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SPACE PROPULSION SYSTEMS. PRESENT PERFORMANCE LIMITS AND APPLICATION AND DEVELOPMENT TRENDS+

Rolf D. Bühler* and Roger E. Lo**

ABSTRACT

We give a summary about typical spaceflight programs and their propulsion requirements as a basis of comparison for possible propulsion systems of the present and near future. In addition to chemical propulsion systems, we can consider solar, nuclear, or even laser propelled rockets with electrical or direct thermal fuel acceleration. Also, we can consider "non-rockets" which have air-breathing devices and solar cells.

The chemical launch vehicles including Ariane and shuttle will be considered together as candidates for transportation of payloads into Earth orbit and return. They have similar technical characteristics and transportation costs. Examples are given of how to improve the engines and their structure. We give a brief discussion of a possible improvement of payload by using air-breathing lower stages. This is followed by a summary on chemical upper stages and kick stages and a discussion of the future possibilities of supplying energy fuels.

A summary about the electrical energy supply installations is given which gives the performance limits of electrical propulsion. The electrostatic ion propulsion systems which are widely used for trajectory control and for primary propulsion systems are discussed in detail. After a short description of previously used resisto-jets and plasma miniature engines and magnetic coil attitude control sy-

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stems, we discuss the development possibilities of thermal, magnetic and electrostatic rocket engines. The state of development of the nuclear - thermal rocket and propulsion concepts for the distant future are described.

1. INTRODUCTION

Just like in aviation, space flight first of all depends on performance capacity and the weight of the available propulsion systems.

Space flight propulsion systems are basically limited because of a lack of outer support mass such as the Earth's atmosphere which supports the air-breathing engines. Therefore, in most cases they have to carry along the support structure, that is, they are rockets. Compared with other transportation methods, the propulsion system of space flight including the fuel usually represents the greatest part of the total mass, and the payload usually represents a very tiny part of the total mass (a few percent or even a few promille). Even small performance changes in the propulsion system can have a drastic effect on payload fraction and therefore on the cost per kg of payload. For this reason and because of the high cost of the payload itself, the best and most reliable propulsion systems are also the most economical ones.

For propulsion systems into and out of an orbit, there are a very large number of degrees of freedom (for example, compared with aviation engines). Zero gravity, a vacuum and radiation of the sun which lasts up to 24 hours, allow the following:

very small specific thrust levels, accelerations down to 10^{-4} g_o

enormous solar cells areas on extremely light structures

limited expulsion of poisonous fuels

limited shielding of nuclear reactors, etc.

For this reason, there exists a wide range of possible propulsion systems for such programs in space: these extend from the high energy triergol-chemical systems up to the nuclear and solar electrical rockets, as well as magnetic coils, solar cells, etc. The specific thrust, duration of propulsion and, therefore, the flight trajectories for satisfying such missions, vary according to the different propulsion systems over a wide range. In addition, the various propulsion systems have various advantages and disadvantages, such as electrical energy facilities, which count partially as the payload. They have large areas which make maneuvering difficult or there exists safety and waste removal problems for nuclear reactors.

Therefore, it is very difficult to make comparisons of the various propulsion systems for space flight and to give a balanced overview.

In the following paper we will attempt to evaluate and list the present and expected performance levels of various propulsion systems using appropriate applications (flight programs) and we will make comparisons.

We hope that our paper will make it easier for the systems engineer, research planning engineer and development planning engineer, as well as other readers associated with space flight problems, to understand the possibilities and limitations of space flight as far as they are limited by the propulsion systems.

NOMENCLATURE

A	m^2	Cross sectional area
$a_o = \frac{F}{m_o}$	m/s^2	Takeoff rate of rocket
$c_e = \frac{F}{\dot{m}}$	m/s	Effective outflow speed
F	N	Thrust
g_o	m/s^2	Earth acceleration
h	km	Flight altitude
I_{ges}	Ns	Impulse requirement
I_s	s	Impulse per weight
Ma	-	Mach number
$\dot{m}$	kg/s	Mass throughput of fuel or propellant carried along
m_{ANT}	kg	Mass of propulsion system including fuel (gross thrust)
m_B	kg	Mass at shutdown
m_L	kg	Payload
m_S	kg	Structural mass (net)
m_T	kg	Fuel mass
m_W	kg	Mass of propulsion system (dry)
$m_O \equiv$ GLOW	kg	Takeoff mass
n		Stage number
N_E	W	Performance - electrical
N_F	W	Thrust - performance
$t_{BR}=t$	s	Burning time, propulsion duration
Δv	m/s	Propulsion capacity, requirement

α	W/kg	Specific performance
ε	J/kg	Specific energy
η		Energy efficiency
μ		Mass ratio
$\sigma = \frac{F}{m_w}$	m/s^2	Specific thrust of propulsion system (thrust divided by propulsion system dry mass)
LEO		Low Earth orbit
GEO		Geostationary Earth orbit (about 35900 km)
VTO		Vertical takeoff loss unit
VTOHL		Vertical takeoff, horizontal landing, reuseable space transport
VTOVL		Vertical takeoff, vertical landing, reuseable space transport
RPV		Remotely Piloted Vehicle

2. POSSIBLE PROPULSION SYSTEMS, PROPULSION CAPACITY, AND PROPULSION REQUIREMENT FOR TYPICAL FLIGHT PROGRAMS

2.1 PROPULSION POSSIBILITIES FOR SPACE FLIGHT

Table 2-1 gives a morphology of selected propulsion systems for space flight, and also gives an indication about the development status of various propulsion systems. Accordingly, the following are candidates as energy sources (or storage units) for space propulsion systems:

-- Chemical energy carriers, including internal heat in the fuel carried (combustion, decomposition, cold gas) alone or in combination with gases of a planetary atmosphere.

-- Solar energy.

-- Nuclear energy carriers (radio isotopes, nuclear fission, later nuclear fusion).

-- Electrical power plants stationed on the Earth or large satellites, whose energy is transmitted to the propulsion modules using lasers or microwaves (called "laser propulsion" here).

We will not discuss other energy sources in space (for example, free radicals).

According to the basic mechanism of producing thrust or propulsion, there are two kinds of propulsion systems: Rockets and "non-rocket". Rockets are the majority of all propulsion systems used today and planned for the future and produce their thrust or propulsion impulse by expelling mass which is carried along.

The fuel acceleration can occur thermally (by gas expansion), electrically or magnetodynamically or wave mechanically (particle radiation or light radiation), or even purely mechanically. At the present time and in the near future, only the thermal and electrical rockets will be used. Active photons (waves) or particles (neutrons, protons) radiators are limiting cases which will not be discussed here.

The non-rockets produce the propulsion impulse by support on planetary atmospheres, magnetic fields or the radiation field of the sun, including the solar wind. The acceleration of the atmospheric gas can also occur thermally (ramjet), partially mechanically (turbofan), or electromagnetically (MPD-"ramjet"), or this can happen purely passively using a fluid-mechanic method (atmospheric braking).

The more or less passive support on planetary gravitational fields (gravitation gradient attitude control, planetary swingby maneuvers) here is considered as a method for reducing the propulsion requirements and is not discussed as a means of active propulsion. Active support by gravitational fields using gravitational waves (anti-gravity) will also not be discussed here.

For each of the three classification groups (energy, support mass, thrust mechanism) there are hybrids which are mixed or combination propulsion units which follow quite naturally. Some of the important ones include the following (or they could be important in the future):

a) Energy source - hybrids
chemical + solar electric
" + nuclear thermal
" + nuclear electrical
" + laser

b) Support mass combination propulsion systems
Atmosphere + rocket (turborocket, etc.)

c) Acceleration mechanisms
Thermal + electrodynamic
(for example, light arc MPD)

In addition, there are engines which can collect fuels in space and which can store them. Later on, they can expel them like rockets. This is an idea for the future. Table 2-1 shows several energy hybrid systems like the chemical-electrothermal resistojet engines already used. The light arc/plasmadynamic engines implies a mixing of thermal and electrodynamic methods.

In addition to rockets, non-rockets and their combination, there is also the possibility of using Earth-bound or planet-bound takeoff aids (catapults, cannons, rocket sleds, etc.) which have appeared in the literature. Today, there are designs for horizontal takeoff air-breathing lower stages (HTOHL) for moon departure, which will not be discussed here.

2.2 PROPULSION REQUIREMENT FOR TYPICAL FLIGHT TESTS AND PROPULSION CAPACITY

The propulsion capacity of a space vehicle (Δv_{ANT}) must be at least as large as the propulsion requirement (Δv_{CHAR}) of the given flight program or task group, including all of the associated conditions such as payload, flight time, launch site, and launch direction, etc.

$$\Delta v_{ANT} \geq \Delta v_{CHAR}$$

Since space flight propulsion systems are usually rockets, the definitions of propulsion capacity and propulsion requirement is based on the rocket equation (momentum equation).

The Δv-values for propulsion capacity and propulsion requirement (in the future they will be set equal and called Δv) refer to the fictitious velocity increase of a rocket in a gravity-free and drag-free space, which actually provides the energy or momentum

ENERGIEQUELLE	STÜTZMASSE	SCHUBMECHANISMUS	SYSTEME	(Status)*
5 chemische Verbrennungs-, Zersetzungs-, und innere Wärmeenergie 6 ($\leq 1,5 \cdot 10^7$ J/kg)	7 Brennstoff = Treibstoff	thermisch 8	chemische Raketen	E
			Kaltgassystem	E
	Atmosphäre + Brennstoffe	9 mechanisch-thermisch	Turbo-Staustrahl-Rakete	NZ/HT 11
Solarenergie 12	mitgeführter Treibstoff 13	elektrodynamisch 14	solarelektrische Raketen 15	E-NZ
16	"	solarthermisch 17	solarthermische Raketen 18	NZ
19 18	Solarstrahlung und -wind	Strahlungsreflexion 20	Solarsegel 21	NZ
22	planetares Magnetfeld	magnetische Wechselwirkung (Drehimpuls) 23	Magnetspulen-Lageregelung 24	"
Nuklearenergie 25	mitgeführter Treibstoff 26	thermisch 27	nuklearthermische Raketen 28	NZ/HT
29 Radioisotope, Kernreaktor	" " 30	elektrodynamisch 31	nuklearelektrische Raketen 32	NZ*
		37	Kombination nuklear-thermisch - elektrisch (Dual-Mode)	FZ
" 33	Atmosphäre 40	elektrodynamisch 41	Nuklear-MHD-"Staustrahl" 42	FZ
Kernfusion 43	Fusionsplasma + Treibstoff 44	thermisch 45	Fusionsraketen 46	FFZ
50 Kraftwerk (stationär)	mitgeführter Treibstoff 51	thermisch 52	"Laser"-thermische Raketen 53	FFZ
54 Laser oder Mikrowellenübertragung	" 55 "	elektrodynamisch 56	"Laser"-elektrische Raketen 57	FFZ
	Atmosphäre 58	elektrodynamisch 59	Laser-MHD-"Staustrahl" 60	FFZ

ENERGIEHYBRIDE SYSTEME 61

chemische + solare Energie oder nukleare Energie 61	Brennstoff (chemisch) 62	thermisch (einschließlich elektrischer Aufheizung) 63	chemisch-elektrothermische Raketen 64	E
chemische + nukleare Energie 65	Brennstoff (chemisch) 64	thermisch 65	1) Isotope-Monopropellant-Raketen (NIMPHE) 66	NZ/HT
	" "	" 68	2) nuklear-thermische Raketen mit Nachverbrennung 69 ($H_2 + O_2$) 70	FZ
chemische + Laser-Energie 71	Brennstoff 72	thermisch 73	lasergestützte chemische Raketen 74	FFZ

*E Deployable, NZ near future, HT technology already available, FZ distant future, FFZ far distant future.

ˣSmall experimental versions already flown in USSR, but additional technology required for practical sizes (about approximately 300 kw).

TABLE 2-1: Selected propulsion systems for space (morphology)

ORIGINAL PAGE IS OF POOR QUALITY

Key to Table 2.1:

1 - Energy source; 2 - Support weight; 3 - Thrust mechanism;
4 - Systems (number*); 5 - chemical combustion, decomposition and
internal heat energy; 7 - fuel = propellant, atmosphere + fuels;
8 - thermal; 9 - mechanical-thermal; 10 - chemical rockets H,
cold gas system H, turbo-ramjet rocket NZ,HT; 12 - solar energy;
13 - fuel carried along; 14 - electrodynamic; 15 - solar electric
rocket N-HZ; 17 - solar - thermal; 18 - solar thermal rockets NZ;
19 - solar radiation and solar wind; 20 - radiation reflection;
21 - solar cell NZ; 22 - planetary magnetic field; 23 - magnetic
interaction (angular momentum); 24 - magnetic coil attitude control E;
25 - nuclear energy; 26 - fuel carried along; 27 - thermal; 28 - nu-
clear thermal rockets NZ/HT; 29 - radioisotopes, nuclear reactor;
31 - electrodynamic; 32 - nuclear electric rockets NZ*; 33 - combina-
tion nuclear-thermal-electrical (dual-mode); 40 - atmosphere;
41 - electrodynamic; 42 - nuclear MPD "ramjet"; 43 - Nuclear fusion;
44 - fusion plasma + fuel; 45 - thermal; 46 - fusion rockets FZ;
50 - generating station (stationary); 51 - fuel carried along;
52 - thermal; 53 - "Laser"-thermal rockets; 54 - laser or microwave
transmission; 56 - electrodynamic; 57 - laser electrical rockets;
58 - atmosphere; 59 - electrodynamic; 60 - laser MPD ramjet;
61 - Energy-hybrid Systems; 61a - chemical + solar energy or nuclear
energy; 62 - fuel (chemical); 63 - thermal (including electrical
heating); 64 - chemical-electrothermal rockets; 65 - chemical plus
nuclear energy; 66 - fuel (chemical); 65 - thermal; 66 - 1) isotopes,
monopropellant - rockets (NUMPHE); 69 - 2) nuclear-thermal rockets
with afterburner; 71 - Laser energy + chemical energy; 72 - fuel;
73 - thermal; 74 - laser supported chemical rockets

change of the space vehicle required for the mission. Appendix 1 gives a quick reference of the details.

We would also like to point out that the effective exit speeds c_e and the fuel specific impulse I_s as well as Δv can be adapted for partially air-breathing combination propulsion systems (see Chapter 3). Usually the effective exit speed is defined for rockets.

The previously defined Δv propulsion requirement is given in Table 2-2 for a number of missions for impulse-accelerated trajectory transfers#. The numbers given represent typical values for the flight missions and can vary depending on the mission conditions (launch site, launch direction, launch acceleration, influences of the atmosphere, mission time, height and inclination of the target trajectory).

For example, in the case of Earth satellites, the propulsion requirement varies between about 8.7 km/s for an equatorial ascent into an Earth orbit (LEO) to about 13.5 km/s for missions into a Geostationary trajectory (GEO) for non-equatorial launch sites. During the ascent through the Earth's atmosphere, a velocity increment of about $\Delta v \approx 3$-4 km/s could be produced to save fuel using air-breathing combination propulsion systems.

The propulsion requirement of an LEO-GEO trajectory transfer of interorbital spacecraft with subsequent return to the initial trajectory requires an energy expenditure according to Table 2-2 which is comparable with that of an ascent from the Earth into a low Earth orbit.

At the present time, only chemical rocket propulsion systems with a thrust-to-weight ratio of >1 are possible for flight missions

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#In other words, the trajectory arc covered during the acceleration phase only covers a small angle compared with the central mass. Usually this condition is satisfied for chemical propulsion or even nuclear-thermal propulsion.

	Mission	Δv [km/s]	Missionsdauer	Bemerkungen
5	ERDE-LEO 1) Anteil in Atmosphäre	8,7 - 10 / 3 - 4	15 Min. / -	Schub/Startgewicht > 1 Möglichkeit: Luftatmer-Einsatz
10	ERDE - GEO 2)	12,9 - 13,5	5,4 ~~Std.~~ HRS	
11	LEO - GEO (-LEO) chemische Antriebe / elektrische "	4 (9) / 6 (12 5)	5,3 ~~Std.~~ HRS / 0,5-1 ~~Jahr~~ YRS	13 Aufspiralen mit kontinuierlichem Antrieb
14	ERDE - FLUCHT	12,4	-	
5	ERDE-MOND-ERDE	18	10 ~~Tage~~ days	Apollo-Mission ohne Mondlandung
[illegible]	ERDE - MARS	13,2	0,8 ~~Jahre~~ YRS	
[illegible]	ERDE - MARS-LANDUNG - ERDLANDUNG	34,5	1,4 ~~Jahre~~ YRS	
[illegible]	ERDORBIT - SATURN ORBIT 3)	11,8 (21,9)	5,7 (2.0) ~~Jahre~~ YRS	17 ( ) Schnellere Übergänge (Nicht-Hohmann-Transfer)
18	ERDE-URANUS	17,1 / 17,1	8 ~~Jahre~~ YRS / 5 ~~Jahre~~ YRS	ballistisch direkt / via Jupiter-Swingby
19	SOLAR-PROBE	17 7	~~Jahre~~ YRS	Außer-Ekliptik 90°-Mission mit Jupiter-Swingby
22	KOMETENSONDE (ENCKE-Vorbeiflug)	15,8	0,3 ~~Jahre~~ YRS	Vorbeiflug-Geschwindigkeit ≥ 7 km/s 23
[illegible]	KOMETEN-RENDEZVOUS	13,8+12,1 (SEP) 4)	2,7 ~~Jahre~~ YRS (SEP)	Halley-Vorbeiflug mit Tempel-2-Rendezvous
26	KOMET-"SAMPLE-RETURN"	> 35 - 27	5,9 ~~Jahre~~ YRS (SEP)	ballistisch nicht praktikabel
27	FLUCHT AUS SONNENSYSTEM	17,5	30 ~~Jahre~~ YRS	Zeitangabe gilt für Flug jenseits Neptun
25	STURZ ZUR SONNE	32,3	0,18 ~~Jahre~~ YRS	ballistisch nicht praktikabel

1) LEO niedriger Erdorbit
2) GEO geosynchroner Orbit
3) aus 500 km Parkorbit
4) SEP Solar-elektrischer Antrieb (25 kW)

1) LEO low Earth orbit
2) GEO geosynchronous orbit
3) from 500 km parking orbit
4) SEP solar electrical propulsion (25 kW)

TABLE 2-2: PROPULSION REQUIREMENTS OF SEVERAL MISSIONS FOR IMPULSIVE PROJECTORY TRANSFERS

Key to Table 2-2:

1 - Table: 3 - mission duration; 4 - remarks; 5 - Earth-LEO[1)]
8 - thrust/take-off weight; 9 - part in the atmosphere;
9a - possibility: air-breathing use; 10 - Earth-GEO[2)]; 11 - Leo - GEO (-LEO) chemical propulsion; 11a- electrical propulsion;
13 - Spiraling upwards with continuous thrust; 14 - Earth Escape;
15 - Earth-Moon-Earth; Apollo Mission without Moon landing;
15a- Earth-Mars
15b- Earth-Mars - Landing - Earth Landing
15c- Earth orbit - 3)
Saturn orbit
17 - Faster transfers (non-Hohmann transfer); 18 - Earth-Uranus: ballistic direct; via Jupiter Swingby; 19 - solar probe; out of ecliptic 90° - mission with Jupiter swingby; 22 - comet probe (ENCKE-flyby); flyby speed; 24 - comet rendezvous: Halley-flyby with tempel-2 rendezvous; 26 - comet sample return; not practical ballistically; 27 - escape from solar system; type given is for flight beyond Neptune; 28 - crash into the sun; not practical ballistically.

