5 Earth Orbit on-Orbit Operations in Near-Earth

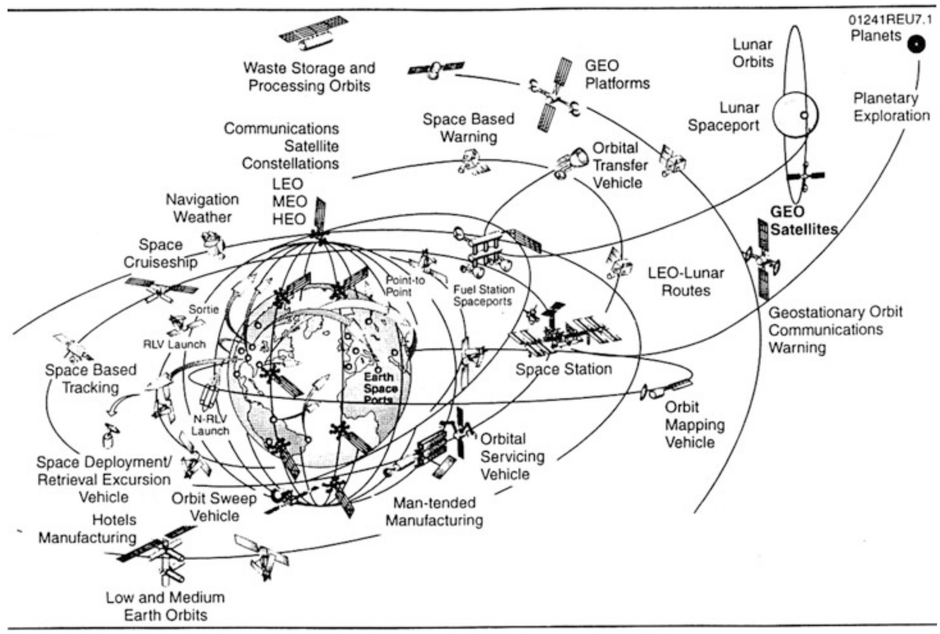
Although not in the frontline technical or popular press, a critical element in reaching space beyond Earth is establishing the space infrastructure around the planet Earth. The concept of this infrastructure as a train marshaling and switching yard is appropriate, a difference being the difficulty of access. The rail control center serves as a center of operations for switching, long-haul train assembly, transfer of goods, refueling, and repair. Likewise, the orbital stations serve as centers for switching payloads between carrier and the required orbit, long-haul space exploration vehicle assembly, transfer of goods to human habitats and manufacturing facilities, return, refueling, and repair coordination. This is no trivial activity, and it will take a commitment as dedicated as the Apollo program to achieve. In this day and age, a correspondent return on investment must be shown. In a step-by-step discussion, we will document the resources necessary to supply what is needed by this space infrastructure as a function of the vehicle-integrated propulsion systems.

Chapter 4 presents those propulsion systems with which we can effectively build reduced oxidizer-to-fuel ratio launchers that are lighter and smaller than conventional expendable rockets. In fact, the remotely powered, directed electromagnetic energy system of the late Professor Leik Myrabo requires even less carried onboard propellants, a huge advantage in seemingly resource-absent space. Note that as long as the principal launchers are expendable launchers for military and commercial needs, the available payloads will be *only* those suitable for infrequent expendable rocket launches. As discussed in the context of Chap. 2, the payloads will then remain consistent with Conestoga wagons until there is an operational railroad equivalent. Until a sustained-use launch system is operational, the payloads that warrant a high launch rate system will remain the subject of design studies only! Until such sustained-use launch system is operational, the flight rate will remain insufficient to build the global space infrastructure needed to support space operations. If the Space Shuttle main propellant tank would have been slightly modified to permit its use as a space structure before the retirement of the STS in 2011 (Hale 2012), like the empty S-IVB third stage of the Saturn V and second stage on the Saturn IB (Bielstein 1980), this could have been the beginning of a space infrastructure as a first step (Taylor 1998, 2000). However, the Space Shuttle main tank was instead intentionally crashed into the ocean, wasting such valuable asset.

Assuming the capability existed for sustained space launches to establish an operational near-Earth orbit space infrastructure, there are serious performance and propellant refueling challenges that need to be immediately addressed. Because of the activity required by the elements of the near-Earth orbit infrastructure, the quantity of propellant required in space and, more importantly, the quantity of launcher propellant required to lift from the Earth surface that very propellant into low Earth orbit (LEO) are truly prodigious, unless a non-chemical rocket is used. For a true space transportation system to exist, a transportation system network has to be built just as it was for the US transcontinental railroad. The late Dr. William Gaubatz, formerly of McDonnell Douglas Astronautics and former director of the Delta Clipper DC-X program (Butrica 2003; Stine 1996; Hannigan 1994), attempted to anticipate what the future might hold if a space transportation system actually would exist, as shown in Fig. 5.1. Dr. Gaubatz shows the elements necessary to build the infrastructure, but unfortunately does not address the assets required to establish and sustain that infrastructure.

Figure 5.1 presents a functional orbital infrastructure, including space habitats, free-flying facilities, and power stations at several levels of development, using prior work of Dr. Gaubatz. Table 5.1 lists the orbital vehicles and platforms and their diverse functions facilitating a true space infrastructure. Future global space is a crowded and busy place although a snapshot would show only single elements (points) in space, not trajectories. The key enabling space structures are the fuel station spaceports and orbital servicing vehicles. Without these vehicles, movement between orbital

P.A. Czysz et al., *Future Spacecraft Propulsion Systems and Integration*,
Springer Praxis Books, DOI 10.1007/978-3-662-54744-1_5



Growth of the Spaceways Routes

Fig. 5.1 Growth in spaceway routes. Future space infrastructure envisioned by Dr. William Gaubatz, if enabled by a space transportation system and in-space operations system to support the infrastructure

planes and altitudes is limited to specific satellites, such as GSO communication satellites with integral geotransfer propulsion. With servicing centers equipped with construction module storage, they can supply components for orbital, lunar, and deep-space vehicle assembly in space. The operations center/space station provides a system to launch and control missions to the Moon, the planets and deep space. Like the USSR plan introduced in Chap. 2, there are lunar spaceports and lunar orbiting satellites. There are space deployment and retrieval vehicles as well as a waste storage and processing facility in high orbit. Then, this outlook provides a comprehensive projection of future space if a suitable (a) scheduled, frequent, and sustained transportation and (b) heavy-lift capability are available as the key prerequisites. In short, these two "critical mass" enabling elements are needed to plan for the future, not the current status quo.

What is not shown in the visualization by Dr. Gaubatz, see Fig. 5.1, is a solar power station that beams power to the Earth's surface, space assets, or a power station warehouse that provides hardware for the power satellites in geostationary Earth orbit. It remains to be seen whether a solar power station has the energy conversion efficiency to provide affordable energy to Earth or space assets comparable to nuclear power stations. In this context, a source of excellent information on solar power stations is from reports by H.H. Koelle, formerly at the University of Berlin (Koelle 1961; 1995). In fact, the singular reliance on photovoltaic power generation may doom all power stations until a more efficient and more durable conversion system can be identified. As proven by the NASA LDEF (Long Duration Exposure Facility) materials evaluation satellite, space is a very hostile environment and we have yet to identify slowly deteriorating or non-deteriorating materials and construction concepts, including those for solar panels. Nikolai Anfimov, of the Russian TsNIIMash (Central Research Institute of Machine Building), in a private communication with author P.A. Czysz, had stated that the hub of the Russian MIR orbital station, exposed to the space environment for 15 years in orbit, was so riddled with cosmic-Galactic radiation particles (e.g., *Fe* ions) that it was beginning to leak, even though there were no visible holes. Clearly, the complexity and extensive nature of the space infrastructure implies that a significant commitment of human and monetary resources is necessary if we are to go beyond the two currently operating solitary orbital stations (ISS and Tiangong-1) with limited capability.

In fact, infrastructure demands access to space and maneuvering in space. Spacecraft must be lifted to orbit more economically than possible now and, once there, must be able to change their orbit to reach propellant depots, crewed maintenance and space stations, space hotels, TLC (Telecommunications) and scientific satellites, and rescue vehicles ("lifeboats"). All these in-space missions require more economical in-space propulsion for space tugs and orbital maneuvering vehicles (OMVs) not only to raise or lower orbit and to rendezvous, but also to change orbital plane. Operation must be more economical both in terms of energy and mass. Accordingly, this is the focus of this chapter.

