed to be 0.5. Because the heat chamber and the nozzle are considered ideal, the expansion in the nozzle can be assumed to be an isentropic process. The exhaust temperature can be calculated from the relation [12,13]

$$T_e = T_{hc} \frac{1 + (\gamma - 1/2)M_{hc}^2}{1 + (\gamma - 1/2)M_e^2} \tag{6}$$

With Eqs. (1–6), the thrust can be calculated for different altitudes above JSL. Table 1 shows the results for altitudes between 40 and 90 km and at four different heat chamber temperatures $T_{hc} = 600$, 900, 1200, and 1500 K. The thrust as a function of altitude is shown in Fig. 2.

Because during steady flight the thrust equals the drag, the lift can be calculated from the thrust if the lift-to-drag (L/D) ratio is known. The maximum L/D ratio $k_{\max}$ can be calculated from the Kuchemann's relationship [14]:

$$k_{\max} = \left(\frac{L}{D}\right)_{\max} = \frac{4(M+3)}{M} \tag{7}$$

where M is the flight Mach number. For $M = 3$, $k_{\max} = 8$. However, this is an idealized empirical calculation valid on Earth. The exact L/D ratio will depend on many factors (flyer geometry and configuration, angle of attack, etc.) and needs to be determined during detailed design by means of experiment or CFD simulations.

For this analysis, the L/D ratio will be assumed to be $k = 2.5$. This is considered a conservative estimate with the real value expected to be higher.

The lift is calculated as follows:

$$L = kR = 2.5R \quad (R = D) \tag{8}$$

The lift as a function of altitude at different heat chamber temperatures is shown in Fig. 2.

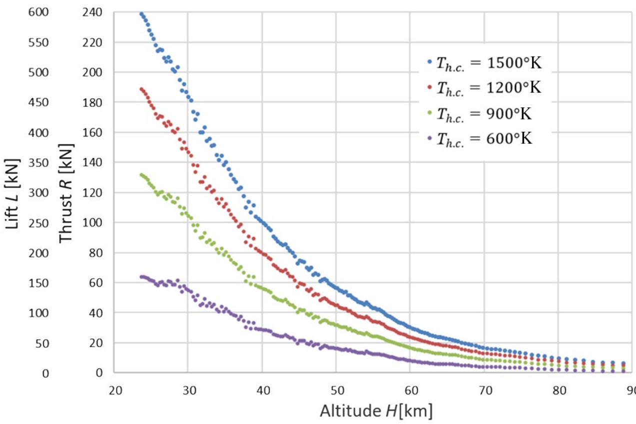

Fig. 2 Resulting lift (L) and thrust (R) as a function of altitude for different heat chamber temperatures ($T_{h.c.}$).

To determine the maximum allowable flyer mass, the weight needs to be calculated. The gravitational acceleration changes insignificantly within the altitude range of interest and is assumed constant at 23.2 m/s^2. The resulting flyer masses for different temperatures $T_{h.c.} = 1500$, 1200, 900, and 600 K and at different altitudes are shown in Fig. 3.

Note that the result shown in Figs. 2 and 3 is scattered. The calculated distributions are based on the physical parameters at discrete time points as measured by the Galileo probe. After the input data have been carefully analyzed, it has been determined that the density data exhibit deviations, which result in the observed scatter. Instead of measuring the density directly, Galileo relies on a reading from the accelerometer unit of the Atmosphere Structure Instrument (ASI) to evaluate the local density. This method is described by Seiff and Knight [15] and has been used successfully on the Earth, Mars, and Venus. However, atmospheric turbulence and other effects, like gusts of wind, can affect the accelerometer output, which causes the scattering in the density data.

The distributions show that heavier flyers will require a higher heat chamber temperature for horizontal flight at given altitude. Table 2 shows a summary of the altitudes for steady flight for 1000 and 2000 kg flyers at different heat chamber temperatures.

The flight altitudes are graphically shown in Fig. 4. It is evident that a flyer with 1000 kg mass and reactor capable of producing 600 K in

Table 1 Flight parameters and resulting thrust at different altitudes and for varying heat chamber temperatures