Earth-LEO and for the launch or the landing on many other celestial bodies. Trajectory maneuvers in orbits around the sun or planets such as, for example, LEO-GEO trajectory transfers, could be carried out using thrust accelerations of 10^{-3} - 10^{-5} g_o . For such missions, the propulsion system can have a low specific thrust if there are no mission conditions on the flight duration or conditions on the avoidance of degrading solar cell performance. This might be imposed because the van Allen radiation belts of the Earth might have to be traversed rapidly in order to avoid degradation of solar panels. Such trajectory spiraling requires a higher Δv propulsion requirement because of the large gravitational losses caused by the long propulsion times. For example, for LEO-GEO transverse, and if an electrical propulsion system is used, about 6 km/s is required in addition to increments for trajectory inclination changes.

For missions to other celestial bodies, the Δv propulsion requirement increases rapidly to values which cannot be mastered even in the future by chemical rockets, especially for flights with return of the spacecraft to the Earth (this is because the payload fractions decrease to about 10^{-5} or less). Use of the swingby method usually only results in a slight improvement but can bring about a substantial reduction in flight time.

The performance limits of chemical rocket systems are given in the following Fig. 2-1.

Fig. 2-1 shows the payload ratio μ_L as a function of the propulsion capacity Δv for chemical and non-chemical rocket stages. The required Δv ranges of various groups of missions are shown along the abscissa.

Accordingly, a single-stage chemical rocket with today's technology reaches its performance limits at about Δv ~ 7 km/s. This means that for ascent into a low Earth orbit (LEO), at least a one and a half to two stage launch vehicle is required. The shaded band shown in Fig. 2-1 shows the optimum number of stages for a chemical rocket

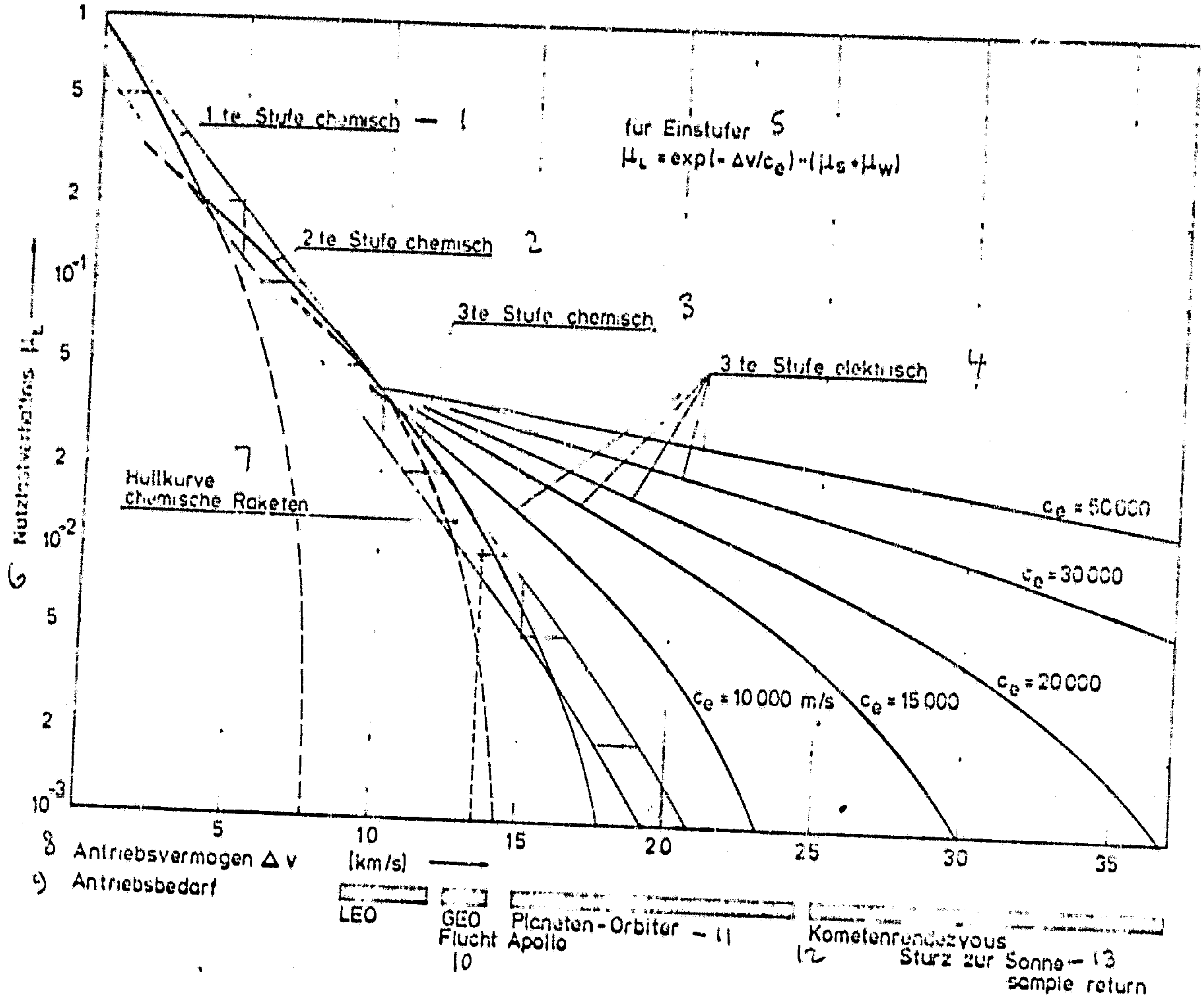

Fig. 2-1: Payload ratio as a function of propulsion capacity with chemical or electrical third stage.

1 - 1st stage chemical; 2 - 2nd stage chemical; 3 - 3rd stage chemical; 4 - 3rd stage electrical; 5 - for single stage vehicle; 6 - payload ratio; 7 - envelope of chemical rockets; 8 - propulsion capacity; 9 - propulsion requirement; 10 - escape; 11 - planet orbiters; 12 - comet rendezvous; 13 - crash into the sun.

and this is the configuration with the best possible payload ratios. We see from this that each Δv increase of about 6.5 - 7 km/s corresponds to a payload deterioration by a factor of about 10. In other words, for the same payload mass, the launch vehicle takeoff weight must be multiplied by 10

According to Fig. 2-1 the propulsion requirement of chemical rockets is limited to values of $\Delta v \leq 21$ km/s for payload ratios of 10^{-3}. In order to cover the propulsion requirement of high energy missions, it is advantageous to use electrical or nuclear-thermal propulsion systems for further acceleration of the spacecraft.

In the case of launch from a low Earth parking orbit LEO, the figure shows third stages with optimized electrical propulsion systems having different exit speeds c_e. Such devices, for example, are used for interorbital freight traffic LEO-GEO-LEO ($\Delta v \sim 12$-18 km/s).

From the curves, we can see that already exit speeds of c_e = 10 km/s have advantages for missions close to the Earth. For c_e > 20 km/s, the payload decrease is reduced with increasing Δv compared with chemical rockets, by a very substantial amount.

Depending on the propulsion requirement, permissible mission duration and importance of fuel costs in LEO for a transport mission, exit speeds of c_e = 30 - 80 km/s can be used, as will be discussed in Chapter 4.

The low specific thrust of electrical propulsion systems means that the production of a required Δv increment requires long propulsion times and, therefore, long mission times. In order to reduce the flight time, in the case of interplanetary missions, the payload is accelerated using chemical third stages to a velocity which is substantially above the escape velocity. However, the reduced flight time must be brought about with a deterioration in the payload ratio by a factor of 3. Even for interorbital traffic, it is planned to use electrical propulsion only above the van Allen belt (altitude about 15,000 km) in order to avoid excessive degradation of solar cells.

3. CHEMICAL ROCKET PROPULSION SYSTEMS AND ROCKET ENGINES

Chemical rocket propulsion systems are used in space flight for a large number of missions which can be represented as follows, ordered according to increasing thrust or total impulse:

-- Attitude control propulsion for alignment of spacecraft

-- Attitude control propulsion for positioning spacecraft and for maintaining position

-- Attitude control propulsion for aligning heavy objects, for example, rocket stages, shuttle orbiters, space laboratories

-- Trajectory correction propulsion for changing direction, such as mid-course corrections

-- Kick stages for large trajectory changes, for example, perigee and apogee propulsion systems

-- Retro-propulsion for transfer into orbits around planets or for landing on them (also for launch)

-- Transfer propulsion and propulsion modules for transfers between trajectories with several course changes

-- Upper stages of carrier rockets to include transfer trajectories and parking trajectories

-- Lower stages and boosters of launch vehicles for launching from the Earth's surface.

The thrust requirement covers about 9 decades from 10^{-3} to 10^{+7} N. The only new category in the future will be returnable transfer propulsion modules, which must deliver a substantial propulsion increment, such as is required for transportation from the Earth's surface to a low Earth orbit.

3.1 PRESENT AND FUTURE SPACE FLIGHT LAUNCH VEHICLE SYSTEMS

Table 3-1 gives the most important launch vehicle rockets used today or which are in an advanced stage of development. Because there is no data available, we do not give any data for launch vehicles from the Peoples Republic of China and the reuseable space transport vehicles probably under development in the USSR. With the exception of the space shuttle, all of these launch vehicles are vertical take-off expendable systems. All of them require 2 or 3 stages to reach low Earth orbit. The space shuttle will be launched with a recoverable 1st stage booster and a 2nd stage which will operate in parallel starting with launch, which is supplied with a non-reuseable fuel tank. The 2nd stage, the orbiter, reaches circular orbit using additional on-board propulsion systems and can land aerodynamically. The space shuttle, therefore, is a two and one-half stage reuseable device, type VTOHL. This not only means we have reached the limits of modern technology in many ways, but we also expect a first substantial reduction in transportation costs into space by using the space shuttle. Fig. 3-1 shows the specific transportation costs for expendable units and future reuseable space transport systems [23]. Two factors reduce the specific transportation costs: total size of the unit and number of launches. An example of the first kind is Saturn V. The Thor/Delta rocket is an example for the second kind. In addition, the costs are reduced because the units can be reused. The value for the space shuttle is 3000 DM/kg and is substantially below the trend curve because it is not completely reuseable. The original target value of 800 DM/kg can probably be reached by stepwise improvements.

From this and from the influence of size, one obtains the value for future heavy space transport systems of about 80 DM/kg (20 $/lb), which has been mentioned frequently.

Bezeichnung — 1		m_0 [t]	m_N [t] (LEO)	[illegible] [t] (LEO)	F [t]	a_0 [g]	$c_{[illegible] Boden}$ [m/s] 3	Typ	n
USA:									
	Saturn V, 5xF1	2928	(125)	4,6	34[illegible]	1,25	2490	VTO	(2)
	Titan III C	635	1[illegible],4	1,8	900	1,42	3110	VTO	3
	Atlas-Centaur	139,5	5,0	3,6	167,2	1,34	2740	VTO	3
	Delta 2910	134	1,5	1,1	1[illegible]	1,41	1620/2580	VTO	2 1/2
	Scout	18,1	0,165	0,9	45,4	2,51	2100	VTO	4
	Space Shuttle, 3xSSME	2028	29,5 (14)	1,5	2760	1,38	3563	VTOHL	2 1/2
UdSSR: 2									
	Wostok A	309	1,4	0,23	492	1,52	3079	VTO	2
	Wostok A 1	295	4,750	1,61	612	2,07	3079	VTO	3
	Kosmos B 1	~ 53,5	0,45	0,84	~ 66	1,23	2133	VTO	2
	Kosmos C 1	37,5	~ 0,70					VTO	2
	Proton D 1 e	800	~ 22,7	2,84	1620	2,02		VTO	3
Europa:									
	ARIANE	203	4,6	2,3	280	1,38	2730	VTO	3
Japan:									
	N	90	1	1,1	150	1,67	[illegible]	VTO	3 1/2
Indien:									
	SLV-3	21	0,05	0,24	30	1,43	2600	VTO	4

TABLE 3-1: Selected launch vehicles

1 - Notation; 2 - USSR; 3 - ground

The costs and the transportation capacity limited by the technology represents a challenge. This is a challenge to overcome the present day performance and program limitations. It should be realized that in order to overcome them can be justified from the further development of space flight. By reading the present literature (including a large study of NASA [1]), one can already clearly see the future goals:

-- In the range of small to medium yearly transportation expenditures, including personnel transportation: achievement of complete recoverability and reuseability with simultaneous reduction of the number of stages of launch vehicles with horizontal landing characteristics, of the size comparable with the space shuttle.

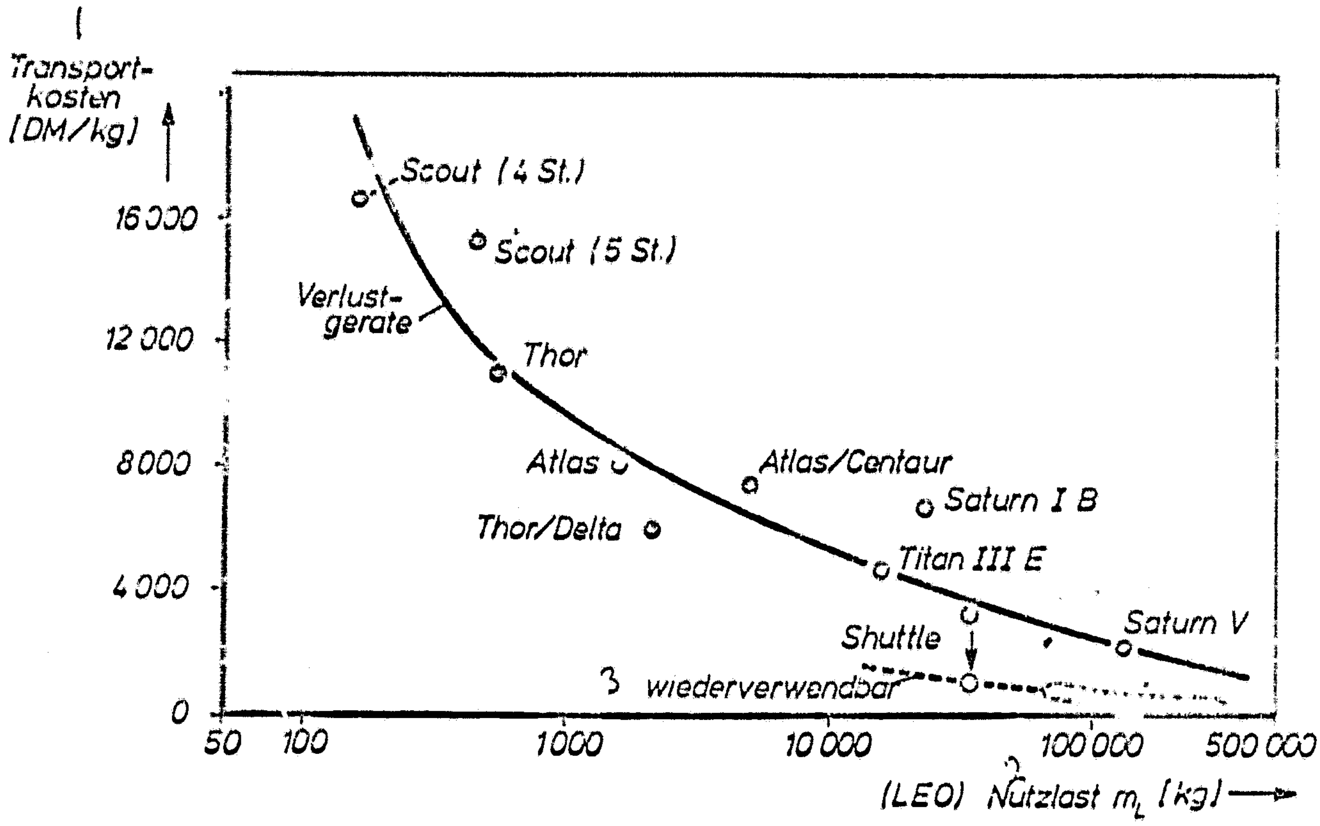

Fig. 3-1: Specific transport costs (LEO) as a function of payload.

1 - transportation costs; 2 - payload; 3 - reuseable

-- In the range from large to very large yearly transportation costs, especially for equipment: construction of large-scale units with several 100 ton payload capability. The individual stages will land vertically using clusters of engines and these can be reused. However, horizontal landing is also being seriously considered.

This trend to medium sized winged VTOHL devices with only a single stage if possible (SSTO: Single-Stage-to-Orbit) cannot yet be clarified on the basis of numerous studies [2]. This result is controversial but it is based on the fact that the logistics of single space vehicles is much simpler, and there are important consequences for the operational costs in spite of the reduced payload fraction.

The large transport vehicles (so-called Heavy Lift Launch Vehicle, HLLV) were first the topic of the majority of project studies with single stage, vertical takeoff and landing systems (VTOVL). Two new project studies again seem to consider two-stage systems with vertical takeoff and vertical or horizontal landing (VTOVL or VTOHL) for both stages [23, 89].

These space transport systems just mentioned, however, are not within the limitations of today's technology. Their development assumes numerous and substantial advances in the direction of even lighter structures and better propulsion systems. In addition, there is the new requirement for minimum maintenance after each flight.

The materials and design methods will not be discussed here. However, we can say that the new materials will play a fundamental role just like new design methods (for example, computer controlled interval construction methods and control configured vehicle (CCV) technology).

We expect the following improvements in propulsion systems, both in the components and in the systems:

-- High pressure engines
These allow high relaxation ratios and consequently high specific impulse, even on the ground. The space shuttle uses 3 space shuttle main engines (SSME) at 203 bar. An additional increase in the combustion chamber pressure will require overcoming technological limitations of numerous components [3]. The oxidizer must be used for cooling such engines. Gas generators [4] operating stoichiometrically are requisites for realizing expansion cycles at ultra-high pressures. Turbopump technology is especially critical here. The specific mass must be reduced by increasing the revolution rates. Analyses have shown that the probable upper limit is 600 bar for the combustion chamber pressure.

-- Nozzles with variable area ratios
The matching of the area ratio to the decreasing external pressure during ascent of carrier rockets promises to bring about a substantial improvement in the integral performance. This can be done using extendable nozzle extensions in two stages (dual position nozzles) or in four stages (multi-position nozzles). There is another possibility of using "unconventional" nozzles with internal or external nozzle surfaces. Ring throat nozzles or linear slit nozzles can be integrated very well in the base area of winged launch vehicles (for example, the Aerospike engine of Rocketdyne). This can be done with super-elliptical nozzle areas for conventional engines [5].