Table 5.1 Space infrastructure vehicles and missions, from Fig. 5.1

	Orbital system	Function	Orbit
1	Sustained-use launcher	High frequency, modest payloads	LEO/MEO
2	Expendable launcher	Low frequency, heavy payloads	LEO
3	Point-to-point transfer	Points on Earth or orbit	
4	Operations center/space station	Operations coordination/research	LEO/MEO
5	Orbital servicing vehicle	Maintains in-orbit vehicles	All
6	Fuel station spaceport	Refuels orbital vehicles	LEO
7	Space-based manufacturing	Human based low *g* manufacturing	LEO
8	Man-tended manufacturing	Robot based micro*g* manufacturing	LEO/GEO
9	Orbital sweep vehicle	Orbital cleanup vehicle	All
10	Waste storage and processing vehicles	Processes and disposes human and manufacturing wastes	HEO
11	Navigation/weather	Supports travel network	LEO/MEO
12	Orbital mapping vehicle	Measures resources and geography	LEO/MEO
13	Space-based warning	Military and asteroid warning	HEO/GEO
14	Space-based hotel	Space tourism facilities	LEO/MEO
15	Space Cruiser vehicle	Human transport and rescue	LEO
16	Communication satellite constellations	Supports telecommunication systems	All
17	Orbital transfer vehicle	Orbital altitude/plane change	All
18	LEO-lunar vehicle	Transport to Moon and return	LEO
19	Space deployment retrieval vehicle	Recovers spent vehicles	All
		Replaces spent vehicles	
20	Space excursion vehicle	Placement of new systems	LEO
21	GEO platforms/satellites	Micro*g* and magnetic field space	GEO
22	GEO communications and warning vehicles	Fixed equatorial position	GEO
23	Lunar spaceport system	Lunar transportation/research hub	Lunar
24	Lunar orbital vehicles	Support lunar activities	Lunar
25	Planetary exploration vehicles	Near- and deep-space vehicles	LEO/Lunar

5.1 Energy Requirements

The concept of the train yard as a center of operations for switching, long-haul vehicle assembly, transfer of goods, refueling, and repair is not unrealistic for the first-generation space infrastructure. As we shall see, the energy requirements are greater for mobility in the vicinity of Earth than to reach LEO. There is a clear need for a nuclear-powered tug for orbital transfer from LEO to geostationary orbit (GSO) and return, see Chap. 7. There is also a need for collecting, repair, or disposal of non-functional satellites in LEO and GSO; for the refueling of sustained-use satellites; for orbital busses and tugs; and, generally speaking, for sustained in-orbit operations and maintenance. As we shall see, this implies a first step that must be taken as far as vehicle-integrated propulsion to anticipate the future.

5.1.1 Getting to Low Earth Orbit: Energy and Propellant Requirements

At nonrelativistic speed, all of the classical orbital mechanics from near-Earth to the edge of our solar system and beyond are based on Newton's fundamental law of gravitational attraction. The assumption is that the gravitational force, $\vec{F}_g$, acts throughout the universe in the same way. Newton's law of universal gravitational force between the mass of two bodies, m_1 and m_2, with the distance, r, between the center of mass of the two bodies is given by:

$$F_g = G \cdot \frac{m_1 \cdot m_2}{r^2} \tag{5.1}$$

The universal gravitational constant, G, is

$$G = 6.67408 \cdot 10^{-11} \frac{\mathrm{m}^3}{\mathrm{kg\,s}^2} \tag{5.2}$$

Gravity is probably one of the most mysterious forces in the universe. In fact, while our everyday experience of gravity is commonplace, our understanding is very limited. The law has been well tested on Earth and in the vicinity of the Earth. However, when astronomers attempt to use Newton's fundamental law of gravitational attraction to predict the motion of stars orbiting the center of the Galaxy, they sometimes must grapple with strange results. The most distant man-made objects are *Pioneer 10* launched in 1972 and *Pioneer 11* launched in 1973. *Pioneer 10* is now more than 8 billion miles from Earth. On January 23, 2003, the tracking stations picked up the last feeble transmission from the probe's radioactive isotope (plutonium)-powered transmitter (Folger 2003; Wolverton 2004). As *Pioneer 10*'s feeble signal faded from detection, the spacecraft seemed to be defying Newton's law of gravity because it was slowing down as if the gravitational attraction from the Sun was growing stronger the farther away it traveled. *Pioneer 11* also slowed down in a similar manner. The *Ulysses* space probe, which has been orbiting the Sun for 13 years, has also behaved in a manner characteristic of an unknown force slowing it down.

In the case of the *Pioneer 10* and *Pioneer 11* probes, the most distant man-made objects, this perceived irregularity led for some time to postulate that Newton's law changed with increasing distance. *Pioneer 10* launched in 1972 and *Pioneer 11* launched in 1973. *Pioneer 10* is more than 10 billion miles from Earth. On January 23, 2003, the tracking stations picked up the last feeble transmission from the probe's radioactive isotope (plutonium)-powered transmitter (Folger 2003). As *Pioneer 10*'s feeble signal faded from detection, the spacecraft seemed to be defying Newton's law of gravity because it was slowing down as if the gravitational attraction from the Sun was growing stronger the farther away it traveled. *Pioneer 11* also slowed down in a similar manner. The *Ulysses* spacecraft orbiting the Sun also behaved as if an unknown force slowed it down. This so-called *Pioneer Anomaly* was eventually explained as reported in Chap. 9 (Turyshev et al. 2004), but there is some scant evidence that perhaps gravity does not act in the same way on a galactic scale. Our Galaxy makes one rotation in about the time from when dinosaurs began to inhabit the Earth to now. Perhaps on that time and distance scale, gravity may act differently. Until more is understood, we will continue with the traditional assumption of gravity acting the same throughout the universe, but also need to acknowledge that the farther we travel and the longer we are in space, we may be departing from the expected.

The law of gravity rules the attraction between two masses. When in motion, then the law that governs the two-body problem (that is, a large central body and a moving smaller body) yields Kepler's three laws of motion. Although gravitational forces can be formulated for N number of bodies, the only analytic (closed-form) solutions found are for $N = 2$. Numerical solutions are possible, but these involve the use of the largest computers; they are used only when the simple two-body problem is suspect (such as predicting a Mercury orbiter trajectory) or high navigational accuracy is required (Logsdon 1997). The Keplerian *circular* orbit relationships between two bodies are given below (Koelle 1961):

$$V_{\text{circular}} = \sqrt{\frac{M \cdot G}{r}} \quad [\text{km/s}] \tag{5.3a}$$

$$V_{\text{circular}} = \sqrt{\frac{\mu}{r}} \quad [\text{km/s}] \tag{5.3b}$$

$$V_{\text{circular}} = \sqrt{\frac{\mu}{R_0 + h}} \quad [\text{km/s}] \tag{5.3c}$$

$$\text{Period} = 2 \cdot \pi \sqrt{\frac{r^3}{\mu}} \quad [\text{s}] \tag{5.4a}$$

$$\text{Period} = 2 \cdot \pi \sqrt{\frac{(R_0 + h)^3}{\mu}} \quad [\text{s}] \tag{5.4b}$$

where μ = gravitational constant = $M{\cdot}G$, M = mass of the central body, r = radius from the spacecraft center of mass to the center of mass of the central body, R_0 = planet radius, and h = altitude above surface.

The gravitational parameters and the orbital speeds for a 200-km orbit and escape are given in Table 5.2 for selected bodies.

From Eqs. (5.1), (5.3a, b, c) and (5.4a, b), the orbital velocity decreases and the orbital period increases as the spacecraft altitude is increased, see Figs. 3.6, 3.7 and 3.8. The two-body equations assume non-rotating masses. If the central body is rotating, then its rotation can add a velocity vector increment to the launcher vehicle, dependent on the latitude of the launch site and the launch azimuth. Figure 5.2 shows the required velocity increment from the Earth's surface to the orbital altitude (in nautical miles).

Both the non-rotating Earth and rotating Earth (launch site at the Equator) velocity increments required are shown in Fig. 5.2. These are not the velocities in orbit, but the velocity increment (energy increment) that determines the mass ratio to reach simultaneously the given orbital altitude and required orbital speed. The speed of the Earth's surface at the Equator is 463.6 m/s (1521 ft/s). That reduces the launch speed increment (ΔV) to 7331.05 m/s (24,052 ft/s) if the launcher is launched due east (90° latitude from true north) at the Equator. If the launcher is launched due west, then the launcher must cancel out the easterly motion, so the

Table 5.2 Gravitational characteristics of nearby planets and Earth's Moon

	Venus	Earth	Moon	Mars	Jupiter
μ (km^3/s^2)	324,858.8	398,600.4	4902.8	42,828.3	126,711,995.4
R_0 (km)	6061.8	6378.14	1737.4	3397.0	71,492
V_{200} (km/s)	7.203	7.784	1.680	3.551	42.10
V_{esc} (km/s)	10.187	11.008	2.376	5.022	59.538

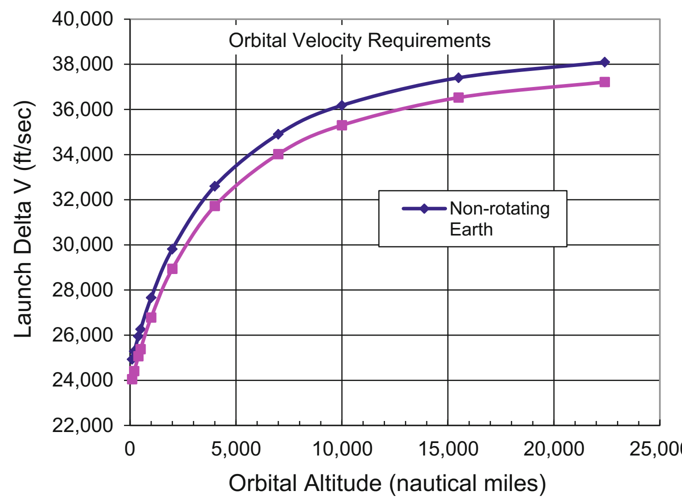


Fig. 5.2 Launch velocity increment to reach Earth orbit

launch speed increment (ΔV) is 8258.25 m/s (27,094 ft/s). For a true east launch, the launch velocity increment as a function of the launch site latitude *La* is:

$$V_0 = V_{\text{circular}} - 1521 \cdot \sin(La) \quad [\text{ft/s}] \tag{5.5}$$

For a due east launch, the inclination of the orbit is equal to the latitude of the launch site. Figure 5.3 shows the velocity increment for the launch ΔV as a function of the launch site azimuth for a due east launch with a number of launch sites indicated. In reality, the launch azimuth will not always be due east. The launch azimuth for a non-rotating Earth at a given orbital inclination and launch site latitude is:

$$\sin A_z = \frac{\cos i}{\cos(La)} \tag{5.6}$$

with A_z = launch azimuth from true north, and i = orbital inclination.

