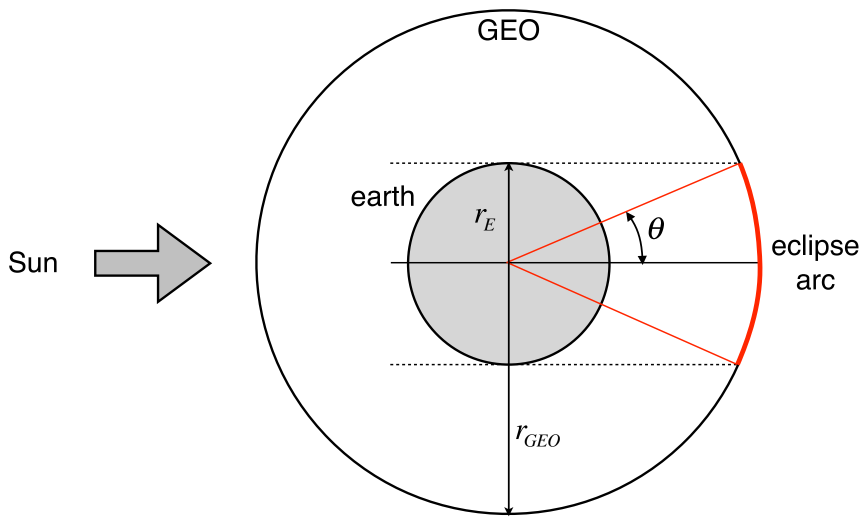
Session 13 : **Electrothermal Augmentation**

The performance of conventional hydrazine monopropellants, or any other type of chemical space propulsion device for that matter, could be improved by the addition of heat from an external, in this case electrical, source. This is particularly obvious for missions that are power intensive as driven by the payload, such as in telecommunications satellites. In this case, the use of electric propulsion is advantageous as the value of α for the propulsion system is reduced significantly since the bulk of the power is required by the payload. As previously determined, lower α's mean higher optimal I_{sp} and overall higher mass savings.

As an example to illustrate this situation, we analyze the use of electrothermal augmentation of hydrazine thrusters on the military satellite DSCS III:

Geostationary satellites are most of the time exposed to the sun, but they still are subject to eclipse periods around the two vernal points (March 21, September 21); when the intersection of the orbital plane (Equator) and the ecliptic plane points to the sun.



With reference to the figure above, the maximum eclipse duration will be,

$$t_e = 2\frac{\theta}{\Omega_{\text{GEO}}} = 2\frac{\sin^{-1}(r_E/r_{\text{GEO}})}{\sqrt{\mu/r_{\text{GEO}}^3}} = 2\frac{\sin^{-1}(r_E/r_{\text{GEO}})}{2\pi/24\text{hr}} = 1.16 \text{ hr}$$

which occurs once per day in the "eclipse season". If the satellite is to remain active during these occultations, enough battery capacity must be incorporated. But obviously, these batteries have ample time to recharge even in eclipse season, and are idle most of the time.

Typically, the solar array is dimensioned with at least 15% extra capacity (at EOL) in order to allow for battery charging. For the military satellite DSCS III:

Array output (BOL/EOL)	977/837 watt (28V±1%)
Batteries	750 watt 1968 watt-hr (100% depth of discharge) 1180 watt-hr (60% depth of discharge)
Payload requirements	723 watt

Thus, there is the possibility of using occasionally both the batteries and the arrays to provide power for overheating the gas generated by a hydrazine decomposition chamber and increase the performance. In the case of the DSCS III, the power available for electrothermal augmentation (ETA) is:

	BOL	EOL
Array power (W)	977	837
Battery power (W)	750	750
Payload power (W)	-723	-723
Power available for augmentation (W)	**1004**	**864**

If the firing is for 1 hr, the batteries provide 750 watt-hr, which is 38% depth of discharge only, and should not compromise battery life.

To calculate the performance achievable, we can use the enthalpy equation again; this time, since the temperature is going to be high and the residence time in the heater long, we can assume equilibrium is reached, which essentially means $x = 1$ (all NH_3 decomposed),

$$\mathrm{N_2H_4} \quad \rightarrow \quad \mathrm{N_2 + 2H_2}$$

Thus, the gas enthalpy (in kcal for the 32 g/mol of hydrazine) is,

$$h_{gas} \;=\; (-2.83 + 7.75\theta + 0.183\theta^2) + 2(-1.967 + 6.6\theta + 0.367\theta^2)$$

Correspondingly, the specific heat at constant pressure (in cal/g°C) is,

$$c_p \;=\; \frac{1}{32}\frac{dh_{gas}}{d\theta} \;=\; 0.6547 + 0.0573\theta$$

The molecular mass of the gas is,

$$M_{gas} \;=\; \frac{M_{\mathrm{N_2}} + 2M_{\mathrm{H_2}}}{1+2} = 10.67 \text{ g/mol}$$

And with this we can compute the specific heat at constant volume,

$$c_v \;=\; c_p - \frac{R}{M}$$

and the ratio of specific heats,

$$\gamma \;=\; \frac{c_p}{c_v}$$

For example, at $\theta = 2$ we have $\gamma = 1.319$ and at $\theta = 1.4$ we get $\gamma = 1.335$.

To find the gas temperature, we again perform an energy balance, but this time we include the electrical energy per unit mass E added to the flow,

$$h_{gas}(T) \;=\; h_{\mathrm{N_2H_4}}(\ell, 298°\mathrm{K}) + E$$

For example, say there is some electrical power P available for augmentation in a thruster operating with a fixed mass flow rate. Then, the electrical energy added to the flow would be,

$$E \;=\; \frac{\eta P}{\dot{m}}$$

where the efficiency η accounts for the losses in transforming available electrical power into heating power. We then use this energy to compute the gas properties and determine the thruster performance.

Conversely, we could start from a desired temperature (and consequently engine performance) and compute what energy (and power) would be required to achieve this goal. This strategy is implemented in cases where structural considerations involving the robustness of the thruster materials limit the operation temperature to some maximum value. For instance,

$$E[\mathrm{J/kg}] \;=\; (h_{gas}[\mathrm{kcal/32g}] - 12) \times 130,180 \qquad \text{where} \qquad 130,800 \;=\; \frac{4180\ \mathrm{J/kcal}}{0.032\ \mathrm{kg/mol}}$$

As an example, assuming $\eta = 0.75$, $P = 1004$ W and a temperature limit of 1800 °K, we compute, for an engine producing 0.813 N of thrust,

$$E \;=\; 2.862 \times 10^6\ \mathrm{J/kg} \qquad \text{and} \qquad Isp \;=\; 315\ \mathrm{sec}$$

A considerable improvement over conventional hydrazine thrusters (∼250 sec).

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