-- Mixed mode propulsion systems
These improve the performance drastically, especially for marginal missions such as the SSTO. The concept is based on an analysis of R. Salkeld [6] and simultaneously or sequentially uses fuel with a high density and therefore a relatively low specific impulse in conjunction with fuels with a high specific impulse and therefore a low density. The mixed operation allows higher payloads than if one uses only one of the two fuel pairs.

Parallel operation gives improved overall results. This means a mixed mode SSTO launch vehicle in the simplest case could operate with F1 (engine of the 1st Saturn V stage) in conjunction with the SSME engines. These would be supplied from a common LOX tank and separate RP1 and LH_2 tanks. Since the substantial weight reduction connected with the use of such engines can be easily calculated for the two fuel combinations, these are now being studied at the request of NASA. Rocketdyne [7] is investigating an SSME modification which will burn hydrogen and RP1, propane or methane in conjunction with LOX. In this engine, the two combinations can be burned in series. At Aerojet, the concept of a dual expander enginer [4] has been developed, which burns LOX/RP1 at 414 bar in a central chamber at the same time as it is burning LOX/LH_2 in an outer ring chamber at 207 bar. Therefore, in this operational stage it can produce 2700 kN thrust during the launch phase (75% of this is produced by LOX/RP1). In the higher atmosphere, only the ring burner chamber operates, and its gap nozzle produces over 800 kN thrust with a simultaneous altitude matching effect.

-- Air breathing propulsion systems

In addition to the mixed mode principle, there is a second possibility of improving space vehicle propulsion systems on the systems plane. This is done by using air breathing engines. This not only substantially improves the fuel specific impulse, but also results in a substantial increase in the propelled mass. This solution, therefore, places special requirements on the overall systems analysis. The engine performance data also depend on the flight speed and the flight altitude. Fig. 3.2 gives the relationships between power increase using air breathing engines, additional propulsion system mass and the payload ratio of the pure rockets and air breathers for a single first stage. These are space transport vehicles with parallel operation of rockets and air breathers. The uppermost curve represents the ideal case where the increase of the exit speed $c_{e\ eqiv.}$ is assumed without additional engine mass. This is equivalent to the specific impulse. The average curve variation shows the increase in mass of the propulsion system due to air breathing propulsion, which is required to reach the corresponding equivalent exit speed $c_{e\ eqiv.}$. Rocket structural mass, additional mass of the air breathing system and rocket payload mass give the variation of the second qualitative curve. The region between this curve and the ideal curve represents the possible gain in payload by air breathing engines.

As the specific impulse increases or the equivalent exit speed $c_{e\ eqiv.}$ increases, first the payload gain will increase due to air breathing engines up to a certain maximum. Then it decreases again until finally no payload gain can be achieved anymore. At this point the increase in the specific impulse is completely consumed by the

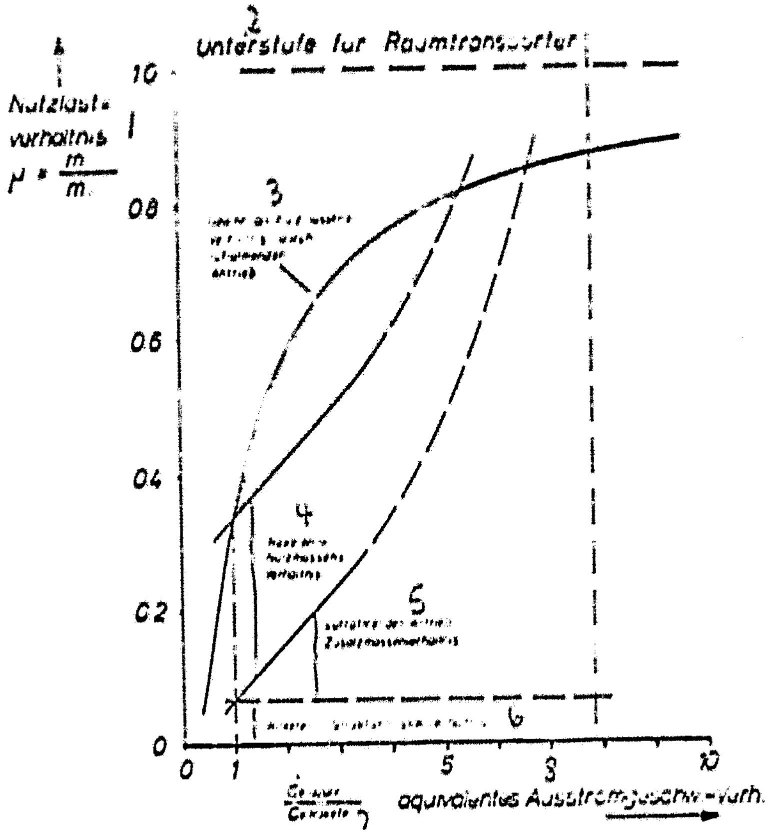

Fig. 3.2: Qualitative representation of possible payload gain using air-breathing propulsion for the first stages of space transport vehicles.

1 - payload ratio; 2 - lower stages for space transport vehicles; 3 - gain in payload ratio using air-breathing propulsion; 4 - rocket payload ratio; 5 - air-breathing propulsion. Additional mass ratio; 6 - rocket structural mass ratio; 7 - rocket; 8 - equivalent exit velocity ratio.

mass increase of the propulsion system. A further increase would not make sense.

Using air breathing engines, we can expect substantial increases in the payload ratios in the future, if the additional masses of the propulsion unit are split off at the edge of the atmosphere.

The previously mentioned improvement possibilities are found again in the results of studies performed recently. The values shown in Fig. 3-3 for Saturn V and space shuttle show the price which must be paid for reusing the vehicle and returning it for additional missions, and the cost for horizontal landings. The curves of constant payload correspond to the four regions defined by NASA. The improvement suggestions can be found along this line of the shuttle payload: according to results which have been especially generated by NASA Langley, VTOHL launch vehicles can be reduced to about 60% of the mass of the present shuttle vehicle with a simultaneous reduction of the number of stages to 1, if mixed mode propulsion systems (No. 2 in Fig. 3-3; details in Table 3-2) are used in conjunction with advanced lightweight structural techniques [4]"

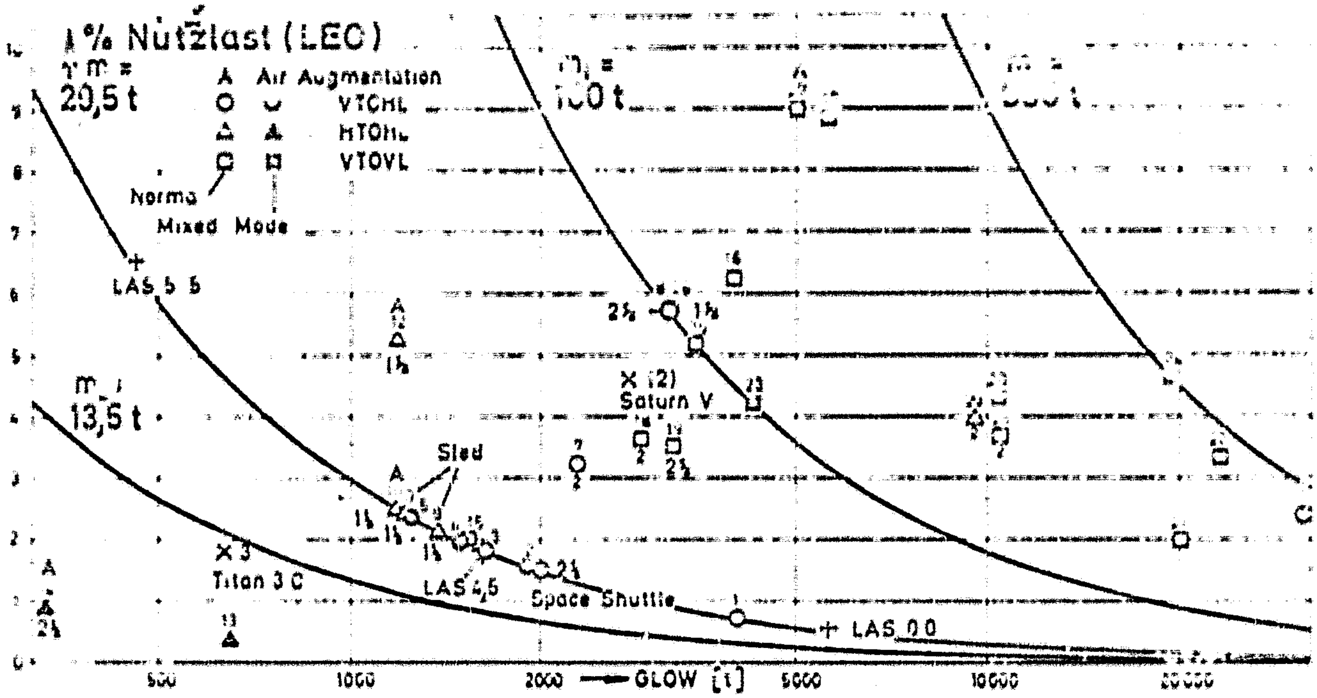

Fig. 3-3: Conceptual carrier systems for the four payload ranges up to 900 tons defined by NASA: payload fraction as a function of take-off weight (GLOW). (x: existing VTO-devices for comparison, numbers above points: running numbers in Table 3-2. Numbers next to or below points: number of stages n, single-stage vehicles without indication of stage number).

1 - payload

and central configurations. (Ref. 5: CCV devices are lighter if one drops the requirement for wideband flight stability. The stability is produced within narrow limits using computer controlled devices*. Horizontal takeoff vehicles could be made even lighter, but then require a rocket sled as a launch aid [9, 10]*. Even if conventional technology is used, a HTOL launch vehicle with two turboair jet boosters is even lighter. When it reaches Mach 3.5, it is detached at about 16 km altitude and these are brought back just like RPVs [11]*. Assuming advance construction techniques, and ramjet propulsion systems with supersonic combustion, the payload of these launch vehicles

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*This refers to the number in Table 3-2.

increases from 29.5 to 62 tons [12]*.

For the payload range above the shuttle, the VTOVL studies are most numerous in number. We expect that in individual steps the orbiter as well as the booster of the shuttle will be changed and therefore payloads of up to over 100 tons per flight will be transported [7, 18, 19]*. These launch vehicles will then be used to an increasing degree for space near the Earth up to GEO. On the other hand, the development of very heavy launch vehicles depends almost entirely on the decision about energy satellites in the space industrialization and colonization. Here again there are conventional designs (for example, 21: the Neptune concept)*, advanced designs (24: mixed mode)* and optimistic designs (17: CCV design)*. Air breathers provide the highest payload fraction (27: rocket with non-conventional nozzles and ramjet propulsion with external hydrogen combustion under supersonic conditions)*. Which of the advanced technologies will be used will finally be decided by other questions, including cost effectiveness, which depends greatly on the frequency of the flights.

-- High energy and advanced chemical rocket fuels

In addition to the present advanced technological concepts, discussions are continuing on the use of fuels with a higher energy content. For launch vehicles it is clear that the known possibilities of metal combustion (in Triergols) and the replacement of oxygen-based oxidizers by fluor oxidizers will not take place because there is not cost effectiveness and because of environmental reasons. Accordingly, LOX/H_2 will be the combination with the highest effective specific impulse for launch vehicles.

-- Non-conventional fuels

In addition to air breathing engines, only on conventional fuels, excited species, radicals and similar substances are possibilities beyond the ones discussed above. This is because non-chemical propulsion can hardly be expected for the near future.

Such fuels will certainly revolutionize the technology of launch vehicles. This would mean that exit speeds of up to 6,000 m/s

Nr.	Bezeichnung 1	m_0 (t)	m_L LEO (t)	μ_L (%)	Typ	n	Ref.	Anmerkungen 2
1	Konv. SSTO	4.500	29,5	0,7	VTOHL	1	8	I_s = 4.464 m/s
2	Konv. MM Shuttle	1.905	29,5	1,5	VTOHL	1	1	2 x F 1, 6 x SSME
3	Adv. Langley Shuttle	1.633	29,5	1,8	VTOHL	1	8	25 % leichter, I_s = 458 m/s
4	Adv. Langley MM Shuttle	1.4[illegible]8	29,5	2,0	VTOHL	1	8	25 %, RJ5/LOX, H_2/LOX
5	MM-CCV Shuttle	1.[illegible]56	29,5	2,3	VTOHL	1	8	25 %, RJ5/LOX, H_2/LOX
6	Geflügeltes HLLV	31.162	907	2,9	VTOHL	1	1	117 m hoch, DPN 5
7	Growth Shuttle	2.313	75	3,2	VTOHL	2	9	SSME + LOX/MMH Booster
8	Konv. HLLV	3.175	181,5	5,7	VTOHL	2 1/2	1	2 Booster + ET
9	Sled L. SSTO	1.332	29,5	2,1	HTOHL	1 1/2	10	Startschlitten
10	Adv. Sled SSTO	1.220	29,5	2,4	HTOHL	1 1/2	11	Startschlitten
11	LA-Booster SSTO	1.160	29,5	2,5	HTOHL	1 1/2	11	2 x Mach 3,5 Booster
12	Adv. LA-Booster SSTO	1.180	62	5,2	HTOHL	1 1/2	11	Advanced Scramjets
13	MM-HTOHL	644	2,3	0,4	HTOHL	1	12	H_2/RP1/LOX
14	Airlaunch MM	329	2,9	0,9	HTOHL	2 1/2	13	1. Stufe: Flugzeug; 4 2. Stufe: MM + ET
15	Koelle Studie	1.514	30	2,0	VTOVL	1	14	12 x SSME
16	Koelle Studie	4.000	250	6,2	VTOVL	1	14	CCV-Design
17	Koelle Studie	5.650	500	8,8	VTOVL	1	14	CCV-Design
18	Shuttle Growth HLLV	2.858	104	3,6	VTOVL	2	9	SSME + LOX/MMH Booster
19	Shuttle derived HLLV	3.220	113	3,5	VTOVL	2 1/2	9	4 SRB, ET
20	SPS-HLLV	10.433	450	4,3	VTOVL	1	9	MM : SSME + LOX/MMH Booster
21	Neptun	20.000	400	2,0	VTOVL	1	15	
22	NASA-HLLV	23.000	771	3,4	VTOVL	1	1	4 x LOX/RP1 Booster
23	MM-HLLV	4.309	181,5	4,2	VTOVL	1	1	H_2/RJ5/LOX, UN
24	MM-HLLV	19.505	907	4,6	VTOVL	1	1	H_2/RJ5/LOX, UN
25	Langley-HLLV	3.500	181,5	5,2	VTOVL	1	8	25 % leichter
26	Drop Tank HLLV	3.175	181,5	5,7	VTOVL	1 1/2	1	8 x LH_2 ET
27	Ext. Ramjet HLLV	5.035	454	9,0	VTOVL	1	1	Ext. H_2-Verbrennung 3
28	Boeing HLLV ballistisch	10.472	391	3,7	VTOVL	2	89	1.Stufe RP-1/H2/LOX 2.Stufe H2/LOX
29	Boeing HLLV geflügelt	9.566	381	4,0	VTOHL	2	89	1.Stufe RP-1/H2/LOX 2.Stufe H2/LOX

30

TABLE 3-2: Launch vehicle concepts

1 - Name; 2 - Remarks; 3 - combustion; 4 - aircraft; 5 - 117 m high, DPN; 6 - winged HLLV; 15 - 17 - Koelle study; 28 - Boeing HLLV ballistic; 29 - Boeing HLLV winged; 30 - 1st stage

could be achieved with 15% of atomic hydrogen in normal hydrogen [21]. If pure atomic hydrogen were used, one could even reach 20,000 m/s. Hydrogen and ammonia in the metallic state also reach the same value. Theoretically, these could be produced with a large energy supply at high pressures and extremely large magnetic fields. One could reach values of up to 30,000 m/s for excited states of helium, so-called helides. The payload fractions of such fuels are obvious. This would mean that a VTOVL-SSTO transport vehicle could carry 3629 tons of payload and GLOW could provide 200 tons of payload in LEO [1]. Controlling these extremely sensitive materials and considering the fact that their manufacture will require breakthroughs in low temperature technology and magnetic field technology, however, means that they will be developed in the very distant future.

3.2 KICK AND TRANSFER STAGES

The most important existing upper stages or those which are under development or being conceived are the perigee and apogee kick propulsion units. The kick stages and the modules are shown in Table 3-3. Except for Centaur and Transstage, most of the units of this category use solid fuels. This is because for the same structural mass they give much higher overall impulse (see Fig. 3-4). According to a contract of the USAF for the development of upper stages for the "Space Transport System" (STS, Space Shuttle), solid fuels will be used. Three axis stabilized modules weighing 2700 to 9700 kg will make up the Interim Upper Stage (IUS) in one to three stages. In addition, SSUS - A and D will be used and spin stabilized for the payloads previously transported by ATLAS and DELTA (see Table 3-4).

The trend to liquid propulsion systems, however, is also clear here. In the range of medium energy fuels, for the Jupiter orbiter GALILEO, a spin stabilized propulsion system based on the SYMPHONIE technology is being developed. No solid fuel system would have been capable of providing pulses which are finely structured over many individual propulsion periods.