Equation (5.6) defines the minimum inclination for an orbit as the latitude of the launch site and a true east or west launch (90° or 270°). For the rotating Earth case, a correction to the launch azimuth and velocity must be made by the vector addition of the eastward velocity of the Earth and the launch velocity vector. But Eq. (5.6) will give the minimum azimuth and a good first-order value. For a Sun-synchronous orbit (SSO at 98°) from a launch site at 45° latitude, this value is −11.4° or an azimuth of 348.6°. For the International Space Station (ISS) orbit (55°) from Kennedy (28.5°), the azimuth angle is 40.7° or just north of northwest. Consequently, when the Space Shuttle launched from Kennedy, the spacecraft had to roll to position the wing plane perpendicular to 40.7° and then proceed along its launch trajectory.

Given the incremental velocity required to achieve a circular orbit, the next step in discussing the infrastructure logistics is to determine the quantity of launch propellant required to place a given quantity of propellant into LEO for interorbit maneuvering.

5.2 Launcher Propulsion System Characteristics

Section 3.1 provides the governing equations and methodology for determining launcher size to achieve a given velocity increment with a given payload mass. The sizing process is the same for determining the quantity of launch propellant required to place a given quantity of propellant (payload) into LEO. The difference is that for a fixed-volume payload bay, each propellant combination has a different bulk density and therefore a different tank volume occupied for a fixed propellant mass. Overall, the role of the propellant delivery vehicle is analogous to that of an Air Force tanker aircraft. Its role is to deliver fuel to in-flight operational

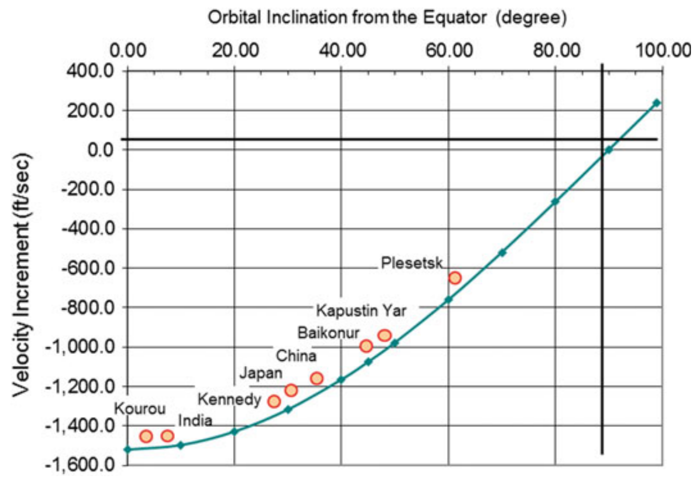


Fig. 5.3 Velocity increment to 200-nm orbit for orbital inclination. Some launch centers are indicated

vehicles on demand and on a sustained operational basis. In this case, the role of the LEO tanker is to routinely deliver propellant to an orbital refueling station in LEO. Being a dedicated tanker, the cargo container is a propellant tank, with provisions for transferring propellant in orbit. In microgravity, special design considerations are necessary (e.g., that the propellant is adjacent to the transfer pumps), but much of this has been accomplished for some time in space and is a known and established design practice.

In all cases, the LEO tanker is an automatic vehicle that has sustained, frequent use and routine exit and entry of the atmosphere attributes. In short, it is not an expendable or a reusable expendable vehicle. As a consequence, the best configuration choice for the LEO tanker is the hypersonic glider or airbreathing launcher, as shown in Figs. 2.18 and 2.19, and Figs. 4.39–4.41. With the following, four different launcher propulsion systems are evaluated for the tanker to LEO mission:

(1) Hydrogen/oxygen rocket, e.g., based on the Pratt & Whitney XLR-129 (Mulready 2001).
(2) Hydrogen/oxygen LACE rocket, based on the Pratt & Whitney XLR-129.
(3) Rocket ejector ram–scramjet airbreathing to Mach 10, transitioning to a hydrogen/oxygen rocket, based on the Pratt & Whitney XLR-129.
(4) Rocket ejector ram–scramjet airbreathing to Mach 12, transitioning to a hydrogen/oxygen rocket, based on the Pratt & Whitney XLR-129. (Note: All rocket technology was sold in 2013 from Pratt & Whitney to Aerojet-General, of Sacramento, CA.)

The design payload selected here is 19 t (41,895 lb) of propellant with a bulk density of 999.4 kg/m^3 (62.4 lb/ft^3). A launcher for the design payload was sized for each of the four propulsion systems. For different propellant densities, the size and weight of the launcher is different. These corrections are discussed later in this chapter and are given in Fig. 5.4.

5.2.1 Propellant Ratio to Deliver Propellant to LEO

The propellant ratio is defined here as the propellant mass burned by the launcher to achieve LEO, divided by the propellant load carried to LEO. Mass and density of the propellant affect the size of the launcher, and this sensitivity has been evaluated. The launchers are sized using the methodology described in Chap. 3. The vehicle assumptions are the same as outlined in Chap. 4, except that a permanent propellant tank replaced the accessible payload bay. For the design payload and payload density, the sizing results are given in Table 5.3.

The propulsion system selection determines the key parameter for an orbital tanker, which is the propellant burnt to lift the orbital maneuver propellant, divided by the propellant delivered. The LACE rocket is an adaptation of an existing operational rocket engine and requires good engineering design and testing, but it is not a technological challenge. The LACE rocket offers a greater than 50% reduction in the propellant required to deliver the design payload of 19 t of propellant to LEO, as shown in Table 5.3 and Fig. 5.5.

Because of the LACE rocket's greater thrust over drag ratio *T/D*, the propellant ratio is slightly better than a rocket ejector ramjet utilizing atmospheric air up to Mach 6. A piloted vehicle is at a disadvantage for an orbital tanker in that the provisions for the pilot increase the propellant required for delivering the orbital propellant to LEO. Clearly, transitioning to an airbreather vehicle configuration offers the potential to reduce the propellant required to deliver the orbital maneuver propellant by 38 and 52%, respectively. Proceeding beyond an airbreathing Mach number of 12 results in an increase in the propellant required to deliver the orbital maneuver propellant, see Fig. 5.5.

The important conclusion from this analysis is that as a first step, basing the propulsion system on an existing rocket motor