Galileo probe data							
Altitude above JSL, km	Pressure, Pa	Density, kg/m^3	Engine mass flow rate, G_1, kg/s	Undisturbed flow speed, v_1, m/s	Exit temperature, T_e, K	Exit velocity, v_e, m/s	Thrust, R, N
Heat chamber temperature–$T_{hc} = 600$ K							
90.4	1,058	0.002	2.6	2,774.9	178.6	3,143.0	947.3
80.1	1,632	0.003	4.2	2,655.4	176.3	3,133.9	1,999.9
60.0	4,374	0.010	12.2	2,468.8	171.4	3,113.7	7,893.4
40.2	13,450	0.033	39.4	2,380.2	168.7	3,101.7	28,406.7
Heat chamber temperature–$T_{hc} = 900$ K							
90.4	1,058	0.002	2.6	2,774.9	267.9	3,849.3	2,765.3
80.1	1,632	0.003	4.2	2,655.4	264.4	3,838.2	4,943.8
60.0	4,374	0.010	12.2	2,468.8	257.1	3,813.5	16,458.0
40.2	13,450	0.033	39.4	2,380.2	253.0	3,798.8	55,849.9
Heat chamber temperature–$T_{hc} = 1,200$ K							
90.4	1,058	0.002	2.6	2,774.9	357.2	4,444.8	4,297.9
80.1	1,632	0.003	4.2	2,655.4	352.6	4,432.0	7,425.6
60.0	4,374	0.010	12.2	2,468.8	342.8	4,403.4	23,678.3
40.2	13,450	0.033	39.4	2,380.2	337.3	4,386.5	78,985.6
Heat chamber temperature–$T_{hc} = 1,500$ K							
90.4	1,058	0.002	2.6	2,774.9	446.5	4,969.4	5,648.1
80.1	1,632	0.003	4.2	2,655.4	440.7	4,955.2	9,612.1
60.0	4,374	0.010	12.2	2,468.8	428.5	4,923.2	30,039.6
40.2	13,450	0.033	39.4	2,380.2	421.7	4,904.3	99,368.5

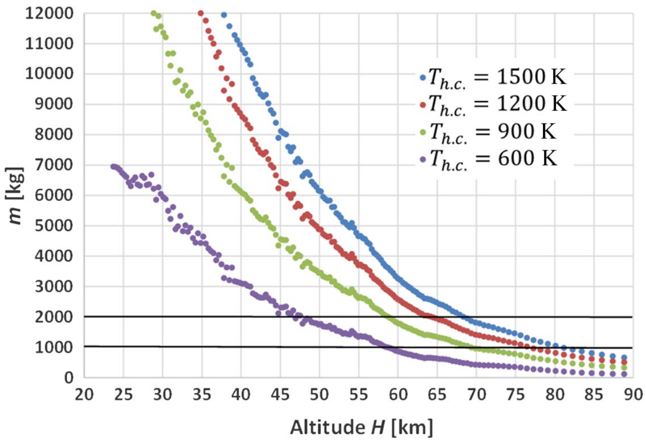

Fig. 3 Maximum allowable mass for horizontal flight as a function of altitude and for different heat chamber temperatures. ~~The horizontal lines represent two hypothetical flyers with 1000 and 2000 kg mass.~~

Table 2 Summary of the altitudes for steady flight for 1000 and 2000 kg flyers at different heat chamber temperatures

T_{hc}, K	Altitude of steady flight with flyer mass of 1000 kg, km	Altitude of steady flight with flyer mass of 2000 kg, km
1500	81	68
1200	77	64
900	69	58
600	58	47

the heat chamber will be capable of flying at 58 km and above the haze layer.

IV. Simplified Heat Chamber Pressure and Reactor Power Calculation

An approximate calculation of the heat chamber pressure and inlet temperature is conducted in this section. With the pressure and temperature known, the required reactor power and approximate thrust can be estimated. Estimating these parameters will provide evidence that the concept is feasible, and that the flyer is capable of sustained flight under the given environmental conditions.

To calculate the required reactor power and resulting thrust, the gas parameters across the heat chamber need to be known. The pressure at the heat chamber inlet can be determined if the pressure losses across the shock system that emerges around the inlet are known. A measure for these losses is the pressure recovery coefficientσ. Figure 5 is showing a diagram of the relation of the pressure recovery as a

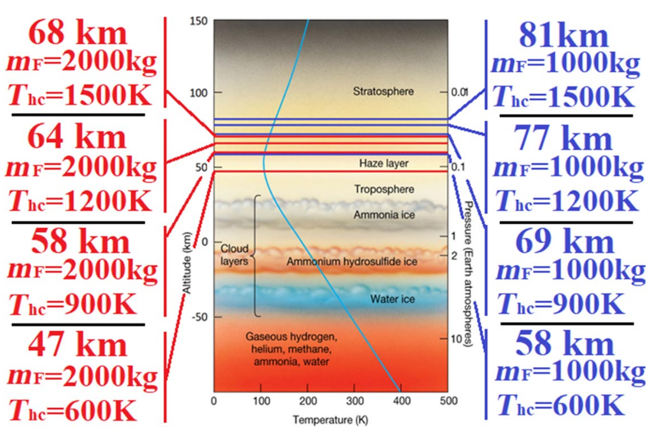

Fig. 4 Flight altitudes for 1000 and 2000 kg flyers and at different heat chamber temperatures. Background picture source: https://www.uccs.edu/; Pearson Education, Inc.