Bezeichnung 3	m_S [kg]	m_T [kg]	Treibst. 5	F [kN]	c_e [m/s]	t_{Br} [s]	I_{ges} [MNs]	Stabil.	Ref.
Oberstufen: 4									
Centaur	2.606	13.560	H_2/O_2	133,4	4.356	443	59,1	6 3-Achsen	24
Transstage	1.750	10.580	N_2O_4/AZ50	71,0	2.963	442	31,3	3-Achsen	24
PGS :									
PO.68	105	665	Isolane 29/9	42,0	2.708	44	1,55	Spin	24
Fw 4	26	275	AP/PBAN	25,4	2.605	30	0,77	Spin	24
TE 364-3	96,5	650	AP/PBAN	42,0	2.805	43	1,82	Spin	24
TE 364-4	95	1.040	AP/PBAA/Al	53,0	2.785	55	2,90	Spin	24
ABM :									
TE-M-616	30	333	AP/CTPB	26,6	2.844	36	0,95	Spin	24
SUM 6 A	34	306	AP/CTPB	90,0	2.805	29	0,66	Spin	24
Sirio	27	177	Feststoff	21,5	2.746	23	0,[illegible]5	Spin	24
Symphonie	18,5	148	N_2O_4/AZ50	0,40	2.971	1.100	0,44	Spin	24
SUM 4	64	642	AP/CTPB	55,0	2.605	33	1,80	Spin	24
1) Mage III	35,3	655	SNIA-FLEXADYNE	35,9	2.903	53	1,90	Spin	25
2) CT 600	61,8	644	N_2O_4/AZ50	0,417	2.991	4.620	1,90	Spin	25
Retro-Antriebe: 8									
Lunar-Orbiter	21	125	N_2O_4/AZ50	0,45	2.726	757	0,34	6 3-Achsen	24
Mariner 9	110	463	N_2O_4/AZ50	1,33	2.775	966	1,28	3-Achsen	24
Viking 75	223,8	1.300	N_2O_4/AZ50	1,33	2.775	2.712	3,61	3-Achsen	24
1) Galileo RPM	189	850	N_2O_4/MMH	0,40 0,01	3.011 2.795	ca. 4.600	2,48	Spin	26
Antriebs-Moduln: 9									
Burner II a	70 30	650 240	AP/PBAA/Al AP/CTPB	44,1 34,3	2.850 2.572	42 28	2,47	6 3-Achsen	27
Burner II-4	197	1.040	Feststoff 7	53,0	2.785	55	2,9	3-Achsen	24
OV 1 B-3	173,3	640	Feststoff 7	42,0	2.805	43	1,8	3-Achsen	24
2) MOPU	126,3	655	SNIA-FLEXADYNE	35,9	2.903	53	1,9	Spin	28
2) Hetam Ic	370	1.600	F_2/H_2	5	4.217	1.350	6,8	3-Achsen	29
2) Limos 1600	220	1.600	N_2O_4/AZ50	1,64	2.991	2.990	4,9	3-Achsen	26

TABLE 3-3: Selected propulsion systems of kick stages and upper stages, retro-propulsion systems and propulsion modules 1) under development, 2) concepts

3 - Name; 4 - Upper stages; 5 - fuel; 6 - axes; 7 - solid propellant; 8 - retro-propulsion units; 9 - propulsion modules

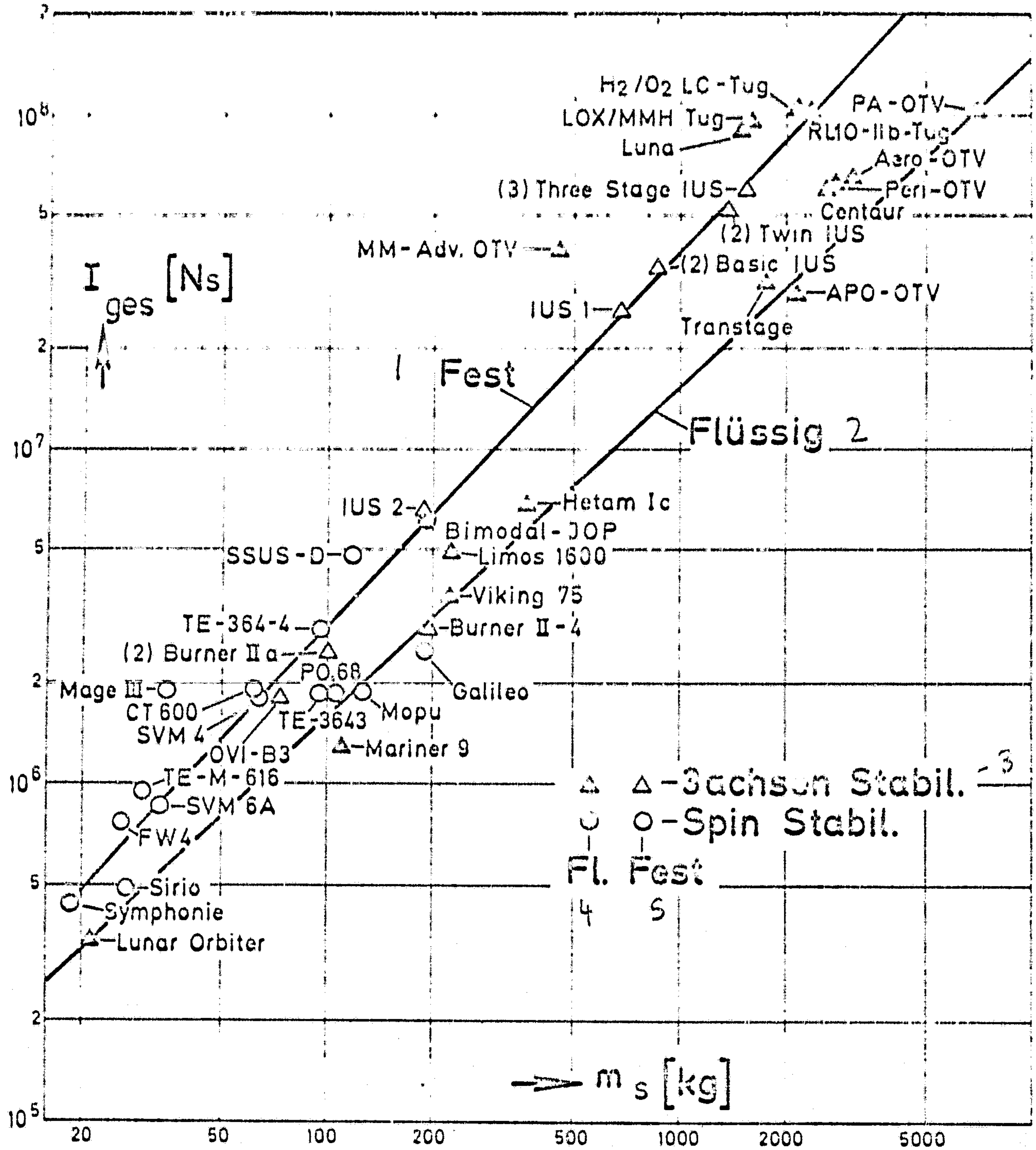

Fig. 3-4: Existing conceptual propulsion systems for transfer stages and kick stages, retro-propulsion systems and controllable upper stages: total impulse as a function of structural mass.

1 - solid; 2 - liquid; 3 - axes stabilization; 4 - liquid; 5 - solid

1 Bezeichnung	m_G [kg]	m_T [kg]	2 Treibstoffe	F [kN]	c_e [m/s]	t_{Br} [s]	I_{ges} 3 [MNs]	Stabil.	Ref.
R L 10 - IIb	2336	23 100	H2/O2		4501		104.0	3	16
R L 10 - IIb	2236	22 886	H2/O2		4501		103.0	3	16
LOX Cooled (engine)	2152	22 577	H2/O2		4642 (ASE)		104.8	3	16
LOX (MMH-Tug)	1584	24 632	O2/MMH		3865		95.4	3	16
MM-Tug	2027	22 648	O2/MMH/H2		3865/4642			3	16
MM-Tug	1780	23 655	O2/MMH/H2		3865/4642			3	16
Perigee OTV	2724	13 426	H2/O2		4619		62.01	3	17
Apogee OTV	2131	6 305	H2/O2		4619		29.12	3	17
PA - OTV	6942	22 225	H2/O2		4619		102.7	3	17
Aeromanoeuvering OTV	3040	22 000	raumlagerfähig 6		2642		64.7	3	18
Basic IUS	~679 ~189	~9 021 ~2 511	Solid Solid	~18.04	~2900 ~2900	145	~22.16 ~7.28 32.44	3	19
Twin Stage IUS	1358	~18 024	Solid		~2900		52.32	3	19
Three Stages IUS	1547	20 553	Solid		~2900		59.60	3	19
SSUS-A Mod.Minut. 3.Stufe 4			Solid		~2900		•	Spin	19
SSUS-D	~117.5	1 679	Solid		~2900		4.87	Spin	19
Advanced OTV	~ 448	~9 020	Mixed Mode		~4300		~39	3	19
Bimodal Jep	184.7	850	F2/N2 H4	4.00	4658		3.96		20
Spalte: 1 5	2	3	4	5	6	7	8	9	

TABLE 3-4: STS upper stages and advanced OTV concepts for LEO/GEO/LEO transport.

1 - name; 2 - fuels; 3- tot; 4 - stage; 5 - column; 6 - storable in space

The development will go from this space storable combination to half or completely cryogenic systems. LOX/MMH is suggested as a fuel combination [16] and LOX/H_2 will certainly be used in future "Orbit Transfer Vehicles" (OTV) [17]. One of the engines with pump propulsion is the "Advanced Space Engine" (ASE) [22] developed by Rocketdyne.

Fig. 3-4 shows that advanced design concepts will lead to the expectation of structural factors just like for solid fuel propulsion systems. In the range of the maximum total impulses, such as are required for LEO to GEO and back, the mixed mode principle could also be used to advantage for marginal missions [16]. The total costs for meeting a given transport task will also decide whether such LEO/GEO/LEO single stage vehicles will be used, or whether non-reuseable machines will be used entirely or partially. After using LOX/H_2 for kick stages, NASA [1] is considering the transition to LF_2/LH_2. This would mean that the state of technology of fluor propulsion systems would be used for a practical purpose. The use of fluor in large amounts (such as is required for high total impulse), however, is doubtful. Instead, we expect that it will be used for missions with somewhat reduced propulsion requirements (2-5 MNs), for example, planetary missions. Here we will wait to see whether the NASA-financed JPL development of a F_2/N_2H_4 propulsion system is successful or not. In addition to a very high specific impulse (see Table 3-4), this bimodal system has the advantage that both the main propulsion and the attitude control system is supplied only from a single fuel tank. Also, two thrust stages (monergol with N_2H_4 and diergol with F_2/N_2H_4) are required for trajectory corrections and retropropulsion, and this requires only a single engine. Summarizing, we can see that high energy chemical fuels will soon be used for deep space missions (high Δv).

3.3 PROPULSION SYSTEMS FOR ATTITUDE CONTROL AND TRAJECTORY CONTROL

The rocket propulsion systems used for attitude control and trajectory control are often called "secondary propulsion units", because usually they are installed in addition to a main propulsion system. Compared to the engines of launch vehicles, they provide a small Δv velocity increment or total impulse and a low characteristic thrust (see Table 3-5).

The thrust range given in the table shows the typical design range of chemical propulsion systems for attitude control and trajectory correction systems [31]. In some cases there are substantial deviations. For example, for the nutation damping of INTELSAT V (communications satellite), 22 N thrust engines are used. If the electrical propulsion is used, to be discussed below, for trajectory control, the engine thrust has values in the range of a few mN. The engines for attitude control and trajectory control of manned spacecraft in future space transport systems occupy a special position.

	[illegible] — 1	[illegible] — 2
4 — Δv - Bedarf [m/s]	0.5 - 20 (≈ 50)[1]	50 - 600 (760)[1]
5 — Gesamtimpuls I_{ges} [Ns]	250 - 100 000 (1,8 10^6)[1]	≈ 10^4 - 3 10^6 (7.5 10^7)[1]
6 — Schub [N]	0.02 - 0.1 15 - 3870[1],[2]	0.1 - 10 (360 - 26 700[1],[2])
7 Betriebsart[3]	P (D)	D
8 Pulsanzahl	10^3 - 10^6	1 - 3650
9 Pulsdauer [s]	10^{-3} - 300[4]	300 - 4.2 10^4 [5]
10 Minimalimpuls [Ns]	10^{-6} - 10^{-2}	—
11 elektrischer Energiebedarf	12 vernachlässigbar bei chemischen Antrieben 0,1 kW bis mehrere kW bei elektrischen Antrieben	

1) SPACE SHUTTLE ORBITER
2) bemannte Raumfluggeräte
3) P Pulsbetrieb D Dauerbetrieb
4) Entsättigungsmanöver mit elektrischen Ionentriebwerken
5) elektrischer Antrieb

TABLE 3-5. Typical design ranges and requirements of attitude control systems and trajectory control systems for satellites.

1 - attitude control; 2 - trajectory correction; 4 - Δv - requirement (m/s); 5 - total impulse I_{tot} (Ns); 6 - thrust; 7 - propulsion type; 8 - number of pulses; 9 - pulse duration; 10 - minimum impulse (Ns); 11 - electrical energy requirement; 12 - negligible for chemical propulsion. 0.1 kW up to several kW for electrical propulsion.

---

1) Space shuttle orbiter
2) manned space vehicles
3) P pulsed opration
   D continuous operation
4) Desaturation with electrical ion engines
5) electrical propulsion

Their thrust level extends into the kN range.

In addition to the general requirements known from the main engines, the requirements for high reliability, low weight, small dimensions, etc., which also apply for trajectory control and attitude control systems, according to the preceding table there are a number of other requirements. This includes pulsing capability, high number of pulses, variable pulse duration of a few milliseconds up to several hours, good reproducibility of the thrust profile with low ignition delay time, low thrust buildup time and decay time of a few ms. The long activation times of the thrust nozzles can occur for attitude control purposes during so-called desaturation maneuvers, especially when electrical engines are used. Such maneuvers are required for attitude control systems, angular momentum storage devices (angular momentum wheels, reaction flywheels, moment torque gyroscopes) and angular momentum generators. In these cases, the influence of disturbing moments on the satellite orientation is compensated first by an angular momentum storage device. The angular momentum accumulated over time is removed during a desaturation maneuver by activating the thrust nozzles. The requirement for the smallest possible minimum impulse (impulse bit) of the actuator system, which is due to attitude control, requires a favorable fuel consumption for stationary operation (limit cycle) during the mission phase. Additional criteria for selecting the propulsion system for attitude control and trajectory control include environmental compatibility and exhaust gas properties. The fuels must be storable for long times (missions of up to 10 years!) and over a wide temperature range in a zero g environment, and the degassing losses and leak losses must be infinitesimally small. The fuels or their combustion products must not falsify experiments and must not precipitate onto the optical surfaces in valves or nozzles (for example, attitude measurement sensors). Also, solar cell degradation should be avoided and the thermal balance of the spacecraft must not be influenced when there is a degradation of the thermal control surfaces.

Many of the fuels often used in primary propulsion systems do not satisfy these requirements because they are not storable, have ignition delays or other difficulties. They cannot be used for attitude control and trajectory control. The propulsion systems developed for this can be classified into solid fuel systems, liquid fuel systems, gas systems and hybrid and electrical propulsion systems (see, for example [32]). Solid fuel systems can only be used in individual cases because of the fact that they cannot pulse, and in spite of their good storability characteristics. In these cases they are used as sublimation propulsion systems for controlling spin. When there is a small propulsion impulse requirement, cold gas systems are used which lie in the lower range of the performance spectrum but show weight advantages up to about $I_{ges} = 10^3$ Ns [30, 31]. For larger propulsion requirements, for example, the important class of geostationary communications, navigations, or weather satellites, the hydrazine liquid system has been used as the standard propulsion unit.

In future high performance satellites, the use of a hydrazine propulsion system leads to unacceptably large weight fractions above 30% according to Fig. 3-5. These will require long mission times of 10 and more years for high accuracy requirements for maintaining the angular attitude and the position [33]. The use of a higher performance two material system compared with the hydrazine system only results in a slight improvement. On the other hand, the use of electrical ion propulsion (for example, RIT-10), with a specific impulse of I_s = 3000 s, an order of magnitude larger than that of chemical propulsion systems, results in a substantial weight savings of the propulsion system. Electrical propulsion including magnetic coil attitude control systems are discussed in Chapter 4.

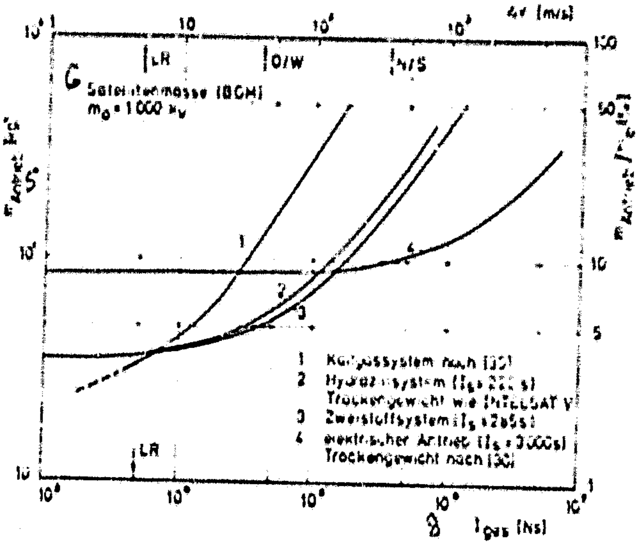

Fig. 3-5. Dependence of the propulsion system mass and its weight fraction on the propulsion capacity for a 1000 kg satellite (note that approximate correspondence of Δv and I_{tot}).
1 - cold gas system according to [30];
2 - hydrazine system (I_s = 220 s) dry weight flight INTELSAT V
3 - two material system (I_s = 285 s)
4 - electrical propulsion (I_s = 3000 s) dry weight according to [30]
5 - propulsion; 6 - satellite mass;
7 - propulsion; I_{tot}

Fig. 3-5 shows the approximate propulsion requirement for attitude control (LR), east-west translation correction (O/W) and north-south angle correction (N/S) for a 7-year mission duration of a geostationary satellite weighing 1000 kg.

For the electrical actuating system, it is assumed in this figure that all of the tasks of attitude control and trajectory control will be carried out by electrical propulsion systems, which consist of four ion engines and have a dry weight of 66 kg [34]. Because of the low thrust electrical propulsion systems [illegible] the weight savings of these propulsion systems cannot be used above $I_{spez} > 10^4$, and therefore cannot be used for many satellite missions. Instead, they require a conventional chemical propulsion system for rapid trajectory changes or attitude changes. However, the electrical propulsion system is very well suited for carrying out trajectory control because of its performance potential, especially for the north-south angle correction of the very important geostationary satellites.

4. ELECTRICAL PROPULSION UNITS AND SYSTEMS

Electrical propulsion units for space flight consist of solar or nuclear rockets (with few exceptions) having electrical fuel acceleration.

Solar or nuclear energy supply installations have a very high percentage of specific energy and are limited only by the available time or lifetime. Present day solar cell installations, for example, produce almost 200 times as much energy per kg and per year than the best chemical fuels (up to about 2×10^9 J/kg year).

Electrical fuel acceleration allows almost freely selectable exit speeds for primary propulsion units in the present required range between 10-100 km/s (or I_s = 1000 - 15,000 sec.), at least for certain engines. One can only use the large specific energies of the nuclear or solar energy sources at the higher exit speeds, higher than that which can be achieved with chemical rockets*. This means that for more complex missions, there can be orders of magnitude increases in the payload fractions (as given in Fig. 2.1).

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*The specific energy actually delivered by a propulsion system

$$e_{ANT} = \frac{c_e^2}{2(1+m_T/m_R)} = \frac{F \tau c_e}{2(m_R+m_T)} = \frac{I_{spez} c_e}{2 m_{ANT}}$$

where m_T, m_R and m_{ANTR} only contains the structural mass required for propulsion, such as fuel tanks. It is clear that high specific [illegible] require correspondingly high exit speeds. For chemical rockets, we have $m_T/m_R \ll 1$, therefore $e_{ANTR} \approx c_e^2/2$. For electrical systems, we have $0.2 < m_T/m_R < 1$ so that e_{ANTR} is somewhat smaller (up to a factor of 2) than $c_e^2/2$.

On the other hand, the electrical propulsion systems have extreme performance limitations, at least as conceived today. The specific performances of the energy supply installations lies in the range 30 - 60 W/kg and in the near future it will be a maximum of 100 - 200 W/kg (see 4.1).

The specific performance decreases to about one-fourth for the engines and the energy conversion. Today the projected electrical propulsion modules will achieve about 8 - 14 W/kg referred to the gross weight. For comparison, the specific performances of the largest chemical propulsion stages are about 2,000 times as large (up to a lift-off weight of 2.5×10^4 W/kg).