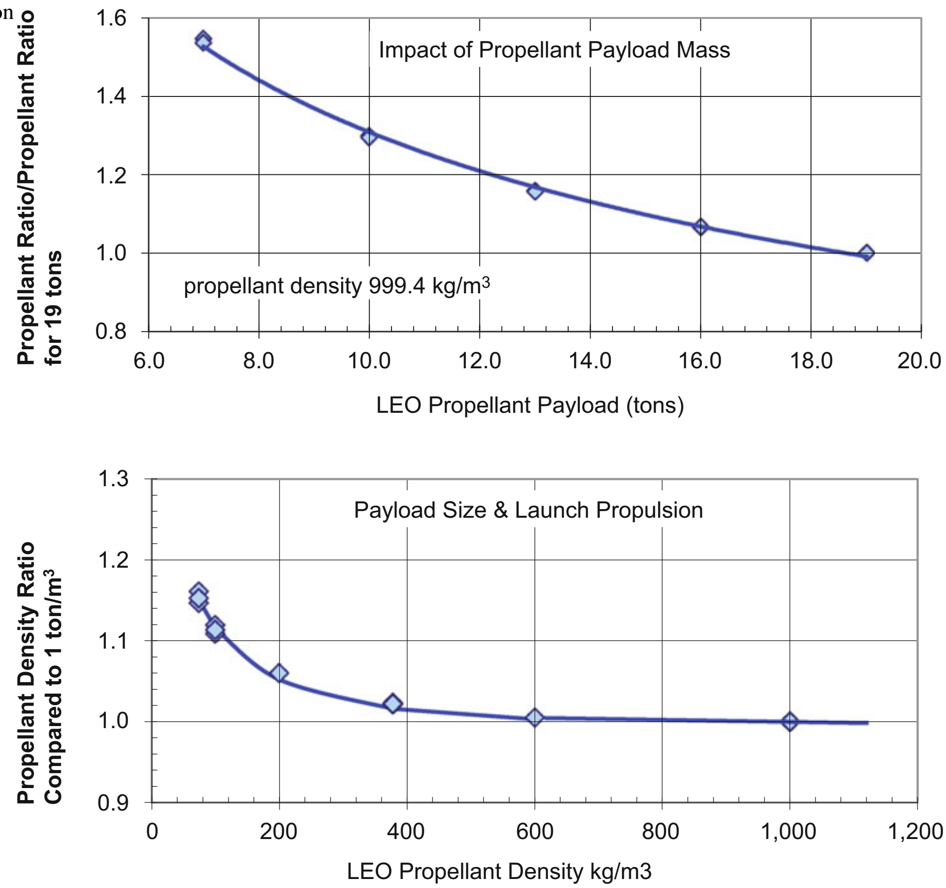


Fig. 5.4 Propellant required as a function of payload mass and density

Table 5.3 Launchers sized to deliver 19 t of propellant to LEO

	H_2/O_2 rocket FDL-7C/D	LACE rocket FDL-7C/D	RBCC Mach 10 airbreather	RBCC Mach 12 airbreather
Planform area (m^2)	600	370	301	268
W_{eng} (t)	27.95	11.85	11.13	8.92
W_{OEW} (t)	97.86	57.9	46.73	40.18
W_{OWE} (t)	116.9	76.9	65.73	59.18
W_{ppl} (t)	892.9	379.2	235.2	181.0
W_{TOGW} (t)	1,010	456.1	300.9	240.1
Propellant ratio (–)	47.0	20.0	12.4	9.53

Design payload is 19 t (41,895 lb) of propellant with a bulk density of 999.4 kg/m^3 (62.4 lb/ft^3)
LACE Liquid Air Cycle Engine; *RBCC* Rocket Based Combined Cycle

(LACE rocket), offers a 57% reduction in the propellant required to deliver the orbital maneuver propellant. It is important to note that this step does not require a technological breakthrough, but only an adaptation of an existing operational propulsion system. A key observation is, even with the best propulsion system for the launcher, it requires 10 lb of launcher propellant to deliver 1 lb of orbital maneuver propellant to LEO. It becomes clear that the orbital maneuver vehicle needs to be a very efficient user of orbital propellant.

In the above exercise, the design payload selected has been 19 t (metric tons). If that payload mass is increased, there is a gradual decrease in the percentage of the propellant required to deliver the orbital maneuver propellant, as shown in the top chart of Fig. 5.4. However, if the payload is instead decreased, the propellant required to deliver the orbital maneuver propellant increases quickly. As shown in the top chart of Fig. 5.4, at 7 t LEO propellant payload, the propellant required to deliver the orbital maneuver

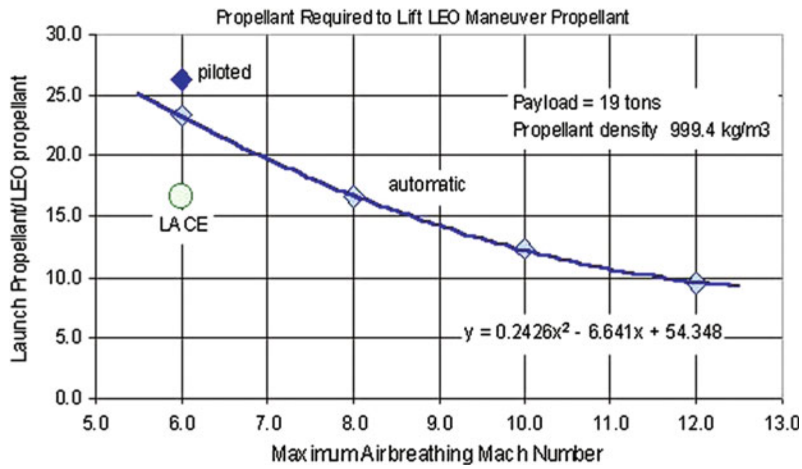


Fig. 5.5 Launch propellant required to lift orbital maneuver propellant to LEO with a rocket ejector ramjet. All-rocket ratio = 47

propellant is 50% greater compared to the 19 t LEO propellant payload case. The correlating curve fit is:

$$\frac{W_{\text{ppl}_{\text{launcher}}}}{W_{\text{ppl}_{\text{LEO}}}} = 3.5531 \cdot W_{\text{pay}}^{-0.4339} \tag{5.7}$$

where W_{pay} is in t.

Orbital maneuvering vehicles (OMVs) are powered by a mix of propulsion systems and propellants. A parametric sizing effort has established the variability of the ratio of launcher propellant to propellant payload with payload propellant bulk density and payload mass. A representative set of orbital maneuver propulsion systems is given in Table 5.4. This is only meant to span a selected range of relevant systems and is by no means all-inclusive or comprehensive. The density I_{sp} (bulk specific gravity, SG, times I_{sp}) is a measure of the relative volume taken by the propellant system. In that respect, the hypergolic propellants take always less volume compared to a hydrogen-fueled system.

For propellant bulk densities greater than 700 kg/m^3 (43.7 lb/ft^3), there is no change in the propellant/payload ratio. That is, the propellant payload volume does not influence how much propellant is required to deliver the orbital maneuver propellant; in contrast, the payload mass has a major impact. For propellant bulk densities less than 700 kg/m^3 (43.7 lb/ft^3), there is an increase in the propellant required to deliver the orbital maneuver propellant. That is, now both, the propellant mass and the volume of the orbital maneuver propellant, determine the size and volume of the launcher. The result is an increase in propellant required to deliver the orbital maneuver propellant, as shown in the bottom chart of Fig. 5.4. The correlation curve fit for propellant bulk densities less than 700 kg/m^3 (43.7 lb/ft^3) is:

$$\frac{W_{\text{ppl}_{\text{launcher}}}}{W_{\text{ppl}_{\text{LEO}}}} = 3.189 - 0.3524 \cdot X + 0.0263 \cdot X^2 \tag{5.8}$$

where $X = \rho_{\text{ppl(LEO)}}$, the propellant density in LEO.

The range of launcher propellant required to lift one mass unit of orbital maneuver propellant to LEO ranges from 47 to 9.5. Compare this to a Boeing 767-200 carrying 216 passengers over a 5800 km distance: The fuel consumed is 2.6 mass units per one mass unit of payload. The oxidizer-to-fuel ratio for the airbreather to Mach 12 is 3.14, and the resulting fuel-to-payload ratio is 3.02. That implies that the airbreathing launcher is only about 16% less efficient in its propulsion system flying to Mach 12 than a Mach 0.85

Table 5.4 Characteristics of space propulsion systems for orbital maneuvering vehicles

	Hypergolic rocket	Hydrogen/oxygen rocket	Solar electric	Nuclear electric
Fuel	Hydrazine	Hydrogen	Lithium	Lithium
Oxidizer	Nitrogen Tetroxide	Oxygen	none	none
Bulk density (kg/m^3)	1229	378.0	533.7	533.7
I_{sp} (s)	290	460	3200	9000-plus
Density I_{sp} (s)	357	174	1705	4797

transport. Concorde, flying 100 passengers at Mach 2.04 over a 6300 km distance, consumed about 8.3 mass units of fuel per one unit payload mass. Consequently, the air-breathing launcher is more efficient than Concorde in terms of fuel usage.

After finding the propellant required to lift the orbital maneuver propellant to LEO, the task remains to establish how much orbital maneuver propellant is required.

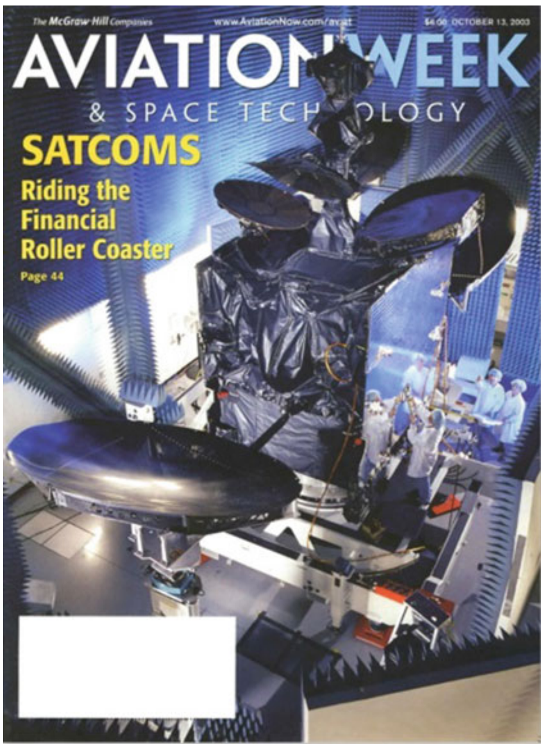


Fig. 5.6 Representative reference satellite (Covault 2003)

5.2.2 Geostationary Orbit Satellite Size and Mass

The first step is to examine a number of GSO satellites from the open literature and determine a representative reference value. The goal is to generate a "reference GSO satellite" that is heavy enough to represent future satellites and then provide a reasonable estimate of the orbital propellant required. Table 5.5 gives the dimensions of the satellite main body with all antennas folded. The mass ratio determined for the "beginning-of-life" mass and the "empty" mass is the propellant required for maintaining the GSO orbit and station-keeping due to orbital precession.