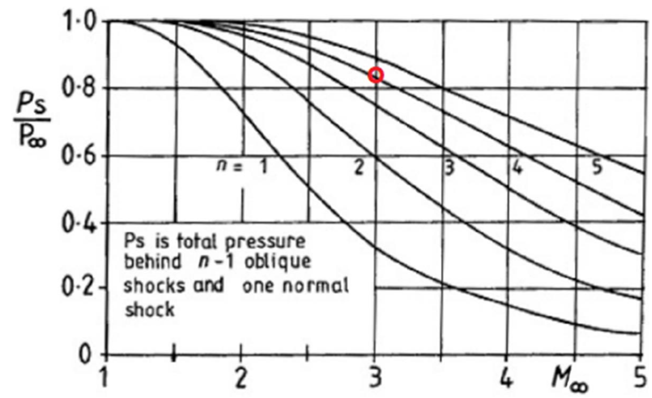

Fig. 5 Shock pressure recovery for freestream Mach number and number of oblique shocks [16].

function of the freestream Mach number and the number of oblique shocks of the inlet of airbreathing engines.

For a supersonic inlet with three oblique and one normal shock at flight Mach number $M = 3.0$ the pressure recovery is around 0.85.

After the normal shock, the gas flows through the subsonic diffusor before entering the heat chamber. The subsonic diffusion losses can be assumed to be 2% [17]. The total pressure recovery of the diffusion between the freestream and the heat chamber inlet is then

$$\sigma = \sigma_1 \sigma_1 = 0.833 \quad \text{with } \sigma_1 = 0.85; \quad \sigma_2 = 0.98 \tag{9}$$

Because the gas diffusion is not calculated explicitly, the heat chamber Mach number needs to be assumed. For consistency, the heat chamber Mach number is assumed to be $M_{hc} = 0.5$, as in the previous section. The following calculation considers the ambient conditions at 60 km altitude and 600 K reactor/thrust chamber outlet temperature. The heat chamber gas parameters for these conditions are [12,13]

$$p_{hc} = \sigma . p_1 \left(\frac{1 + (\gamma - 1/2) M_{hc}^2}{1 + (\gamma - 1/2) M_1^2} \right)^{\frac{-\gamma}{\gamma - 1}} \quad p_{hc} = 102\text{kPa}; \quad \frac{p_{hc}}{p_1} = 23.3 \tag{10}$$

$$T_{hc} = T_1 \left(\frac{1 + (\gamma - 1/2) M_{hc}^2}{1 + (\gamma - 1/2) M_1^2} \right)^{-1} \quad T_{hc} = 391 \text{ K} \tag{11}$$

The stagnation temperature is the maximum temperature that can be achieved by diffusion without adding heat and is calculated as follows:

$$T_s = T_1 (1 + (\gamma - 1/2) M_1^2) \quad T_s = 417 \text{ K} \tag{12}$$

Temperature [Eq. (11)] is just 26 K lower than the stagnation temperature [Eq. (12)]. This shows that the flow temperature does not vary considerably within the possible M_{hc} range. For this reason, the reactor power and thrust calculation based on assumed $M_{hc} = 0.5$ are expected to be accurate.

Because the heat capacity ratio does not change significantly within the altitude range of interest, the result [Eq. (12)] also shows that the temperature at the heat chamber inlet can be expected to be around 420 K. This is an important result for the reactor and structural design.

The pressure rise due to supersonic and subsonic diffusion is calculated [Eq. (10)] to be 23.3, which is comparable to the performance of airbreathing turbojet engines. These levels should be adequate for sufficient thrust to be generated for sustained flight.

As in the previous section, the free flow parameters at a cross section of $A_1 = 0.5$ m^2 will be used to determine the mass flow:

$$G = \rho_1 c_1 M_1 A_1 \quad G = 12.23 \text{ kg/s} \tag{13}$$

This is an idealized estimate. In reality, the mass flow will be measured in a smaller cross section within the engine, with the density and speed of sound being higher and the Mach number being lower, due to diffusion.