The maximum possible specific thrust for the electrical systems is also very small, so that the achievable accelerations lie in the range

$$10^{-5} - 2 \times 10^{-4}\ g_o$$

This means that in the near future

1. these propulsion systems can only be used from a parking orbit,

2. in order to reach a typical Δv value, relatively long acceleration times are required, typically months or years, and

3. for many typical flight missions, for example lifting Earth satellites from a low orbit into a synchronous orbit, electrical systems will have a higher propulsion requirement (Δv) because of gravitational losses, compared with chemical or nuclear-thermal systems with a higher acceleration.

For very complex missions with a high propulsion requirement and long flight paths, for example flights to the outer planets, electrical propulsion can save a great deal of flight time. This is because with a high payload, they make available higher Δv values and therefore result in faster transfer trajectories (higher energy) than the long duration Hohmann transfers. According to Stuhlinger [77], the break-even point in time should be reached already with Mars missions.

The promising applications for electrical propulsion systems will in the foreseeable future be used for the following missions:

1. Attitude controlled small satellites:
   Resistojets, electrothermal hydrazine engines, pulsed plasma engines.

2. Trajectory control of synchronous satellites with a long lifetime ($\Delta v \approx 350 - 500$ m/s):
   Ion and pulsed plasma engines.

3. Missions with a large propulsion requirement (interplanetary missions, comet probes, "sample return", solar-polar-mission, etc.), which can barely be carried out with chemical rockets or not at all: approx. 10 - 100 kW solar electric, ion engines, 25 kW SEPS, 48 kW extended performance SEP thrust system.

4. Drag compensation of satellites near the Earth: nuclear electric.

5. Interorbital traffic LEO-GEO and back for transportation and maintenance of Earth satellites (OTV): $\Delta v \approx 12 - 16$ km/s, 25 kW SEPS.

6. Trajectory lifting, position change, trajectory control and attitude control of future satellites (multipurpose platforms, powersat, fabrication satellites, etc.): kW - MW range, ion or plasma (MPD) engines.

Electrical resistojet engines can be used for attitude control of satellites with relatively low propulsion requirements (or small ablations plasma engines) with a specific impulse of only 250 - 350 s (example: VELA reconnaissance satellite). This is because they fill the requirements for pulsed operation, minimum impulse, low ignition delay time, etc. better than, for example, diergol-chemical propulsion systems, whose advantages disappear when they are used in pulsed operation.

In order to perform trajectory maneuvers which do not have to be performed immediately, such as position control, position change, drag compensation, trajectory lifting, etc., the limited specific thrust of electrical propulsion systems is not critical. In these cases, the specific impulse of electrical (ion) engines is at least one order of magnitude greater than for chemical systems. As the satellite weight increases, this leads to weight advantages (see Fig. 3-5) even for relatively small values of $\Delta v \approx 1000$ m/s. In addition, the low thrust of the engine (mN) is advantageous because during the trajectory maneuvers the attitude control is only disturbed slightly because of the smallness of the perturbing propulsion torques.

The performance possibilities and limitations of electrical propulsion systems can be estimated from the following diagram (Fig. 4-1). This shows the relationships between the optimum exit speed and the propulsion requirement as a function of payload fraction and structure weight fraction ($\mu_{LS} = \mu_L + \mu_S$) and the required specific energy of the propulsion system $\alpha\tau$. Since the specific performance α is limited, $\alpha\tau$ is a measure for the propulsion time τ, and therefore the minimum flight time. This α contains the losses of the engines and the energy conversion. The values shown follow from the Tsiolkovski equation which has been optimized by Langmuir-Irving [35] and Stuhlinger [36] for single-stage rockets with energy sources separate from the fuel. The optimized variables are the payload fraction μ_{LS} and flight duration (or propulsion duration) τ, if we assume that the flight mission Δv is given. The exit speed c_e is varied and therefore the ratio of the fuel mass and the net

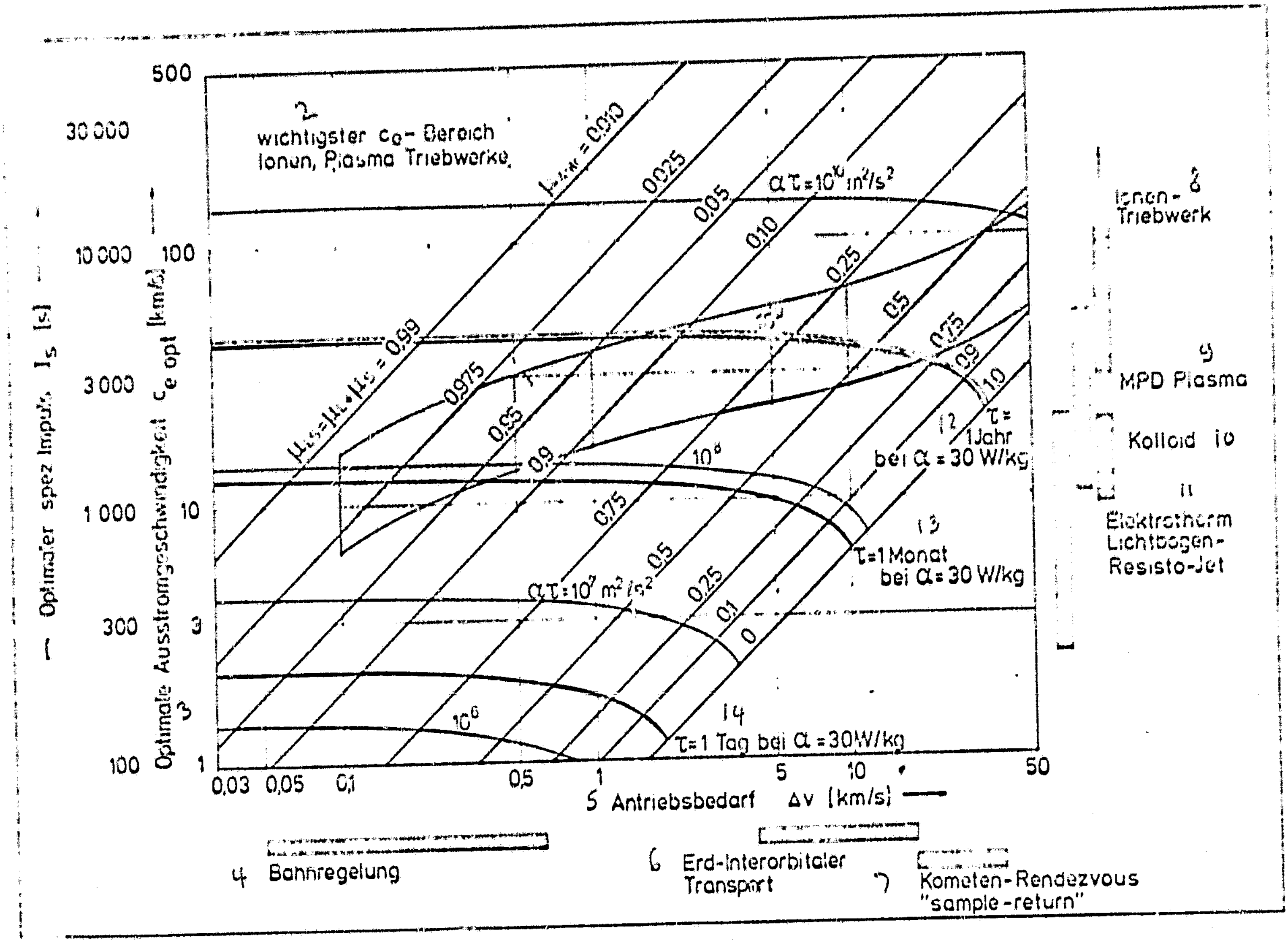

Fig. 4-1: Relationship between effective exit speed c_e, optimum propulsion requirement Δv specific energy ατ and the sum of the payload mass ratio and structural mass ratio μ_{LS} for optimized propulsion systems.

1 - optimum specific impulse; 2 - most important c_e range. Ions, plasma engines; 3 - optimum exit speed; 4 - trajectory control; 5 - propulsion requirements; 6 - Earth-interorbital transport; 7 - comet rendezvous; 8 - ion engine; 9 - MPD plasma; 10 - colloid; 11 - electrothermal light arc resisto jet; 12 - 1 yr. for α = 30 W/kg 13 - τ = 1 mon. for α = 30 W/kg ; 14 - τ = 1 day for α = 30 W/kg

thrust mass (m_T/m_W).

Application of this optimization procedure to flight missions is therefore very rough, because in practice first of all the specific performances α usually vary during the flight (distance to the sun), and secondly in practice the thrust as well as the exit speed will vary during the flight (just like for chemical and dual mode nuclear propulsion systems). Thirdly, the payload will differ for the outbound and return leg of the flight.

In spite of this, this rough optimization gives a good overview about the general relationships, especially for the upward and downward spiraling of LEO - GEO - LEO, where τ also represents the flight direction. From the diagram in Fig. 4-1, we can see the following trends:

1. With an electrical stage, one can achieve almost arbitrarily high velocity values even with good payloads (for example, Δv = 50 km/s) if sufficient time is available (for example, for research missions with unmanned probes).

2. For each given mission (Δv), payload fractions can be traded off against time. For example, for satellite lifting LEO - GEO $\Delta v \approx$ 6 km/s, we find that $\mu_L + \mu_S$ = 0.25 for one month or 0.75 for one year, for specific impulses of 950 sec. or 4500 sec. In the first case, the fuel consumption is much higher and in the second case the flight time is much higher.

3. The shaded band shows that for a relatively small range of exit speeds (about 20 - 80 km/s) one can satisfy a wide range of flight missions, from Δv = 500 m/s for north-south trajectory control up to very high energy planetary probes with Δv > 50 km/s. For a synchronous satellite the mass fraction for the entire propulsion system μ_{ANT} is about 5 - 6%, and for a probe it is more like 75%.

4. The diagram also shows the flight times τ for a given specific performance (α = 30 W/kg). For constant payload the propulsion times increase according to $(\Delta v)^2$ as expected.

The following are important for evaluating, selecting and cost optimizing electrical propulsion systems, in addition to the payload fraction and the flight time:

1. The electrical energy supply installation often is counted as completely or partially payload at the point of arrival or also in between (power sharing). This means that the effective specific performance of the propulsion system is increased by about a factor of 2 and the cost fraction decreases.

2. For repeated flights (interorbital transport), the higher initial costs over many flights and the lower fuel mass which has to be transported into LEO become more important factors. However, the flight time also becomes important, because the number of units for a given yearly transportation mission is determined by this. Therefore, the chemically accelerated return flight of empty electrical "tugs" can be considered.

Since the performance limitations of electrical propulsion systems depend so strongly on the energy supply units in space, we will now give a short summary of them.

4.1 ENERGY SUPPLY INSTALLATIONS

The availability of electrical energy on the space vehicle is a requirement for any electrical propulsion unit. Depending on the thrust requirement, it is between several watts up to several MW.

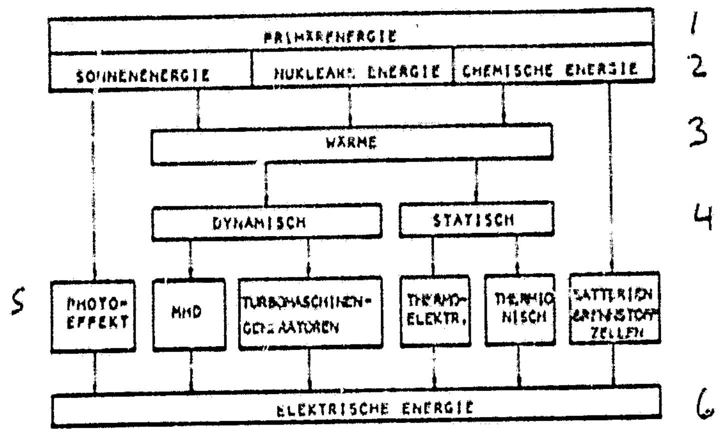

Fig. 4-2. Methods for converting primary energy into electrical current.
1 - Primary energy; 2 - solar energy - nuclear energy - chemical energy; 3 - heat; 4 - dynamics - static; 5 - photoeffect - MHD - turbomachine generators - thermoelectric - thermionic - batteries fuel cells; o - electrical energy.

The production of electrical energy in space vehicles can be done in many ways. Fig. 4-2 gives several possibilities in the form of a block diagram.

Two basic types of energy supply installations must be distinguished:

a) the energy source is carried along,

b) the energy is introduced from the outside (up to the present time only in the form of solar energy, later on possibly using laser radiation or microwave radiation from the ground or satellite generating stations).

The conversion of primary energy into electrical energy is either done directly using the photoeffect in solar cells or chemically-electrically in galvanic elements (batteries and fuel cells). Or this can be done indirectly using thermal energy. The conversion of heat energy into electrical energy can be done statically using

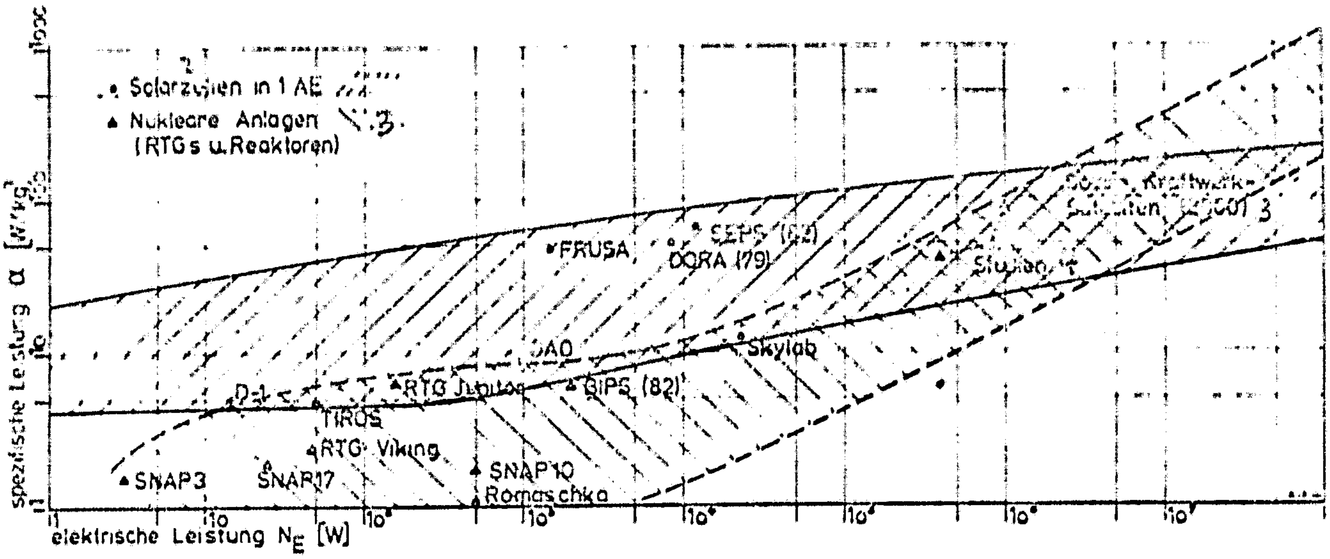

Fig. 4-3: Specific performance of energy supply installations for space applications [37 - 40] (alpha here refer to the mass of the energy supply installation).

1 - specific power α [W/kg] 2 - electrical power; 3 - solar power plant satellite (2000); 4 - JPL studies

thermo elements and thermionic elements, or it can be done dynamically using the kinetic energy making use of MHD generators or turbo-electrical heat machines. For example, there are gas turbines (Brayton processes) and steam turbine processes (Rankine cycles).

The waste heat must be removed in every case using radiation coolers. In these systems, the energy converters have to be carried along and this limits the specific performance of the system. Only two candidates have been accepted from many possible concepts for propulsion energy supply units for the near future: solar cells and nuclear installations (radioisotope batteries (RTG) and reactors). Chemical energy sources cannot be used as primary energy sources for electrical propulsion because of the low specific energy. Chemical batteries and regenerative fuel cells are used in solar installations as storage units during dark periods. The reasons the other possible systems are not considered is because of the simplicity of maintaining the systems and the proven lifetime as well as the exceptional improvements in solar cells as far as weight, efficiency and costs are concerned. These will be reduced even more in future mass productions by several orders of magnitude. In addition, photovoltaic installations do not require the very accurate parabolic mirrors required for solar thermal installations, nor do they have to be exactly aligned towards the sun, which places severe requirements on the attitude control system.

Fig. 4-3 shows the specific performances as a function of performance for such installations, for present day and future concepts. We can see that based on the requirement for a high α, only solar installations are candidates for propulsion today. In the future, newly developed reactors will be more favorable for many propulsion tasks, because in addition to their specific performance they are independent of the solar radiation density, in other words they are independent of the dark periods for Earth satellites and for planetary missions far away from the sun.

For the far future, we can distinguish the following three ranges [40, 41]:

1. Solar cell installations up to about 100 kW (500-1000 m^2) for the region near the Earth and for researching the close planets, comets and the sun (SEP, Fig. 4-5).

2. Nuclear installations above approximately 100 kW to several MW for large satellites and large propulsion tasks in interplanetary space.

3. Solar energy satellites in the gigawatt range with electrical propulsion in the megawatt range for trajectory and attitude control and transport.

4.2 ELECTROSTATIC ENGINES

In these engines the charged particles are accelerated in an electric field which is constant in time. In order to avoid satellite charging, the jet has to be electrically neutralized by means of an electron-emitting electrode (or a plasma neutralizer). Using this engine, one can achieve the highest jet speeds at present.

Electrostatic engines differ essentially in the production of the ions or charged drops:

Direct current gas discharge with electrodes (electron collision ionization)	Kaufman ion engine
Alternating current gas discharge without electrodes	Radio frequency ion engine
Duoplasmatron light arc	Duo Plasmatron ion engine
Ionization of cesium on a hot constant surface	Contact ionization engine
Field ionization on sharp wetted edges or tips	Field emission ion propulsion
Charging of droplets on capillary tips or edges	colloid engine

In the first three engine types, first a neutral plasma is produced and the ions are extracted and accelerated. In the other ones the ions or charged droplets are produced on a surface and are accelerated.

The comparison parameters and their possibilities of improvement are the following:

1. Lifetime	Reduction of the neutral gas fraction (at present 5-20%) in order to avoid grid erosion after charge exchange

2.	Reliability	Ground tests and flight tests
3.	Efficiency	Reduction of the required energy (200-600 eV/ion) of the neutral gas fraction and the velocity distribution
4.	Current density	Increase of the suction field strength for a specified voltage
5.	Concept simplicity	Direct field emission of the fuel
6.	Overall system weight	Simplification of energy conversion installation

In addition, we have, depending on the mission:

7.	Switchability	Reduction of heating times (for attitude control)
8.	Fuel compatibility (for large thrust, near the Earth, OTV and large satellites)	Argon, Xenon, nitrogen, glycerin as fuel
9.	Satellite compatibility (especially for scientific satellites)	Electrical and mechanical shielding

In most new evaluation studies, Kaufman engines or radio-frequency engines are selected. Lifetimes of up to 15,000 hours have been demonstrated in ground tests and 20,000 hours are possible (additional date in Table 4-2).