The cover of *Aviation Week & Space Technology* of October 13, 2003, has a picture of the Boeing Satellite Systems 601B for broadcast and broadband multimedia services, see Fig. 5.6 (Covault 2003). This is not unlike the reference satellite listed at the bottom of Table 5.5. Having identified a reference satellite, the next question is how much propellant will be required to change its altitude and orbital inclination?

5.3 Maneuver Between LEO and GEO, Change in Altitude at Same Orbital Inclination

The nominal LEO altitude assumed here is 100 nm (185.2 km) or around 200 km (108 nm). Reaching a higher-altitude orbit is usually a two-step process, as shown in Fig. 5.7 for the GSO example.

For a general elliptical orbit, the lowest altitude is the periapsis and the highest is the apoapsis specifically for selected bodies:

Table 5.5 Characteristics of a number of GSO satellites (Karol 1997)

System	Length (m)	Width (m)	Height (m)	Volume (m^3)	Beginning-of-life (BOL) mass (kg)	Empty mass (kg)
ASTRA-1F	4.51	3.41	2.80	43.2	1803	1279
EHF-7	3.35	3.35	3.35	37.7	1224	868
INTERSAT 707	4.69	2.41	2.19	27.2	3649	1760
APSTAR 1A	6.58	2.16[a]	2.16[a]	24.1	584	414
CHINSAT 7	6.58	2.16[a]	2.16[a]	24.1	557	395
N-STAR-B	3.05	2.40	2.20	27.3	1617	2057
INMARSAT III	2.10	1.80	1.71	16.7	1098	778
AMOS-1	1.22	1.68	1.92	10.5	579	410
Reference	3.40	2.80	2.80	26.7	2267	1608

[a]Diameter, cylindrical configuration

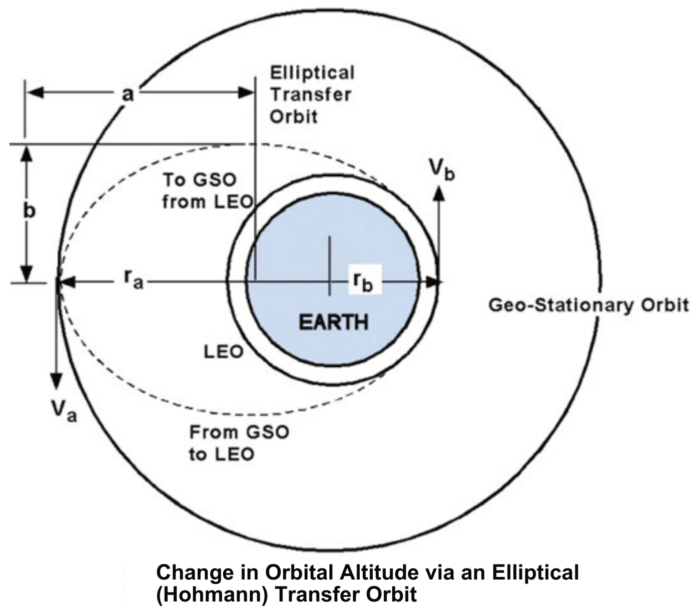


Fig. 5.7 Transfer ellipse to change orbital altitude

General	Sun	Earth	Moon
Periapsis	Perihelion	Perigee	Perilune
Apoapsis	Aphelion	Apogee	Apolune

The first step is an elliptical transfer orbit to the orbital altitude desired, which requires a propulsion burn to leave the low-altitude orbit. The second step is a propulsion burn to match the circular orbital velocity at the desired higher orbital altitude. The process to return to the lower orbital altitude requires a burn to match the elliptical orbital speed at the higher altitude, then a second propulsion burn to match the lower circular orbit speed. This is a minimum energy transfer orbit, or Hohmann Transfer (Logsdon 1997). Equations (5.3a)–(5.3c), Eq. (5.4a) and (5.4b) provide the magnitude of the circular orbital velocity at the desired altitude.

Figure 5.7 shows the geometry for the example elliptical transfer orbit from LEO to GSO. The information needed is the elliptical orbit velocities for the lowest orbital altitude (periapsis) and the highest orbital altitude (apoapsis). The following equations provide the orbital parameters for Kepler's *elliptical* orbits. We obtain for the velocity at the periapsis

$$V_p = \sqrt{\frac{2 \cdot \mu}{R_0 + h_p} + \frac{\mu}{a}} \tag{5.9}$$

The velocity at the apoapsis is

$$V_a = \sqrt{\frac{2 \cdot \mu}{R_0 + h_a} + \frac{\mu}{a}} \tag{5.10}$$

The semimajor axis of the transfer ellipse is given with

$$a = \frac{(R_0 + h_a) + (R_0 + h_p)}{2} \tag{5.11}$$

and the eccentricity, which defines the shape of the orbit, is defined as

$$e = \frac{(r_a - r_p)}{(r_a + r_p)} \tag{5.12}$$

The period of the ellipse is

$$T = 2 \cdot \pi \sqrt{\frac{a^3}{\mu}} \tag{5.13}$$

All Kepler orbits are conic sections. In this general sense, an orbit is a path through space defined by a conic section. There are two closed orbital solutions (circular and elliptical) and two open (not returning) orbital solutions (parabolic and hyperbolic). For a circular orbit, the eccentricity, e, has to be equal to zero. For an elliptical orbit, the eccentricity, e, has to be less than one. For a parabolic orbit, the eccentricity, e, has to be equal to one. For a hyperbolic orbit, the eccentricity, e, has to be larger than one.

To increase orbital altitude, the velocity increments are then

$$\Delta V_1 = V_p - V_{circular,p} \tag{5.14a}$$

$$\Delta V_2 = V_a - V_{circular,a} \tag{5.14b}$$

To decrease orbital altitude, we obtain

$$\Delta V_1 = V_{circular,a} - V_a \tag{5.15a}$$

$$\Delta V_2 = V_{circular,p} - V_p \tag{5.15b}$$

Then, to increase orbital altitude requires a propulsion burn at periapsis to accelerate to elliptical orbit speed, followed by a propulsion burn at apoapsis to increase the spacecraft speed to circular orbit speed at the higher altitude. To decrease orbital altitude, there is a propulsion burn at apoapsis to slow the spacecraft to elliptical orbit speed, followed by a propulsion burn at periapsis to decrease the spacecraft speed and circularize the orbit at the lower altitude. Specifically, transferring from a 100-nm (185.2-km) LEO to a 19,323-nm (35,786-km) GSO orbit (refer to Fig. 5.7 for the geometry of the transfer maneuver and the location of the velocities called out), the orbital velocity for a 100-nm (185.2-km) circular orbit is 25,573 ft/s (7795 m/s). For an elliptical transfer orbit, the orbital velocity at the 100-nm (185.2-km) perigee is 33,643 ft/s (10,254 m/s) and 5235 ft/s (1596 m/s) at the 19,323-nm (35,786-km) apogee. The orbital velocity for a 19,323-nm (35,786-km) circular orbit is 10,088 ft/s (3075 m/s).

5.3.1 Energy Requirements for Altitude Change

Referring to Fig. 5.7, to initiate the transfer maneuver, the spacecraft must be 180° away from the desired point in the GSO orbit. At that point, a rocket burn is required to increase the spacecraft velocity from 25,573 to 33,643 ft/s, an incremental velocity of 8070 ft/s (2460 m/s). The spacecraft is now in an elliptical trajectory toward the 19,323-nm (35,786-km) apogee. When the apogee is reached, the elliptical orbital velocity is 5235 ft/s (1596 m/s), that is slower than the 10,088 ft/s (3075 m/s) required for a GSO circular orbit. Then, at apogee, a rocket burn provides 4853 ft/s (1479 m/s) velocity increment necessary to circularize the orbit, otherwise the spacecraft will continue along its elliptical trajectory. The total velocity increment is 12,923 ft/s (3939 m/s).

In order to return to LEO, the opposite sequence of events is necessary. Again, at the orbital location opposite the location point in the LEO orbit, a retroburn of minus 4853 ft/s (1479 m/s) velocity is necessary to slow the spacecraft to the elliptical orbit apogee velocity of 5235 ft/s (1596 m/s). When approaching the 100-nm altitude, the elliptical orbit speed is approaching 33,643 ft/s (10,254 m/s). In order to achieve a 100-nm circular orbit, a retroburn of minus 8070 ft/s (2460 m/s) is necessary to reach the 100-nm circular orbit speed of 25,573 ft/s (7795 m/s).