The required temperature rise to 600 K that needs to be achieved by the reactor heat is as follows:

$$T = T_{\text{nozzle}} - T_{hc} = 209K \quad T_{\text{nozzle}} = 600 \text{ K} \tag{14}$$

The Jovian gas is assumed as pure hydrogen at 600 K with

$$c_p = 14,550 \text{ J}/(\text{kg} \cdot \text{K})$$

The required reactor power for achieving 600 K is then

$$\dot{q} = c_p G \Delta T \quad \dot{q} = 37.2 \text{ MW} \tag{15}$$

Because the Jovian gas consists of ~14% helium and because helium exhibits much lower specific heat capacity than hydrogen, the reactor power for Jovian gas will be lower.

Heating the gas to 600 K will result in the following pressure at the heat chamber exit:

$$p_{\text{nozzle}} = p_{hc}\left(\frac{T_{\text{nozzle}}}{T_{hc}}\right)^{\frac{\gamma}{\gamma-1}} = 350 \text{ kPa}; \quad \frac{p_{\text{nozzle}}}{p_1} = 79.8 \tag{16}$$

At 350 kPa or 3.5 bar, the maximum heat chamber pressure is about 80 higher than the ambient pressure. For comparison, most liquid fuel rockets work at combustion pressures between 10 and 200 bar. However, they are designed to start from sea level, which translates into chamber to ambient pressure ratio between 10 and 200.

The resulting rocket nozzle inlet to an ambient pressure ratio of 79.8 is comparable to high-performance rockets at Earth. This means that already at 600 K heat chamber temperature, the engine pressure budget will be sufficient for sustained flight.

To calculate thrust, the nozzle exit temperature and Mach number need to be calculated. For consistency, $M_e = 3.2$ is assumed for this calculation. The nozzle exit temperature and exhaust velocity are as follows:

$$T_e = T_{hc\text{out}}\left(\frac{1 + (\gamma - 1/2)M_{hc}^2}{1 + (\gamma - 1/2)M_e^2}\right) = 171 \text{ K};$$
$$v_e = M_e\sqrt{\gamma R_j T_e} = 3114 \text{ m/s} \tag{17}$$

The flyer thrust is then [according to Eq. (2)]

$$R = G(v_e - v_1) \quad R = 7.89\text{kN}$$

The pressure thrust term and all heat losses are neglected in this calculation.

Assuming local gravitational acceleration $g_{\text{Jup}} = 23.2 \text{ m/s}^2$ and thrust-to-weight ratio of 0.4 ($L/D = 2.5$), the maximum allowable flyer mass can be calculated as follows:

$$m_{\max} = \frac{R}{0.4 g_{\text{Jup}}} \quad m_{\max} = 850 \text{ kg} \tag{18}$$

The model predicts 850 kg maximum flyer mass at 60 km altitude and 600 K heat chamber temperature. The masses at different altitudes and heat chamber temperatures are shown in Table 1.

V. Conclusions and Future Work

The conducted calculation shows that flight with an NPRE is possible in the Jovian atmosphere. An NPRE can produce sufficient thrust for sustained horizontal flight at heat chamber temperatures as low as 600 K. The analysis shows that the lower layers of the Jovian stratosphere are sufficiently dense to allow for efficient engine operation.

The calculations are based on assumed compression efficiency, without calculating the supersonic diffusion explicitly. The performance is evaluated directly from the undisturbed flow parameters and depends on reference engine size to determine the mass flow. A real ramjet engine would require a smaller cross section than the one assumed for this analysis, due to compressibility effects.

The cruise flight lift and maximum allowable mass are calculated out of assumed L/D ratio. The accurate L/D ratio will impact the operating altitude but can only be determined after more detailed design is conducted.

For these reasons, the results from this analysis serve as an approximate estimate. However, the determined parameters provide the required input for more detailed analysis. A more accurate assessment of the performance of the flyer will require an explicit gas compression model of the engine. The model will need to calculate the mass flow, pressure losses, and gas parameters throughout the engine explicitly, in order to determinate the thrust.

Another task is to determine the optimal engine inlet geometry for the desired flight altitude. This can be achieved by conducting CFD simulations. After the inlet performance is known, it will be possible to calculate the required reactor power for achieving the various heat chamber temperatures and resulting thrust levels.

After the thrust is defined, the flight performance can be determined more accurately. Work can proceed on the flyer external configuration (outer body, aerodynamic surfaces, experimental payload configuration, etc.). It is essential to determine the L/D ratio accurately and for different flight conditions and altitudes as well as to assess different flight scenarios.

The presented methodology can be used for the development of engines and flyers for atmospheric flight on the solar system's other celestial bodies like Saturn, Uranus, Neptune, Venus, and Titan. Apart from the scientific interest in the exploration of celestial bodies, there is also another emerging interest: the in situ resource utilization. Considering the presence of large quantities of powerful energy resources, such as He-3 on some of the planets, flyer missions may prove to be crucial for further space exploration [18].

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