Since 1962, at least 13 flight tests with ion engines have been carried out. Nine Cs contact engines, 3 Hg- and 2 C_s electron collision ion engines have been test flown in 9 flight tests in the USA. At least 4 ballistic flight tests with electron collision ion engines with Ar, N_2 and air have been carried out in the USSR. Several of them were successful and demonstrated the following:

1. the expected thrust,

2. complete neutralization of the jet (no electrical charging of the satellite was measured), however, they were unexpected difficulties during the operation in the gravity-free space.

In the case of the Kaufman engine or electron collision ion engine, the ions are produced in a low pressure gas discharge and are sucked away in an electrostatic field between the discharge vessel and the acceleration grid, and then they are accelerated. The development time up to a useable unit amounted to 20 years.

In 1977, a comparison study between solar cell and the electrostatic propulsion led to a clear recommendation for electrostatic propulsion. As a consequence of this, NASA now is supporting a long-time low thrust system. In the 1979 budget, there is an item for the initiation of an electrical primary propulsion system for NASA/s planetary exploration program.

The data of the Kaufman eingine selected for this [45] are the following:

Specific impulse:	5000 s
Thrust:	225 mN

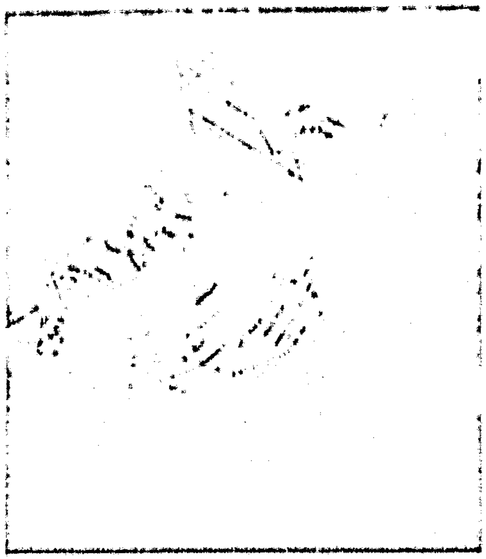

Fig. 4.4: Kaufman engine (Hughes) 30 cm diameter

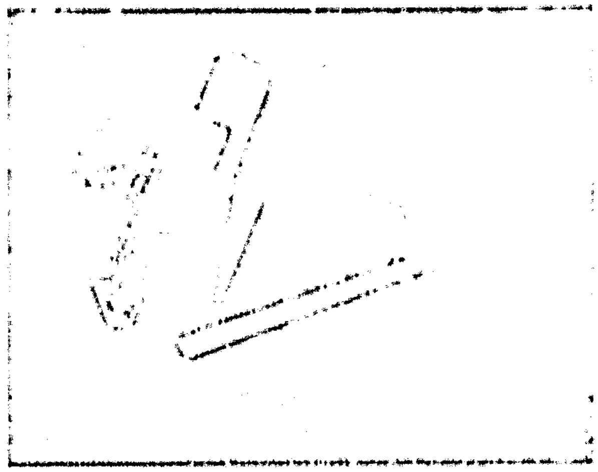

Fig. 4-5: Propulsion unit of 5 standard modules 25-50 kW SEPS

Performance:	6.4 kW
Specific weight:	14 kg/kW
Engine diameter	30 cm

Fig. 4-4 shows previous models of this motor.

Each of the two selected 30 cm engines are collected into a propulsion module together with an energy conversion unit system. Depending on the flight mission, several modules will be put together into a propulsion stage [44]. As an example of an important mission for such a concept, we can consider the rendezvous with the comet Tempel II. Fig. 4-4 shows such a propulsion stage with 10 engines. For flight tasks in the far distant future, engines with diameters between 50 cm to 150 cm have been tested. [46] gives a parametric study on ion engines of various sizes with different fuels. A 50 cm diameter engine with mobile gas fuels is suggested as the cost optimum solution, from many points of view, and as the next stage of ion engine development. 120 cm engines (164 kW) using argon as a fuel are being suggested for transporting power plant satellites [93].

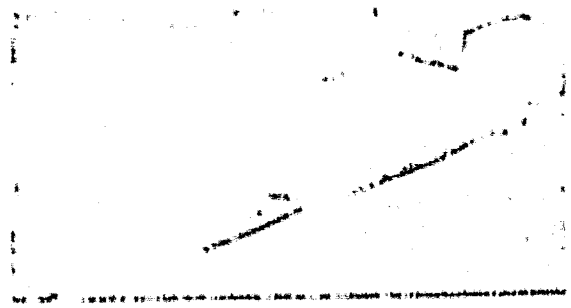

Fig. 4-6: Kaufman engine system (8 cm diameter, Hughes)

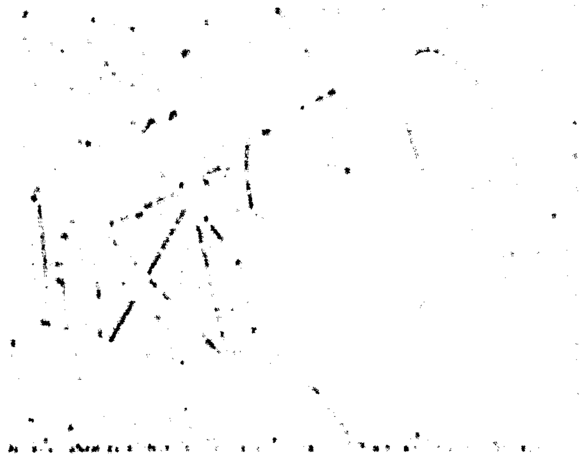

Fig. 4-7: Radio frequency engine

In the secondary propulsion system area, NASA's goal is to develop and flight qualify an 8 cm Kaufman eingine system [47] (Fig. 4-5).

A flight test is planned from the shuttle in 1981 and on the Air Force satellite STP P 80-1.

In 1982 and 1983, only five satellites are possible candidates for using this secondary propulsion system. However, it would also be suitable for stabilizing large space stations such as the geostationary platform.

The data are given in Table 4-2.

In Germany, a radio frequency ion engine, the RIT 10, is being developed by the University at Giessen in collaboration with the DFVLR and MBB [90], Fig. 4-6. In this type, the ionization of the gas is done in a high frequency gas discharge without an electrode. This propulsion system is at the present time in the final stage of development and is undergoing lifetime tests and flight compatibility tests. It will be ready to fly in the next year. The characteristics are given in the summary Table 4-2. ESA plans to flight test this unit in 1983 on the European heavy satellite.

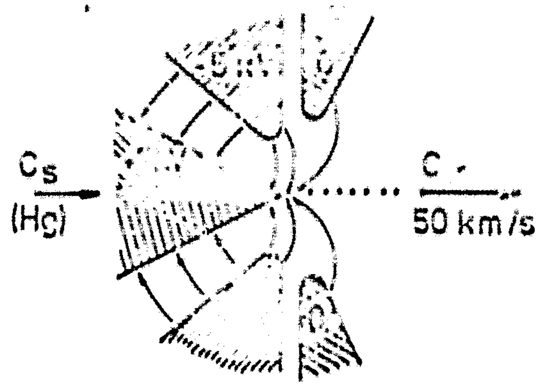

Fig. 4-8: Principle of field emission engine

Because of the high state of development of both systems and the comparable flight times, it is possible to compare the weights of the components with those of the Hughes 8 cm engine system.

Components

Thrust Performance	Hughes 8 cm	RIT-10
	4.5 mN 175 W	5-10 mN 270 W
Engine	2.3 kg	4.4 kg
Gimbal	1.5 kg	
Fuel supply tank	2.7 kg	0.8 kg
Fuel	8.4 kg	5.0 kg
Energy conversion	10.1 kg	7.0 kg
	25.0 kg	17.2 kg

TABLE 4-1. Weight comparison of the total Hughes 8 cm and RIT-10 systems.

The table clearly shows the great proportion for energy conversion.

RIT engines with 35 cm diameter acceleration grids (RIT 35 and RIT 35 L) are being developed as RIT type main engines. These engines have performances between 3.2 and 4.5 kW for a thrust between 0.14 and 0.2 N. Hg and later on Xe are planned as fuels (see Table 4-2).

A 5 cm Kaufman engine is planned in tests for 1982 in Japan on the European Test Satellite III.

The Kaufman engine T 5 has been developed in England. However, no flight test is planned at the present time.

In France all activity was concentrated on the contact ionization engine, but was given up later on (see Table 4-2).

Field emission propulsion is uncomplicated electrical propulsion method which expels metal ions with a relatively high speed. A large voltage produces a high electrical field along a sharp edge, which is wetted by a liquid metal. The surface of the metal (for example cesium) is deformed into small needles in this field. At the tips of these needles the field becomes so strong that the metal ions are torn out directly and are accelerated (Fig. 4-8).

Field emission propulsion with Cs is being developed essentially by the ESA [49], which has decided on the development program up to the end of 1983.

In conjunction with this, the University of Stuttgart is performing basic research on a field emission propulsion unit with mercury as a fuel.

Colloid engines were developed at TRW and other USA research laboratories and in Europe at the ESA. However, they were abandoned because of their low specific impulse. Today this concept is being developed further by Phrazor Technology at the request of the U.S. Air Force.

The following Table 4-2 gives a summary about the more recent electrostatic propulsion units of western countries. Even though Cs contact ion engines were first developed in the USA and were used the most in flight tests, they are not shown in this table. Development on them ceased at the end of the 60s.

Typ 1	2 Treibstoff	3 Spez. Impuls s	4 Wirkungsgrad	5 Schub mN	6 Lebensdauer h	7 Entwicklungsstand
Kaufman SERT II 1970 NASA 1975	Hg	5600 5000	.67 .65	23 26	6000 215 Zyklen 8	Flugversuch 9
Kaufman ATS-6 1974 NASA	Cs	4000	.60	4.5	kein Wiederstart 10	Flugversuch 9
Kaufman 5 cm Ø Japan	Hg	3100	.44	2.1	-	Flugbereit 1981 11
Kaufman 8 cm Ø Hughes	Hg	2600	.56	5	15000 20000 gesch. 12	Flug 1981 13
Kaufman 30 cm Ø Hughes	Hg, Ar	3000	.70	130	6000 23000 gesch. 12	Lebensdauerversuche 14
Kaufman 120 cm Ø Boeing	Ar	9000	.73	3500 (164 kW)	(200 Tage) 16	Studie 15
Kaufman T5 England	Hg	3000	.66	10.2	im Test 17	
RIT-10 Deutschland 22	Hg	3200	.54	5-10	6000 1100 Zyklen 8	18 Flugbereit 1979
RIT-35 35 cm Ø Deutschland 22	Hg	3300	.72	143	im Test 17	Experimentalmodell 19
Feldemission 23 ESA	Cs (Hg)	10000	.85	100mN/m	-	Komponentenentwicklung 20
Kolloid Phrasor, USA	Glyzerin	1200	.70	4-5	10000 gesch.	
CCI,5 Kontakt-Ionisation Frankreich 24	Cs	6700	.52	1.5	-	aufgegeben 21

TABLE 4-2: Data for new electrostatic engines [42, 43, 48].

1 - Type; 2 - fuel; 3 - specific impulse; 4 - efficiency; 5 - thrust; 6 - lifetime; 7 - state of development; 8 - cycles; 9- flight test; 10 - no restart; 11 - ready for flight; 12 - est.; 13 - flight; 14 - lifetime experiments; 15 - study; 16 - 200 days; 17 - tested; 18 - ready for flight; 19 - experimental model; 20 - component development; 21 - given up; 22 - Germany; 23 - field emission; 24 - 5 contact ionization France

4.3 ELECTROTHERMAL PROPULSION

One kind of electrothermal propulsion unit (resistojet) uses electrical resistance heating for increasing the enthalpy of the fuel and therefore the specific impulse. The structure is very simple: a thermally shielded rhenium tube is heated with ohmic heating up to 2600°K. The fuel flowing through, such as H_2, NH_3 or N_2, is heated convectively and is relaxed in a following nozzle. Compared with electrostatic propulsion systems, we can distinguish the following:

Advantages:

-- well suited for pulsed operation
-- simplicity of the total system (no energy conversion)
-- high reliability
-- high thrust density
-- high state of development: flight tested
-- high efficiency (70-80%) for stationary operation.

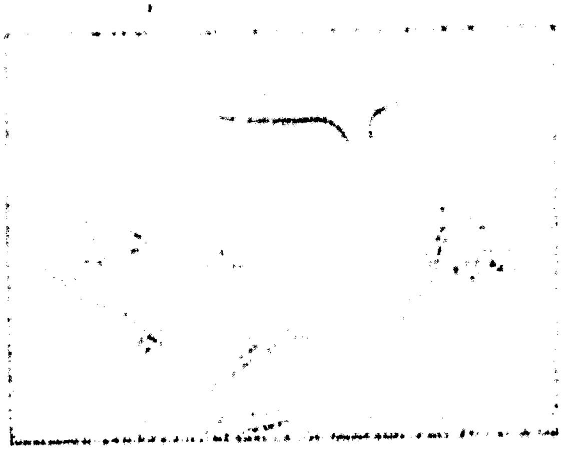

Fig. 4-9: Pulsed NH_3 resistojet (AVCO)

Disadvantages:

-- low specific impulse, limited by wall temperature (320 s with NH_3, 670 s with H_2)
-- low lifetime 8,000 h

Because of these advantages, these engines were used for attitude control in the early 60s and successfully over a dozen Vela satellites (see Fig. 4-9).

The data for the system are as follows:

Input power 6.5 W

Thrust	133 μN,	445 μN,	1.3mN,	4.5 mN
Specific impulse	150s,	180s,	180s,	165s

Because of the high thrust density, resistojets with a 3 kW input power are being built both in England and at Marquardt in the USA. Hydrogen is being tested to increase the specific impulse. A specific impulse of 670 to 800 s, a thrust of 0.65N and an efficiency of about 77% has been achieved. Storage problems for hydrogen are one disadvantage.

Today the development is concentrated on electrothermal hydrazine propulsion (HiPEHT) Fig. 4-10. Hydrazine (N_2H_4) is stored in liquid form and decays into a mixture of N_2, H_2 and NH_3 in a heating chamber when there is a platinum wire not in it. This mixture is then heated in the following vortex heat exchanger to 2000°C and then is expelled to a nozzle [50].

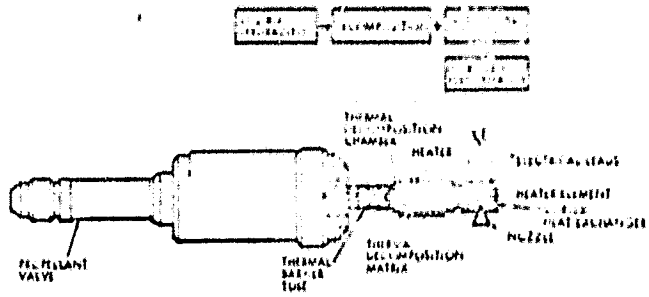

Fig. 4-10: Electrothermal hydrazine propulsion (HiPEHT)

In the USA this type is being developed by TRW for Comsat's Intelsat V:

Specific impulse	220 s without heat exchanger heating
	330 s with "
Thrust	180 - 450 mN
Specific performance	1.3 W/mN

In Europe, ERNO under contract to the ESA and the Gfw is working on the electrothermal hydrazine engine (EHT) [94]. Two engines have been developed: one for the thrust range 0.5 - 2N and one for 0.08 - 0.35 N. The desired specific impulse is 300 s.

A weight comparison of the overall system for N-S trajectory control of the Intelsat V shows that the disadvantage of a low specific impulse is important in spite of the advantages. 191.8 kg are required with HiPEHT. 57.2 kg [51] are required for the 8 cm diameter Hughes ion engine for a lifetime of 7 years.

4.4 ELECTROTHERMAL AND ELECTROMAGNETIC LIGHT ARC PROPULSION SYSTEMS

The engines of this type are dormant at the present time but will be discussed in more detail here. They have a relatively large power density and could be used for future large satellite projects, and they have advantages compared with ion engines.

ELECTROTHERMAL LIGHT ARC ENGINE

The light arc engine is an engine which has been analytically and experimentally investigated for over 20 years. Its principle is based on heating the propellant gas using a light arc in a combustion chamber and there is subsequently relaxation through a nozzle. Since the light arc temperatures extend between several thousand to several tens of thousand degrees Kelvin, depending on the gas type, specific impulses of several hundred to even over one thousand seconds can be reached in the exit jet. One disadvantage of this propulsion system is that the gas dissociates at these relatively high temperatures and partially ionizes. The dissociation and ionization energies, however, mostly cannot be recovered during expansion (frozen flow). Therefore, the efficiency decreases for various gases and certain characteristic specific impulses, to less than 40% depending on the combustion chamber pressure (see Fig. 4-11).

As I_s increases, the efficiency again increases because as the combustion chamber temperature increases, the relative fraction of the reaction heat to the total onthalpy of the propellant gas decreases. Table 4-3 gives a qualitative evaluation of various possible fuels.

The efficiency which can be achieved at high pressures and for exit speeds of about 10,000 m/s is very favorable. This means that this type of propulsion could be considered for certain future missions such as the OTV (Orbital Transfer Vehicle).

As an example of a regeneratively cooled 30kW light arc engine, Fig. 4-12 shows a machine developed at Giannini [53] with a combustion chamber pressure of about 1 bar.

Lifetime tests with a radiation cooled light arc engine have resulted in times to about 1000 hours without noticeable wear of the electrodes and the nozzle.