For the round-trip described above, a total of four rocket firings are required for a total incremental velocity of 25,846 ft/s (7878 m/s), a velocity greater than the incremental velocity to reach LEO!

We conclude that to change orbital altitude requires the expenditure of energy. The energy amount required depends on the altitude change desired. The incremental velocity required to move from a nominal 100-nm or 200-km orbital altitude is given in Fig. 5.8. The incremental velocity curve is highly nonlinear. A 6000 ft/s (1829 m/s) incremental velocity will permit an altitude change of about 3000 nm (5556 km). However, a burn of twice the velocity increment, 12,000 ft/s (3658 m/s) will permit an altitude change of about 13,000 nm (24,076 km), which is 4.3 times larger compared to the first case.

5.3.2 Mass Ratio Required for Altitude Change

The previous section provides the methodology to determine the magnitude of the incremental velocity to achieve a given orbital altitude change, in a fixed orbital inclination. The propulsion systems described in Table 5.4 provide the specific impulse, I_{sp}, for each of the four systems considered. Since there is no atmospheric drag in space, the ideal weight ratio Eq. (5.16) applies:

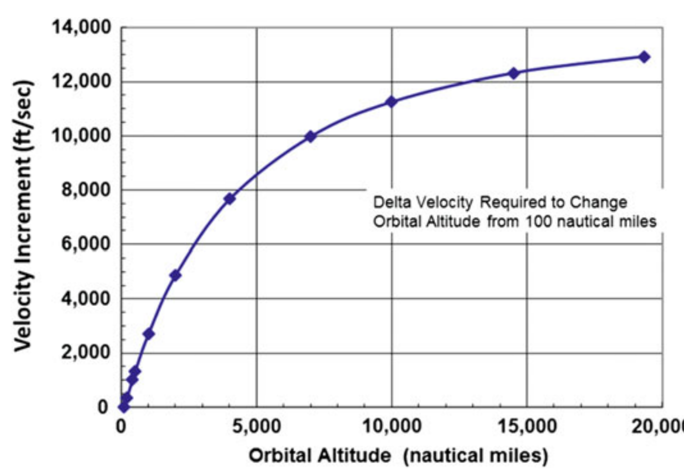


Fig. 5.8 Velocity requirement to change orbital altitude can approach one half of the orbit speed

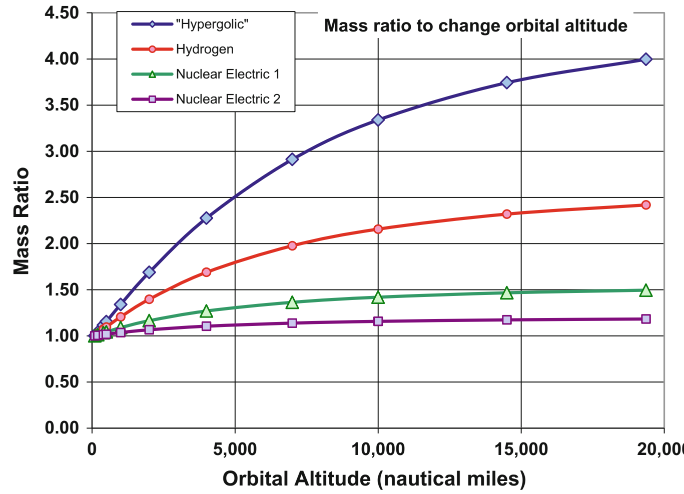


Fig. 5.9 Mass ratio required to change orbital altitude is very dependent on the propulsion system performance (I_{sp})

Table 5.6 Sized orbital maneuver vehicles (OMV) for a one-way mission from LEO to GSO

Propulsion	Gross mass (t)	Propellant (t)	W_{OWE} (t)	W_{OEW} (t)	$W_{\mathrm{R\ one\text{-}way}}$ (–)	Thrust (kN)
Hypergolic	12.01	9.00	3.01	0.738	3.996	58.67
H_2/O_2	7.14	4.16	2.98	0.716	2.418	35.02
Solar electric	4.80	1.59	3.21	0.945	1.134	4.71
Nuclear electric	3.60	0.345	3.25	0.985	1.046	3.53

$$W_{\mathrm{R}} = \frac{\Delta V}{g \cdot I_{\mathrm{sp}}} \tag{5.16}$$

Translating the incremental velocity data and specific impulse data into weight ratio yields Fig. 5.9. The weight ratio for the four propulsion systems described in Table 5.4 is provided as a function of orbital altitude. The weight ratios for the LEO to GSO orbital altitude change are as follows: (1) 4.00 for the hypergolic propulsion system; (2) 2.39 for the oxygen/hydrogen propulsion system; (3) 1.55 for the solar electric propulsion system; and finally, (4) 1.11 for the nuclear electric propulsion system. The acceleration specified for the chemical rocket-powered OMV is 0.5 g. For the electric thruster-powered OMV, the acceleration is 0.1 g.

The gross weight of the one-way OMV is straightforward, and the sizing program balances the propellant required versus the capacity of the propellant tank that determines the operational empty weight (W_{OEW}). The sized OMV for each of the propulsion systems transporting a 5000 lb (2268 t) satellite is given in Table 5.6.

The gross weight for the one-way mission is:

$$W_{\mathrm{TOGW}} = W_{\mathrm{R}} \cdot (W_{\mathrm{OEW_{OMV}}} + W_{\mathrm{satellite}}) \tag{5.17}$$

$$W_{\mathrm{ppl}} = (W_{\mathrm{R}} - 1) \cdot (W_{\mathrm{OEW_{OMV}}} + W_{\mathrm{satellite}}) \tag{5.18}$$

Note that the operational empty weight (W_{OEW}) is essentially constant. It is larger for both electric propulsion configurations because of the solar panels for the solar electric and the radiators for the nuclear electric. As in the case for the launchers, the primary difference in the weights and thrust values is the result obtained for the carried propellant. The propellant mass for the hypergolic rocket is 34 times larger compared with the nuclear electric rocket. The propellant load required is reduced by (a) the increasing specific impulse, I_{sp}, of the propulsion system and (b) the reduction in mass and thus engine thrust and propellant flow rate.

Unlike the space launcher, where the payload is about 1/7 of the W_{OEW}, for the orbital maneuver vehicle (OMV) the payload is larger than the W_{OEW}. The W_{OEW} differs from "empty" or "dry" weight in that all of the fluid lines are filled and any trapped fluids or propellants are included in the W_{OEW}. The operational weight empty (W_{OWE}) is the W_{OEW} plus the payload. That is, the vehicle operationally is ready but without the propellants loaded. The satellite (payload) weight for the OTV is 2.268 t. The Russian *Progress* capsule can deliver 3.5 t to LEO, and the European Space Agency (ESA) Automated Transfer Vehicle (ATV) can deliver 7.67 t to the ISS orbital altitude of 249 nm (Catchpole 2008).

Table 5.7 Payload size versus OMV for a hypergolic propulsion system with a one-way mass ratio of 4

W_{Pay} (t)	W_{GW} (t)	W_{ppl} (t)	W_{OEW} (t)	W_{OWE} (t)	T (kN)
2.268	12.01	8.991	0.735	3.02	58.7
3.50	18.32	13.71	1.106	4.61	89.9
3.650	19.08	14.28	1.148	4.80	93.6
4.00	20.86	15.61	1.245	5.24	102
4.50	23.38	17.50	1.380	5.88	115
5.50	28.40	21.25	1.641	7.14	139
6.50	33.36	24.97	1.891	8.39	164
7.50	38.28	28.65	2.130	9.63	188

If the OMVs in Table 5.6 are extended to different payload masses for the hypergolic propulsion system, the size and mass trends can be established as given in Table 5.7.

For payloads larger than 4.9 t, the 19 t of propellant payload delivered to LEO by the tanker launcher is insufficient for a LEO to GSO mission. This is shown for hypergolic propulsion, because as advanced propulsion enters orbital operations, the propellant requirement will substantially reduce, even for the heavier payloads. The propellant load scales as the mass ratio minus one. Then, for the nuclear electric propulsion system, the propellant load for the 7.5 t payload OMV is only 1.07 t, and for the solar electric propulsion system, it is 4.71 t. However, as long as the principal orbital maneuver propellant of choice is hypergolic, the orbital propellant requirements will steadily increase. The ESA ATV meets a current need. With the Space Shuttle retired, a more substantial thrust OMV is required to reboost the International Space Station (ISS), while some mechanism is required to provide service capability to the Hubble Space Telescope if necessary. If the Hubble Space Telescope was to be placed at the same orbital inclination as the ISS, but at a higher altitude, the Hubble Space Telescope could be serviced from the ISS without the need for an equivalent Space Shuttle system.

The gross weight of the two-way OMV is more complex, because the OMV must carry the return-to-LEO propellant to GSO in the first place. The sizing program balances the total propellant required versus the capacity of the propellant tanks that determines W_{OEW}. The sized OMVs for each of the propulsion systems transporting a 5000 lb (2.268 t) satellite are given in Table 5.8.

The gross weight for the two-way mission is:

$$W_{GW} = [W_R \cdot (W_{OEW_{OMV}} + W_{satellite})] \cdot W_R \tag{5.19a}$$

$$W_{GW} = W_{OEW_{OMV}} \cdot W_R^2 + W_R \cdot W_{satellite} \tag{5.19b}$$

The propellant weight for the two-way mission is:

$$W_{ppl_{to\,LEO}} = (W_R - 1) \cdot W_{OEW_{OMV}} \tag{5.20}$$

$$W_{ppl_{to\,GSO}} = [W_R \cdot W_{OEW_{OMV}} + W_{satellite}] \cdot (W_R - 1) \tag{5.21}$$

As would be expected, the to-GSO and return OMV is significantly larger than the one-way vehicle, see Table 5.8. Other than being larger, the same comments apply to the two-way OMV as the one-way OMV. Launching to GSO with the current multistage rockets, the propellant in the upper stage (usually third stage) contains the propellant for the elliptical GEO stationary transfer orbit, and the GSO circularization propellant is carried in the GSO satellite. Sizing the one-way mission gives some indication of the upper stage propellant mass required to place the payload into GSO transfer orbit. Given the function of the OMV, the two-way mission is the logical sizing mission.

With a conventional rocket-powered OMV (for instance, the *Jupiter* tug proposed by Lockheed Martin), rocket engines of approximately the correct thrust are available. For example, a hypergolic restartable rocket in the 50–60 kN range is available from the Ukrainian Yuzhnoye Design Office and organization and is the YUZ-U-29 rocket propulsion system for the Tsyklon launcher. The specific impulse is 289 s for a total installed engine thrust-to-weight ratio 49.1 and a thrust of 56 kN. The hydrogen/oxygen

Table 5.8 Sized OMVs for the two-way mission from LEO to GSO to LEO

Propulsion	W_{GM} (t)	W_{ppl} (t)	W_{OWE} (t)	W_{OEW} (t)	$W_{R\ Two\text{-}Way}$ (–)	T (kN)
Hypergolic	27.07	23.70	3.37	1.10	16.00	119.5
H_2/O_2	10.98	7.79	3.19	0.925	5.71	53.83
Solar electric	5.99	2.59	3.39	1.12	2.22	5.87
Nuclear electric	3.79	0.494	3.30	1.03	1.23	3.72

rocket in the 35 kN range is available both from the USA and from the former USSR. The collaboration of NPO Energomash, Khimki, has produced a LOX/LH_2 in-development engine of the correct thrust level, the ENM-C-36. The specific impulse is 461 s. The Pratt & Whitney RL10 is also an upper stage candidate. As the RL10 is an expansion turbine cycle, its potential operational life is very long compared to a conventional rocket engine.

Electric-powered engines for the solar electrical and nuclear electrical propulsion systems are a challenge in that there are no engines or engine clusters available in the thrust class required. The largest electric thrusters are in the former Soviet Union and are about 1 N in gross thrust! At 1/10 *g* acceleration, the total velocity increment of 12,923 ft/s (3939 m/s) is achieved in 1.11 h. At 1/100 *g* acceleration, the time required is 11.16 h, and this choice of acceleration would reduce the thrust to the 5–6 kN range, at the expense of the maneuvering time that would increase correspondingly.

The only future electric thrusters that appear capable of such thrust levels are magnetoplasmadynamic (MPD) thrusters, e.g., the VASIMR (Variable Specific Impulse Magnetoplasma Rocket) engine, see Chap. 7 (Diaz 2000). It may not be possible to fabricate solar panels of the size necessary to drive an electric thruster in the 5–6 kN thrust level, given the low-energy conversion efficiency of solar panels. A 0.57 N thruster with a 50% energy conversion efficiency (still unavailable) would require an input from the solar panels of about 30–40 kW. A 5700 N thruster, by analogy, would require an input of some 300 MW to 400 MW, an unheard of power level for solar panels. The largest multimedia communication satellites carry solar panels of 5–6 kW total power. This would be 1000 times greater. At that power level, to reach an incremental velocity of 12,923 ft/s (3939 m/s), the acceleration time is 46.5 days, operationally practicable for some GSO operations but unacceptable to commercial TLC (Telecommunications) operators. An order of magnitude increase in thrust to 5.7 N would reduce the transit time to 4.6 days, a more acceptable level. Consequently, that may be the first objective in developing thrusters for the solar electric OMV.

We now have both the quantity of launcher propellant required to deliver the OMV propellant to LEO and the OMV propellant required, in each of the three orbital maneuver missions. We are now in the position to determine the total mass units of propellant (launcher and OMV) required per unit mass of the satellite for each of the four space propulsion systems.

5.3.3 Propellant Delivery Ratio for Altitude Change

In Fig. 5.10, the ratio of the total mass units of propellant (launcher and OMV) required per unit mass of the satellite is presented for the four in-space propulsion systems and the four launcher propulsion systems, namely those in Table 5.9.

Figure 5.10 shows the dramatic reduction in the total propellant mass (launcher and OMV) required per unit mass of the satellite, for the two electric OMV propulsion systems, by advancing the performance of the launcher propulsion system. By incorporating a LACE system into an existing hydrogen/oxygen rocket, the propellant required to deliver 1 mass unit of propellant to LEO is reduced by 56%. Proceeding to a Mach 12 ram/scramjet produces another 50% reduction in the required propellant to deliver 1 mass unit of propellant to LEO. Clearly, instead of the 190.5 mass units of propellant required, LACE reduces that number to 83.1, and a Mach 12 ram/scramjet further reduces that number to

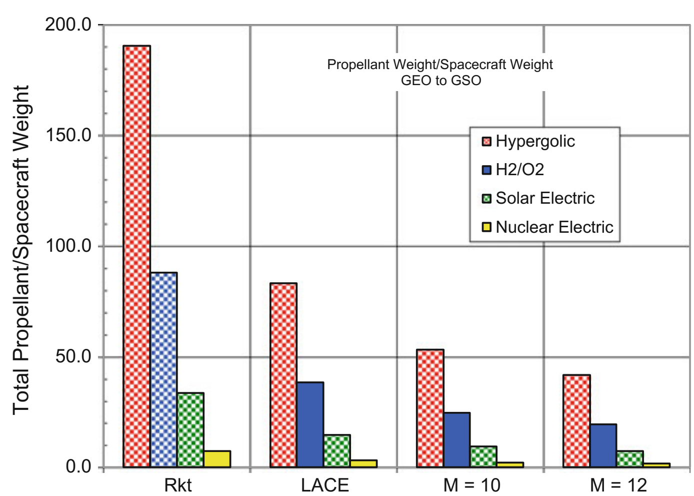


Fig. 5.10 Ratio of total propellant weight/satellite weight

41.8 propellant mass units required to deliver 1 mass unit of propellant to LEO. However, the real gains occur when propulsion of both launcher and OMV propulsion is improved.

Figure 5.11 focuses on the electric propulsion for the OMV and the more efficient launcher propulsion systems. The propellant required to deliver 1 mass unit of propellant to LEO is between 3.5 and 0.5. Then, it becomes practicable to deliver propellant to LEO, as the propellant cost is no more than the propellant to deliver a unit mass of payload in a commercial transport. Although it is nearly prohibitive in terms of hypergolic space rockets and conventional launch rockets to deliver significant quantities of orbital maneuver propellant to LEO (the actual figure is 190.5 kg of propellant per kilogram of LEO propellant delivered), the future holds a dramatic reduction in that quantity by a factor of about 20 just by using hydrogen/oxygen propulsion in space, and a combination of hydrogen/oxygen rocket and airbreathing propulsion for the launcher. With space electric propulsion and the hydrogen/oxygen rocket, plus airbreathing propulsion for the launcher, that ratio can be reduced to the range 1–3 kg of burnt propellant per kilogram delivered to orbit. At this point, the orbital tanker is now competitive with a KC-135 or KC-46 for refueling missions.