	1	2	3	4	5	6	7	8
H_2	••	•	•	•/0	•	•	-	1
LiH	•	0	0	•	••	•	••	4
Li	•	•	•	•	•	•	••	7
NH_3	•	-	•	•	•	•	•	4,2
N_2H_4	•	-	•	•	•	•	•	5,3
N_2	0/•	•	•	0	•	•	-	14
Ne	0	•	0	0	•	•	-	20
Na	0	-	•	•	•/0	•	••	23
Ar	0	0	•	0	•	•	-	40
Xe	-	0	-	0	•	•	-	131
Cs	-	0/-	-	•	-	-	0	133
Hg	-	-	-	•	•	-	0	201

*) sehr gut ••
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Table 4-3: Qualitative evaluation of various propellants for light arc propulsion systems

1 - specific impulse; 2 - for; 3 - availability; 4 - storable in space?; 5 - manipulation; 6 - environmental compatability; 7 - regenerative cooling capacity: 8 - average atomic mass (dissociated) 9 - very good, good, average, poor, very poor

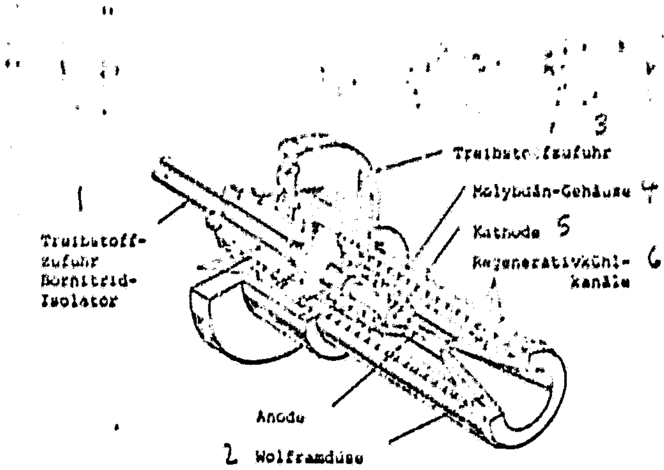

Fig. 4-12: Regeneratively cooled 30 kW light arc engine. Fuel :H_2, Thrust: 3.35N, spec. Impulse: 1010 sec., Thrust deficiency: 54%, combustion chamber pressure: about 1 bar

1 - fuel supply. Boron nitride insulator; 2 - tungsten nozzle; 3 - fuel supply; 4 - molybdenum housing; 5 - cathode; 6 - regenerative cooling channels

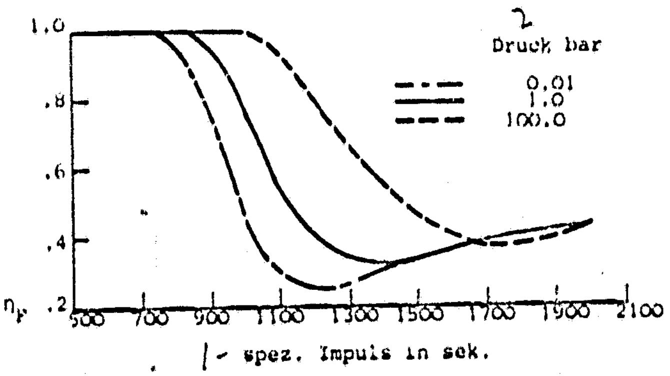

Fig. 4-11: Efficiency of frozen flow as a function of specific impulse for hydrogen at various combustion chamger pressures [52].

1 - specific impulse in seconds; 2 - pressure

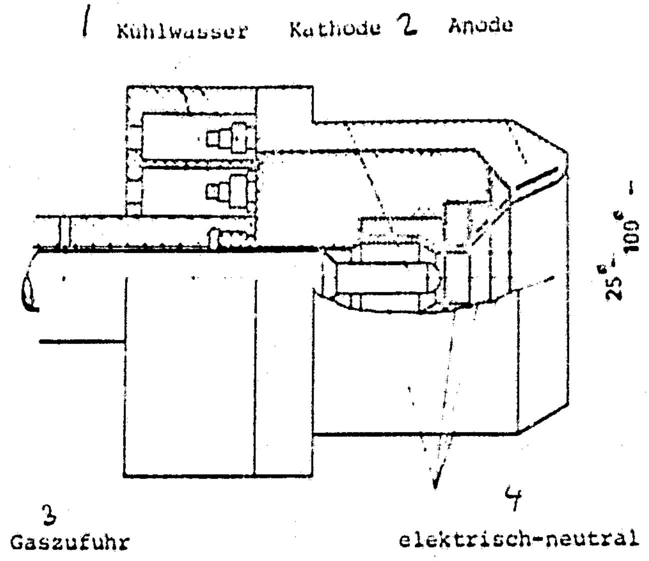

Fig. 4.13: Schematic of an eigenfield accelerator EM 25-100 of DFVLR Stuttgart

1 - cooling water; 2 - cathode; 3 - gas supply; 4 - electrically neutral

Name	1 Herkunft	2 Geometrie: 3 Magnetfeld	4 Kathode	11 el. Leistg. N_e [W]	12 Entladungsstrom I [A]	13 Massendurchsatz $\dot{m}$ [kg/s]	14 Treibstoff	15 Schub F [N]	16 eff. Austrittsgeschw. c_e [km/s]	η eff
HIT A 4	DFVLR Stuttgart	radial	strom- abwärts 5	400	0,3	$7 \cdot 10^{-7}$	Hg	$13 \cdot 10^{-3}$	21	24 %
Low Power MPD engine	NASA Lewis	divergent	strom- abwärts 5	625	1,8	$8 \cdot 10^{-7}$	Xe	$14,8 \cdot 10^{-3}$	18,5	21 %
X - 16	DFVLR Stuttgart	divergent	strom- aufwärts 6	$6 \cdot 10^3$	60	$1 \cdot 10^{-5}$	Ar	0,2	20	33 %
Langley Hall current engine	NASA Langley	divergent	strom- aufwärts 6	$24 \cdot 10^3$	600	$3,4 \cdot 10^{-5}$	Ar	0,5	15	16 %
JAC-3	EOS Pasadena	divergent	strom- aufwärts 6	$34 \cdot 10^3$	300	$9 \cdot 10^{-6}$	Li	0,5	48	35 %
7 Russische	9 Entwicklung	? ?	? ?	500 $3 \cdot 10^3$	? ?	$3 \cdot 10^{-6}$?	Xe Ka	$15 \cdot 10^{-3}$?	8 30	50 % ? ?
8 Japanische	10 Entwicklung Univ. Tokio	divergent	?	?	250-1000	?	17 Ar oder NH_3	?	8 35	20 - 30 %

TABLE 4-4; Characteristics of MPD engines with foreign fields.

1 - origin; 2 - geometry; 3 - magnetic field; 4 - cathode; 5 -downstream of current; 6 - upstream of current; 7 - Russian; 8 - Japanese; 9 - development; 10 - development; 11 - electrical power; 12 - discharge current; 13 - mass throughput; 14 - fuel; 15 - thrust; 16 - effective exit speed c_e; 17 - or

MAGNETOPLASMADYNAMIC ENGINES

The gas is ionized by the very high light arc temperatures and can again be accelerated using electromagnetic (Lorentz) forces. Propulsion systems where the electromagnetic fraction is dominant in the gas or plasma acceleration compared with the thermal acceleration are called magnetoplasmadynamic engine systems or MPD systems. They operate at a relatively low combustion chamber pressure ($5 \cdot 10^{-3}$ - - 0.5 bar) and can be tested only under very good vacuum conditions with reliability because of the interaction between the light arc discharge and the surrounding gas. According to the operational state (stationary, quasi-stationary, pulsed) and their acceleration mechanism (eigenfield, foreignfield) the MPD propulsion systems are classified into various categories. Quasi-stationary pulse durations mean durations of 1 ms and longer.

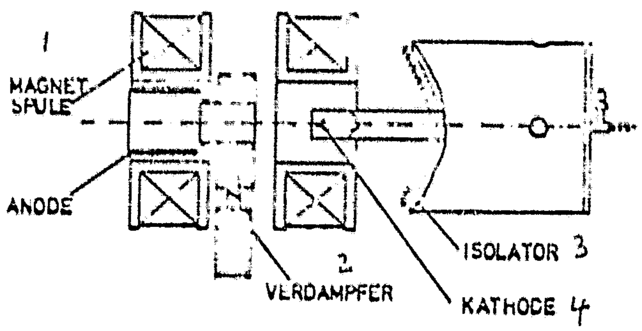

Fig. 4-14: Diagram of the Lithium MPD engine [62] built by EOS

1 - magnetic coil; 2 - vaporizers; 3 - insulator; 4 -cathode.

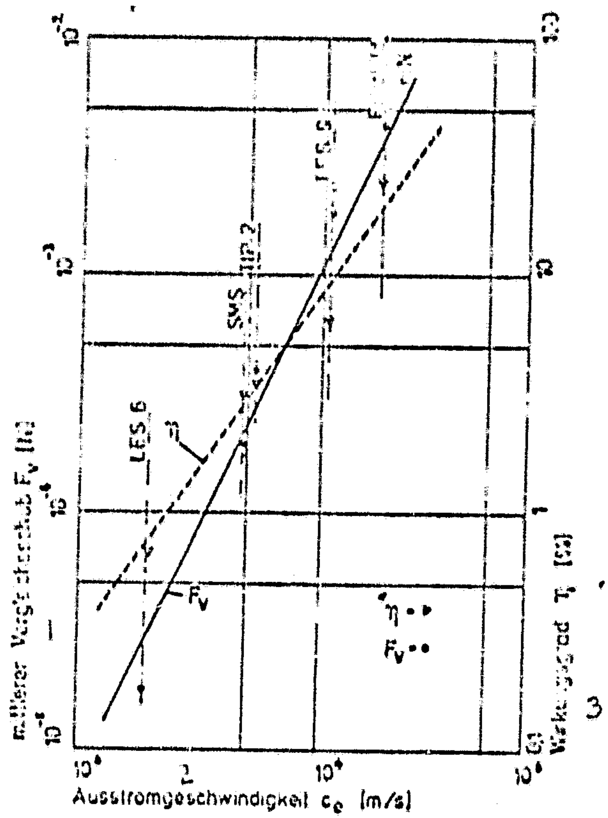

Fig. 4-15: Average comparison thrust, efficiencies and effective exit speed of pulsed Teflon engines

1 - average comparison thrust; 2 - exit speed; 3 - efficiency

STATIONARY AND QUASISTATIONARY EIGENFIELD ACCELERATORS [54, 55, 56].

One example of this type is the eigenfield accelerator EM 25-100 developed at the DFVLR Stuttgart (see Fig. 4-13). Between a ring anode and a pin cathode, a low pressure light arc discharge occurs. The discharge current induces circular magnetic fields which in turn interact with the current which generatres them, and this accelerates the plasma carrying the current.

The electromagnetic thrust component depends only on the geometry and the current intensity ($\sim I^2$) and is only important at high current intensities. In order to exploit the electrothermal thrust component, the device is built in the form of a nozzle. This means that the thrust cannot be arbitrarily increased with current intensity, but is limited by a critical value of $(I^2/m)_{crit}$. This value corresponds to a maximum specific impulse of about 2 - 3000 s [57].

In the device discussed above, for 6000 A and a mass throughput of 1.5 g/s argon, a thrust of 17.3 N was measured at a c_e of 11.5 km/s and an overall efficiency of 22%.

The quasistationary eigenfield acceleration devices have similar geometries as the stationary devices based on the fact that the physical processes are the same. The heat loads are smaller in them and therefore they can be operated at higher currents (20-30kA), which leads to more favorable efficiencies because of the substantially higher MPD acceleration fraction [58].

FOREIGN FIELD ACCELERATORS

In the case of MPD propulsion systems with low currents, the electromagnetic acceleration part is not sufficient because of the low eigen magnetic field. Using a coaxial foreign field, this electromagnetic acceleration part can be increased. Because of the Hall effect of the crossing current density lines, when the magnetic field is applied, azimuthal ring currents are induced. Their interaction with the applied field produces the accelerating Lorentz forces.

Table 4-4 shows several such machines with typical configurations.

The first two devices with low currents operate in the glow discharge range. The plasma density is accordingly lower than for the other types. Both of them have a cathode downstream and outside of the accelerator itself. They differ in the configeration of the applied magnetic field. The HIT A4 (abbreviation for Hall Ion Engine) [59] has a radial field in the acceleration space caused by the coaxial ferrite cores. The Lewis unit [60] produces a divergent magnetic field by means of two coils concentered with the engine axis, similar to the three next devices [61, 62, 63]. These devices operate in the light arc range and have substantially greater power uptake. The thrust and mass throughput is accordingly larger. The fuels which could be used for these engines include easily ionizable alkali metals such as potassium and especially lithium. Fig. 4-1[illegible] shows a typical Li system built by EOS.

The accelerators described, including the Japanese MPD accelerator [48], were developed as laboratory units. The Russian engines of this type [64] have been flight tested since 1971.

PULSED PLASMA ENGINES

In contrast to the large stationary and quasi-stationary plasma engines, pulsed unsteady plasma engines (discharge time μ s) have been available as auxiliary propulsion units for attitude control for several years. These are small simple devices which use electrical discharge of one of several condensers for the thermal or electromagnetic acceleration of the fuel. The fuels usually used is the insulator Teflon, which ablates by means of a discharge through its surface and is then ionized.

Fig. 4-16: The mN engine built by Fairchild for North-South trajectory control

Advantages of this attitude control engine includes the possibility of producing precise and reproducible pulses for an effective exit speed of several thousand meters per second. The efficiencies are small but do increase with the impulse. Fig. 4-15 shows the efficiencies and average comparison thrust F_v as a function of the effective exit speeds c_e [65, 66, 51].

These are Teflon engines which have been built and partially flown and used for attitude control of satellites. The mN engine is used for north-south trajectory control and will be flight tested in 1980 or 1981. It is shown in Fig. 4-16.

A Japanese pulsed MPD engine with argon as the propellant is planned for a flight on Spacelab in 1981 [67].

4.5 MAGNETIC COILS

Attitude control can be done using three magnetic coils installed in the spacecraft control axes of an Earth satellite, and a thrust system can be avoided. This is because they can interact with the magnetic field of the Earth and provide the desired torque.

The passive magnetic stabilization methods (for example, AZUR [68]) has been installed in the meantime in a number of Earth satellites. This includes the German AEROS satellite [69], the ANS, SAS-C, RAE, OAO, OSO-7, LES-5 and the NAVSTAR navigation satellites of the "Global Positioning System" as well as the RCA SATCOM communications satellite [70 - 72]. The magnetic attitude control derives its energy from the environment (solar energy). In other words, there is no fuel consumption and the related contamination of the satellite surroundings caused by the engine gases. Other advantages include a theoretically infinite lifetime because there are no moving parts. The system is also very simple and highly reliable. The system weight can be lower than the corresponding attitude control system using thrusting nozzles, especially for longer missions [71]. There is also the advantage of continuous and variably adjustable torques, which has to be contrasted against the impulsive character of a thrusting system. Also, "pure" torques are generated without any perturbing force components. Magnetic attitude control is always a possibility for flight missions with high requirements for maintaining a certain trajectory, such as for example navigation satellites [72]. In this case the torques are produced with thrusting nozzles which are activated in pairs because of the fact that there are always thrust differences which lead to small disturbing forces and therefore to trajectory disturbances.

For present day orientation accuracy requirements for a satellite, magnetic coils are usually combined with other systems for attitude control, consisting of momentum storage units (flywheels, reaction flywheels, torque gyroscopes) and torque generators. The first American three-axis stabilized communications satellite RCA SATCOM is a typical example of this. Its attitude control system uses a flywheel, magnetic coils and hydrazine engines.

Systems which have been built have dipole moments of about 7.5 - 120 Am^2 depending on the trajectory altitude, and produce torques in the range between $0.25 \cdot 10^{-5}$ Nm to $4 \cdot 10^{-3}$ Nm, and the electrical consumption is between 0.2 W to 12 W per magnetic coil.

5. NUCLEAR-THERMAL, NUCLEAR-ELECTRIC AND COMBINED PROPULSION SYSTEMS

Energy sources used at the present time for nuclear rockets are either nuclear fission reactors or radioisitope decay reactors. There is also the very futuristic concept of a bomb propulsion unit (Chapter 6). For these nuclear fuels, there is no thinkable analogy with the chemical rocket (in other words, the fuel is used as a propellant). This is because the nuclear fuels decay too slowly and are much too expensive. Only the path through the thermal energy remains for practical applications, either the direct heating of the fuel (H_2, NH_3) in the reactor (nuclear-thermal rockets), or the path heat- electrical power-thrust, which applies for nuclear-electric rockets.

For the exit speeds of about 8500 m/s achievable today, the potentially very high specific energy of the reactor is not completely demonstrated (also, its costs are very high) (see footnote, Chapter 4, page 21). This is especially true when a great deal of fuel is heated by the reactor, for example for an often recurring transportation mission near the Earth (shuttle operation) [75].

In the case of nuclear-electrical propulsion systems, the exit speed is much better matched to the specific energy of the reactor. One handicap is the very limited specific thrust, which is the case for all electrical systems of the near future, which limits the acceleration to a maximum of about $10^{-4} - 10^{-3}$ g_o (maximum).

A good compromise seems to be the dual mode concept, which uses the same reactor alternately for a high thrust or electrically to produce a high specific impulse, and also produces substantial electrical power (for example, 100-1000 KW) for other purposes.

5.1 THERMAL FISSION ROCKETS

SOLID CORE REACTOR

A nuclear fission reactor is the energy source and its heat is directly supplied convectively to a propellant gas. In practical projects, the exit speeds of a nuclear thermal rocket using hydrogen as a fuel are limited because of the strength of the separation wall around the fission zone (shielding, moderator) to about $c_e \approx 8500$ m/s. The maximum permissible material temperatures are between 2300 - 2600°K at the present time.

After preliminary research (reactor project Rover and Kiwi), the American project NERVA (Nuclear Engine for Rocket Vehicle Application) developed a hydrogen engine with a thermal power of about 1575 MW, a thrust of 334 kN for a specific impulse of 825 s. The gas temperature at the nozzle inlet was 2360°K and the gas pressure was 31 bar. The dry weight of the engine was 38,000 kg. The specific power therefore would be 3.5×10^4 W/kg without shielding for the crew. The initial acceleration was $1.7\ g_o$.

The thrust can be throttled to 90% [98, 99]. Application studies [75] considered stages with a lift-off weight of about 196 tons (with 136 tons of hydrogen), for example for LEO-moon orbit shuttle missions. NERVA stages can be used as first stages (about 240 tons) from LEO for manned 50 MW solar electric Mars research ferries [77].

In the American project Mini-NERVA (about 1972-73), applications for small nuclear-thermal engines in the 150-400 MW thermal power range for 14 - 50,000 kg lift-off weight stages were investigated. In particular, a 300 MW, 2700 kg engine with 67 kN thrust suitable for interplanetary and interorbital missions was analyzed. The relatively far advanced development of the NERVA engines was stopped around 1974, because at that time a sufficient number of important applications which would justify the costs were still 10 - 15 years in the future and the shuttle project was given precedence for the limited resources of the USA.

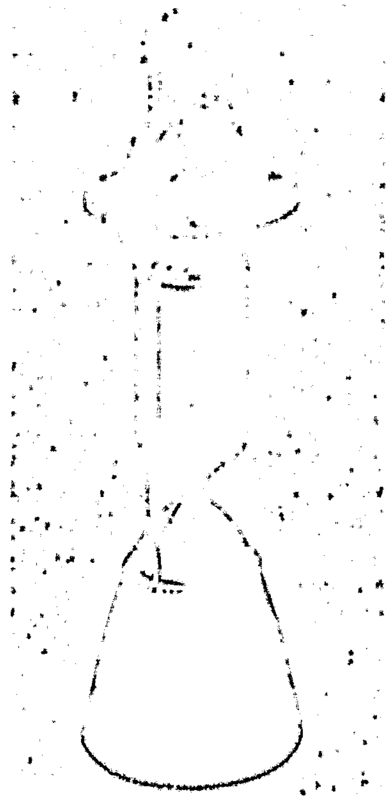

Fig. 5-1: NERVA

LIQUID CORE AND GAS CORE REACTOR CONCEPTS

Research projects have been concerned with the following reactor concepts [100] (estimated possible specific impulse value in parentheses):

1. Fission material is mixed thoroughly in a finely distributed form in the carrier gas "fluidized bed reactor" (100-1100 s).

2. Fission zone in the liquid state, carrier gas pressed through it: liquid core reactor (up to about 1500 s).

3. Fission material gaseous: gas core reactor (up to about 7000 s).

The heating of the carrier gas by radiation was also analyzed. The carrier gas and the gas core are separated by a quartz tube (project "Lightbulb").

These ideas are far from being fully developed and are not likely to be developed in the foreseeable future.

5.2 RADIOISOTOPE-THERMAL ROCKETS

Several propulsion systems of this kind were developed considerably in the USA over the last 20 years.

The following radioisotopes can be used: polonium 210 (138 day halflife), curium 242 (162 days), curium 244 (19.2 yrs.) and plutonium 238 (86.4 yrs.). The engines consist of an isotope capsule surrounded with a radiation shield and heat insulation, from which the fuel (usually hydrogen) draws the heat. Thermal power is in the range of about 1-10 kW.

In the POODLE project [96], a 4 kW polonium 210 hydrogen engine was developed inthe thrust range between about 0.47 to 1.1 N. The specific impulse was 745 sec. at a hydrogen temperature of 2120°K.

Advantages include the relative simplicity and payload improvements compared with chemical rockets. The long flight times for large payloads are disadvantages (similar to electrical propulsion) as well as the required safety measures for possible atmospheric reentry.

The "Nuclear Isotope Mono-Propellant Hydrazine Engine - NIMPHE" is a possible energy hybrid engine. It is an alternative to the electrothermal hydrazine engines described in 4.3. The temperature limits and specific impulse are supposed to be comparable. The advantage of a savings of electrical power (about 250 - 400 W for HIPEHT, or 1 W/mN) is balanced here by the disadvantages of the safety measures for reentry and the required cooling during ascent.

5.3 NUCLEAR-ELECTRICAL ROCKETS AND DUAL-MODE CONCEPT

Nuclear electrical propulsion units in the multi-hundred kilowatt range are still under consideration, even though the development of the SNAP 50-SPUR reactor family (300-1200 kW) has been discontinued in the USA. Technological projects for "out-of-core" as well as "in-core" thermionic systems and turbo electric nuclear reactors are now being investigated.

Studies on a 400 kWe nuclear-electric propulsion system [40, 101] based on an approximately 62 W/kg energy supply system and a 33 W/kg specific thrust radiation power with a 9000 s specific impulse is being considered for investigation of the outer planets, Jupiter, Saturn and others, with sample return. 22-ton stages are compatible with the shuttle and are supposed to carry up to about 12 tons of payload. As a later extension of the 25-50 kW solar-electric system (SEPS), the nuclear-electrical spacecraft will probably play an important role in the far future (after 1990), if the safety and waste removal problems of the reactors can be solved with an acceptable amount of effort.

The dual-mode reactor or nuclear rocket energy center concept has been considered since about 1970 in the USA [104]. It considers alternately heating hydrogen for direct thrust or delivering electrical power using a reactor installation. The same radiation shielding can be used either for short times at a high reactor output (nuclear-thermal rocket) or over an extended time period for electrical propulsion when the reactor is used at a low power level, and can be used for other on-board supply. Many parameter studies and application studies have been carried out for different reactor sizes and power distributions. One of these references [79] resulted in a specific electric power α_E of 60 to 91 W/kg, at 0.1 - 1.0 MWe, for a thermal output of about 212 MW and a 50 kN thrust ("Mini-NERVA" range).

The dual-mode system had the advantage compared with two-stage vehicles that it had high thrust available for the various maneuvers. Therefore, it seems promising for many applications in the remote future.

6. ADDITIONAL PROPULSION CONCEPTS

Among the propulsion concepts discussed here, we would like to discuss four which can probably be developed in the near future. Two others also exist whose technical realization cannot yet be predicted. This includes solar cells and "MPD ramjet" in the form of non-rockets, planetary atmospheric propulsion systems as combination propulsion systems, and solar-thermal, laser and fusion propulsion systems in the form of rockets.

SOLAR CELLS

This concept has been suggested for some time. The radiation pressure of the solar light (0.9×10^{-5} N/m^2 at 1 AE, total reflection) is exploited using a large reflecting surface. In spite of the fact that the radiation pressure pushes radially outwards, solar cell driven spacecraft can also spiral into the sun, if the thrust is directed against the direction of flight by suitably positioning the mirror. After approaching the sun, the cell is transferred to the thrust mode in the flight direction. Therefore, by flying around the sun, the spacecraft gains considerable thrust. The advantage of the solar cell (compared to solar electric systems) is that within the temperature limits of cells (about 330°C) the payload can completely exploit the high radiation density near the sun (16 solar constants at 0.25 AE) without the need for additional large weight expenditure.

The solar cell principle was already used in the U. S. Mariner probes (for example, Mariner IV Mars) for passive attitude control using moveable solar rudders.

In the past four years, NASA considered the solar cell for planetary research programs [87]. Comparison studies (competition) between solar cells and solar electric propulsion were carried out for several complex comet missions (Halley 1985-86 rendezvous)(86, 105).

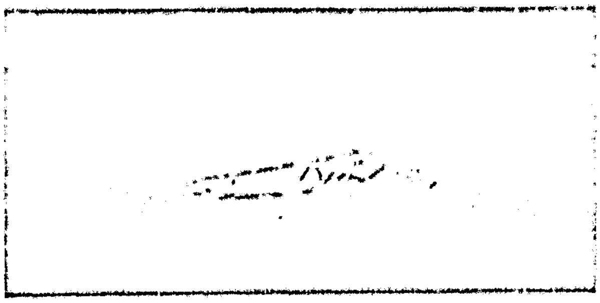

Fig. 6-1: Solar cell with square sailing surface [85]

Enormous solar cells with areas of 6-7 x 10^5 m^2 with extremely light structures (cell weight about 8.1 - 9 g/m^2) were required to drive the 600 - 625 kg payload module at 10^{-3} m/s^2 at the beginning. In about 3-1/2 years the spacecraft would be brought into the retrograde Halley trajectory, after it had been accelerated with the shuttle and the IUS into an Earth escape trajectory. At the beginning, square [85] (see Fig. 6-1) and later on windmill-like rotating solar cells were calculated. Twelve blades 7.5 km long, 8 m wide, covered with approximately 2 μm thick metallic capton or parylene foils made up the cell area (approximately 3 g/m^2). This resulted in a blade mass of 200 kg and a total cell mass of about 5.6 tons [86].

Because of the advanced state of development of solar electric ion propulsion and its great flexibility for changing the flight trajectory, preference was given to the electrical systems (SEPS) and the development of solar cells has been deferred.

CHEMICAL PROPULSION WITH EXPLOITATION OF PLANETARY ATMOSPHERES

Just like for air-breathing chemical propulsion systems, the propulsion systems of planetary probes can also exploit planetary atmospheres. The planetary atmosphere can either be used as support mass if it is composed of neutral gas (example: Mars), or it can be used as an oxygen carrier (example: Venus, for example with Be or Li), or it can be used as fuel (example: Jupiter). In the case of a neutral atmosphere, it is possible to consider propeller propulsion driven by a rocket gas generator. Such a propulsion system with hydrazine gas generator is presently the topic of a demonstration program for an unmanned Mars aircraft (RPV) at the request of NASA [91].

"MPD RAMJET" PROPULSION SYSTEMS

In principle, by interaction of a magnetic field with a gas discharge, it is possible to produce thrust even in a very rarefied partially ionized upper atmosphere. This effect was measured in the early tests of MPD plasma engines in insufficient vacuums. This thrust could be used for compensating for drag in satellites. Various configurations for capturing and accelerating the rarefied atmosphere (plasma) are possible. Studies are in progress in the USA under the title "Space Electric Ramjet" [88].

SOLAR-THERMAL ROCKET

In this propulsion principle, the heat radiation of the sun is concentrated at the focus of a parabolic mirror and is used to heat up the support mass. For a relatively high thrust(> 0.07 N/kg, i.e., substantially higher than for solar electric rockets), exit speeds can be reached which are comparable with those of nuclear-thermal units without the occurrence of safety or cooling problems.

Extremely light mirrors of the required quality were produced in the 60s using electrolytic methods and they were fabricated from one piece. Because of transportation reasons, their size was limited to about 4 m diameter. Foldable mirrors do not achieve the temperatures of about 2000°K [102] required for $I_s > 700$ s. Another important difficulty is the required alignment of the mirror axis towards the sun.

For this reason, solar-thermal rockets do not seem to be competitive. The development of large mirrors, also for solarthermionic energy supply installations, has been deferred.

MASS DRIVER AND PELLET LAUNCHER

Another suggestion for future space propulsion is the "Mass Driver" [84]. Burned-out rocket stages for asteroid masses are pulverized and accelerated electromagnetically. The total length of the acceleration tube is more than 4 km and the total system mass is 170,000 kg. The exit speed of 10 km/s has been calculated corresponding to a specific impulse of over 1000 s.

Another suggestion is the so-called "Rotary Pellet Launcher" for position control of large satellites. Small masses of meteorite or moon rock are mechanically accelerated in a rotating system and expelled so that a recoil is produced [92]. The exit speeds calculated with such a complex device are quite limited.

Both propulsion suggestions would have to be compared with many older suggestions (about 1960), with which pulverized parts or rockes were accelerated as driving agents in plasma engines.

LASER PROPULSION

The external supply of energy using laser radiation is an often-discussed possibility for increasing the specific performance of jet engines. For missions with a high propulsion requirement, one could bring about substantial mass savings doing this (see Fig. 3-3, point LAS). This would mean that the takeoff weight of an SSTO launch vehicle with a payload of 30 tons (into LEO) could be reduced from several thousand tons using purely chemical propulsion to the size of an aircraft, that is about 96 tons [3], if the specific impulse were increased to 1248 s using laser energy. The required available useful power of the laser, however, would be greater than 6.9 GW. In fact, specific impulses of 1325 s can be achieved with heated hydrogen or even 1940 s using 50% carbon additives [82]. But the performances in the GW range are several orders of magnitude greater than one could expect, considering the fast development of military lasers. Hybrid systems with chemical energy contributions require GW lasers [83] as well. In addition, laser electric systems instead of solar electric systems have been discussed [103].

FUSION PROPULSION

Thermonuclear fusion has been of interest for a long time with a potentially higher specific energy and possibly even a higher specific thrust. One concept for bringing this about is similar to the terrestrial laser (electron or ion) radiation fusion, is the suggestion of microhydrogen bomb propulsion [95]. A deuterium-tritium target will undergo a fusion reaction behind the spacecraft. Additional fusion rocket concercs are based on the magnetic inclusion of a thermonuclear fusion plasma, but these belong to the distant future. In addition, the weight of the installation seems to be a major problem.

7. CONCLUSIONS

LAUNCH VEHICLES

The technological state of present chemical rocket propulsion for launch vehicle rockets is represented by the not yet completely developed main engine of the space shuttle. In the near future, we can expect further improvements by increasing the combustion chamber pressure, by using nozzles with variable area ratios and by employing the mixed mode propulsion concept. These improvements will only result in a few percentage gains in the specific impulse, but substantially increase the payload. Improvements beyond this can be foreseen by using air-breathing propulsion in combination with rocket propulsion.

On the other hand, no improvements by the use of new high-energy fuels for launch vehicles can be foreseen in the foreseeable future. Because of cost effectiveness and environmental protection, such fuels must be excluded (with the burning of fluor, hydrogen and metal). For this reason, we believe that the large solid booster propulsion systems will be replaced by liquid propulsion systems.

The mentioned improvements will make possible the development of even more economical and reuseable space transport vehicles with only one or two stages.

For user groups with small yearly transportation requirements, non-reuseable units will still be the most economical solution.

In future developments, one will distinguish between shuttle-like passenger transportation vehicles and large freight transportation vehicles (heavy lift launch vehicles), which will result in substantial cost reductions because of improvements in the structure and materials and because of their large size. The requirements for these vehicles will stimulate new research on engines.

KICK AND TRANSFER STAGES

The solid fuel propulsion method is mostly used in the present day kick stages and transfer stages, including the IUS under development for the space shuttle. Reuseable space tug systems, maneuverable satellites, space stations and interplanetary spaceccraft will require increased propulsion. These requirements lead to the development of two fuel propulsion systems, first with space storable fuel combinations. Later on this will lead to semi-cryogenic and even cryogenic systems. The use of high-energy fuel combinations (with combustion of fluor, hydrogen and metal) can be expected in the far future. Further improvements can be expected by increasing the combustion chamber pressure, using mixed mode and by using the smallest number of stages, and by improving the structural and material characteristics of the overall system.

TRAJECTORY AND ATTITUDE CONTROL SYSTEMS

For missions with low impulse requirement (about $< 10^3$ Ns), the trusted cold gas system will again be considered. Small pulsed plasma propulsion units have been used for many years as attitude control devices. For an average propulsion requirement (about $10^3 - 10^5$ Ns), most catalytic hydrazine systems and NH_3 resistojets will probably be replaced in the near future by chemical-electrothermal propulsion systems. Somewhat above this total impulse requirement of 10^5 Ns, electrical propulsion systems will be used after demonstrating flight worthiness, if the low specific thrust is sufficient for position control, just like for the position control of geostationary satellites. For missions near the Earth which require high maneuverability, N_2H_4 resistojets or isotope engines will be used.

ELECTRICAL PROPULSION

The use of electrical ion engines is certainly to be expected for future geostationary satellites (first for NS angle correction) as soon as these engines are fully developed. The American 8 cm engine will be flight tested in 1981 and the comparable European RIT-10 engine will be flight tested in 1983.

The 30 cm ion engine system developed in the USA for primary propulsion is at present undergoing lifetime tests. A comparable ion main engine RIT-35 is being developed in Germany. The American system is planned for the Halley/Tempel 2 comet mission with a launch in 1985.

A decision was recently made at NASA to use solar electric propulsion systems for interplanetary research. This represents a milestone for electrical engines. This means new dimensions are being explored in interplanetary space flight.

Continuous and quasistationary plasma propulsion seems to be suitable for power requirements beyond 100 KW up to multi-MW range, because they provide substantially higher power densities than do ion engines and they have simpler energy conversion units. With specific impulses in the range between 1000-5000 s, they could be of interest for interorbital transport and the attitude control of large satellites, for example energy satellites. Because of the very limited development expenditure up to the present time, no suitable engines of this type exist for applications, nor are reliable performance data available. This represents a real technological gap.

NUCLEAR AND OTHER PROPULSION SYSTEMS

Among the other propulsion systems, several are technically realizable -- others cannot be predicted at all.

Nuclear-electrical propulsion systems at the present time still offer the best possibility for intensive and manned exploration of the planets, possibly coupled with a nuclear-thermal pre-stage or used as a "dual-mode" system. The nuclear-thermal systems alone seem to be predestined for the heavy transport missions near the Earth. Except for the radioisojet, neither one can be expected before the 1990s, and among the various problems which must be solved, there is the safety problem.

The solar cell could be developed later on in addition to solar electric systems for planetary exploration. The development of laser propulsion will require breakthroughs in the areas of lasers and engines. These propulsion systems could offer substantial advantages for ascent through the atmosphere. At this time they would have to compete with advanced air-breathing engines.

Nuclear fusion rockets, as far as can be predicted today, offer the most likely possibility of a propulsion system with simultaneously high thrust and high specific impulse. This is a ray of hope for researchers and spacemen of the future generations.

Summarizing, we can say that chemical propulsion has reached a development plateau for Earth LEO transport, which only justifies relatively small but important improvements. New developments for it will only be started when the traffic density requires new larger transport systems, and this will be motivated by commercially important application satellites.

For spacecraft propulsion systems in Earth orbit and in planetary space, there already exist important requirements of the military and civilian type for the immediate future, which will give the impetus for new developments of chemical and other systems in the coming decades.

For the complete economic mastery and exploitation of space and space flight, in the distant future completely new propulsion systems will have to be built, which today can hardly be imagined.

This means that many challenging tasks exist for research and development of propulsion systems.

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APPENDIX 1

The propulsion capacity Δv_{ANT} of a space vehicle (rocket) is given by

$$\Delta v_{ANT} = \int_{\tau} \frac{F}{m_R}\, dt = \text{fictitious velocity increase of a rocket in space without forces (mass } m_R(t))$$

Even though space vehicles fly in combined gravitational fields of the sun and planets and the influence of other forces (air resistance, solar pressure, etc.), the propulsion capacity of a system and the propulsion requirement for a mission is expressed in the form of such (fictitious) Δv values.

For single-stage rocket with a constant exit speed c_e, the propulsion capacity is

$$\Delta v_{ANT} = c_e \ln(m_o/m_B)$$

and for multistage (n) rockets, we have

$$\Delta v_{ANT} = \sum_{j=1}^{n} c_{e_j} \ln(m_{o_j}/m_{B_j})$$

The payload $m_L = m_B - m_S - m_W$, so that the payload fraction for single-stage vehicles is

$$\mu_L = \frac{m_L}{m_o} = \exp(-\Delta v_{ANT}/c_e) - (\mu_S + \mu_W)$$

This is somewhat more complicated for multistage vehicles. However, numerical approximate formulas can be derived from an envelope of optimized multistage vehicles (as done in Fig. 2-1).

The propulsion requirement for a flight mission Δv_{CHAR} is defined as the sum of the absolute values of all necessary equivalent "velocity changes" Δv (in the sense given above), which must be provided by the propulsion system, without consideration of the direction or the sign

$$\Delta v_{CHAR} = \sum_i |\Delta v_{i_{ANT}}|$$

Δv_{CHAR} is calculated from the ideal Δv req. for the flight task and the given launch site and the launch direction, including the directional changes, braking and landing maneuvers, plus corresponding Δv values for gravitational losses, especially in the Earth's field, air drag losses during ascent, fuel reserves, etc. (see, for example [97]). The ideal propulsion requirement can be reduced by atmospheric braking upon return and possible swingby maneuvers (see 2-3); however, it increases substantially if higher flight speeds are used for saving time instead of the minimum required.

In this summary, the propulsion requirement and propulsion capacity are simply set equal and called Δv.

For rockets where the energy system is separate from the fuel, the mass ratio

$$\frac{m_o}{m_B} = \frac{\text{initial mass}}{\text{burn-out mass}}$$

can be represented with the parameter α (= specific performance of propulsion system) and the payload ratio including the net propulsion mass and fuel mass (m_W and m_T) in the following form (see Stuhlinger [36])

$$\frac{m_o}{m_B} = \frac{1+c_e^{2}/2\alpha\tau}{\mu_{LS}+c_e^{2}/2\alpha\tau} = \frac{1+m_W/m_T}{\mu_{LS}+m_W/m_T}$$

The quantity α also considers the transfer efficiency and motor efficiency. μ_{LS} is the sum of the structure mass fractions and the payload mass fractions $\mu_L + \mu_S$. According to Stuhlinger, for an electrical or nuclear single-stage vehicle μ_{LS} becomes a symbol function of $\Delta v/c_e$ and $(m_W/m_T) = c_e^2/2\alpha\tau)$

$$\mu_{LS} = (1 - \frac{m_W}{m_T})\exp(-\frac{\Delta v}{c_e}) - \frac{m_W}{m_T}$$

For a given flight mission (Δv), c_e and therefore m_W/m_T can be optimized around the optimum possible ratio of μ_{LS} and the propulsion duration (see Chapter 4.1), if simplifying assumptions are made.