5.4 Changes in Orbital Inclination

An orbital plane change is a much more challenging propulsion space maneuver than an orbit change. A large expenditure of energy is required to achieve a small change in the orbital plane. A propulsive plane change is an impulse turn and is executed with the thrust line perpendicular to the orbital path and in the direction of the plane change. The incremental velocity for an impulse turn is given by Eq. (5.22) for a non-rotating Earth:

$$\Delta V_{pc} = 2 \cdot \sqrt{\frac{\mu}{R_0 + h}} \cdot \sin\left(\frac{\alpha}{2}\right) \tag{5.22}$$

with the standard gravitational parameter for Earth defined as

$$\mu = 1.407645 \cdot 10^{16}\ \frac{\text{ft}^3}{\text{s}^2} \tag{5.23}$$

and with the average radius of Earth

$$R_0 = 3442.5\ \text{nmi} \tag{5.24}$$

As indicated in Eq. (5.22), the higher the orbital altitude, the smaller the incremental velocity for a given plane

Table 5.9 Launcher and OMV propulsion options

Launcher propulsion	OMV propulsion
Hydrogen/oxygen rocket based on the P&W XLR-129	Hypergolic, restartable, long-life rocket closed turbopump cycle rocket
LACE rocket based on the P&W XLR-129	Hydrogen/oxygen restartable, long-life expander or closed-cycle rocket
Rocket ejector ram/scramjet to M = 10 + hydrogen/oxygen rocket	Electric MHD thruster with lithium fuel powered by solar panels
Rocket ejector ram/scramjet to M = 12 + hydrogen/oxygen rocket	Electric MHD thruster with lithium fuel powered by nuclear reactor

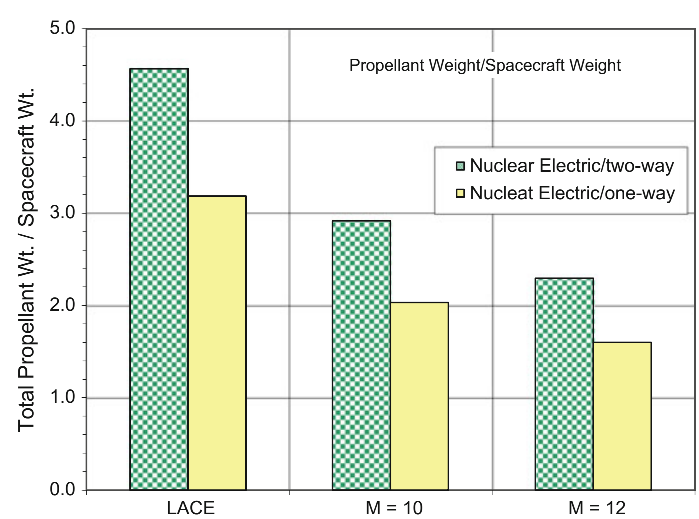


Fig. 5.11 Ratio of total propellant weight to satellite weight for two electric propulsion systems

change. Travelling to that higher orbital altitude requires propellant, as we have just seen in the previous section. Consequently, there is an opportunity for a trade-off, depending on whether or not the change in orbital altitude propellant plus that of the reduced plane change impulse turn is less than the propellant consumed by a dip to lower altitude followed by an aerodynamic turn plane change and pull-up. From Eq. (5.22), the incremental velocity per 1° change in orbital plane is about 446 ft/s (135.9 m/s) at an orbital altitude of 100 nm. Then, a modest 5° plane change requires an incremental velocity of 2230 ft/s (679.7 m/s).

The right sketch in Fig. 5.12 depicts an orbital plane change in LEO and in a higher-altitude elliptical orbit. In order to accomplish this, a rocket burn is required to put the spacecraft into the elliptical orbit, then at apoapsis a rocket burn to rotate the orbital plane and a final rocket burn to return the spacecraft to the lower-altitude circular orbital speed. As we shall see, there is an angle above which this procedure requires less incremental velocity than a lower orbital altitude plane change.

The left sketch in Fig. 5.12 depicts an orbital plane change in LEO performed by entering the Earth's upper atmosphere with a high lift-to-drag ratio hypersonic glider and executing a thrust-equals-drag aerodynamic turn at maximum hypersonic aerodynamic glide ratio, $(L/D)_{\max\ \mathrm{hypersonic}}$. This maneuver requires a hypersonic glider, but it enables a much larger orbital plane change for the same

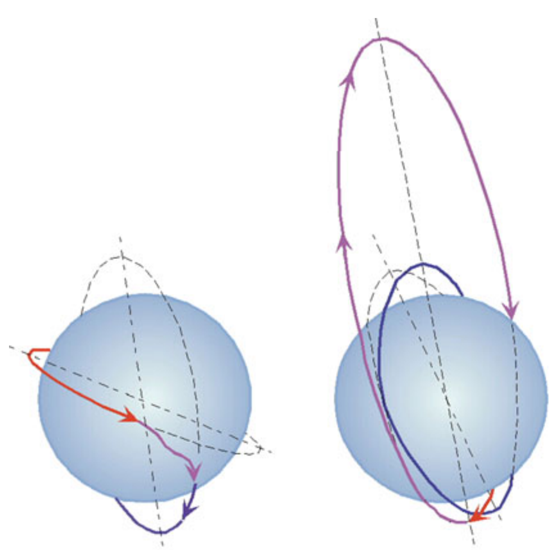


Fig. 5.12 Orbital plane change via an aerodynamic turn in the upper atmosphere (*left*) and an impulse turn executed during an elliptical transfer orbit to 22,400-nm orbit (*right*)

propellant consumed. With conventional rocket propulsion, this method of changing the orbital plane uses always less energy. This was first analyzed and presented by Dr. Wilbur Hankey in 1959 when at the Air Force Flight Dynamics Laboratory at Wright-Patterson Air Force Base, Ohio (Dr. Hankey has been later an Emeritus Professor with the Wright State University in Dayton, Ohio) (Hankey 1988).

5.4.1 Energy Requirements for Orbital Inclination Change

Using Eq. (5.22), the variation in incremental velocity with altitude as a function of plane change angle is given in Fig. 5.13 for five orbital altitudes, from 100 nm (185.2 km) to 19,323 nm (35,786 km). For a 90° plane change at 100-nm orbital altitude, the incremental velocity is just over 35,000 ft/s (10,668 m/s). Compare that to the incremental velocity for the orbital altitude change from 100 to 19,323 nm of 12,900 ft/s (3992 m/s) in Fig. 5.8. As a consequence, the incremental velocity requirements for an orbital *plane* change are much more demanding than an orbital *altitude* change. For an incremental velocity of 12,900 ft/s, an orbital plane change of about 29° is possible. Overall, that is a smaller plane change than required to move from the latitude of NASA Kennedy Space Center (KSC) to the latitude of the International Space Station (ISS).

Shown in Fig. 5.14 is the ΔV for an impulse turn made from the GSO orbital altitude of 19,323 nm (35,786 km), which requires about 11.5 h to execute. This is one of the lower-energy solutions for the plane change. Increasing the altitude of the impulse turn to 36,200 nm (67,042 km) decreases the incremental velocity to about 1000 ft/s (304.8 m/s), but increases mission time to 24 h. As shown, the breakeven orbital plane change is about 50°. Then, if the orbital plane change is less than 50°, it is best executed from the spacecraft's orbital altitude without any orbital altitude change. However, there remains the interesting possibility of using aerodynamics to change the orbital plane (Chudoba et al. 2011; Cerro et al. 2012).

The aerodynamic plane change requires slowing the hypersonic glider to about 22,000 ft/s (6706 m/s) to enter the upper atmosphere between 240,000 and 260,000 ft (73,152–79,248 m) altitude. At that point, the rocket engines are ignited and a thrust-equals-drag ($T = D$) turn through the orbital plane change angle desired, and at the lift coefficient corresponding to maximum hypersonic (L/D), is initiated. The aircraft is then leveled at the correct orbital heading, and the engines are reignited to regain orbital velocity. For the class of hypersonic gliders evaluated, this maneuver requires a total velocity increment of about 1022 ft/s (311.5 m/s) to dip into the atmosphere, turn aerodynamically, and pull up to

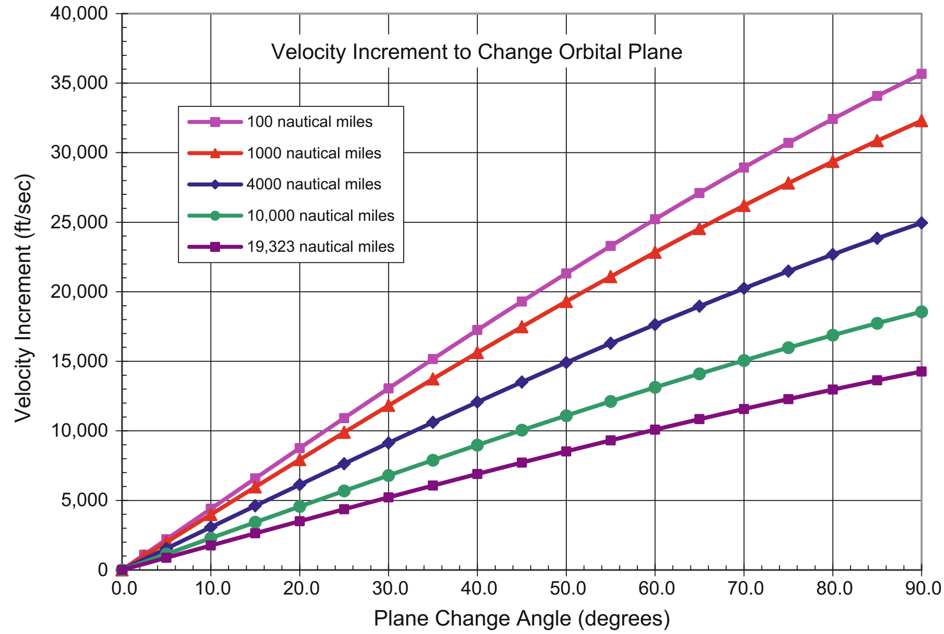


Fig. 5.13 Velocity increment to rotate orbital plane for different orbital altitudes. Higher altitude requires less energy

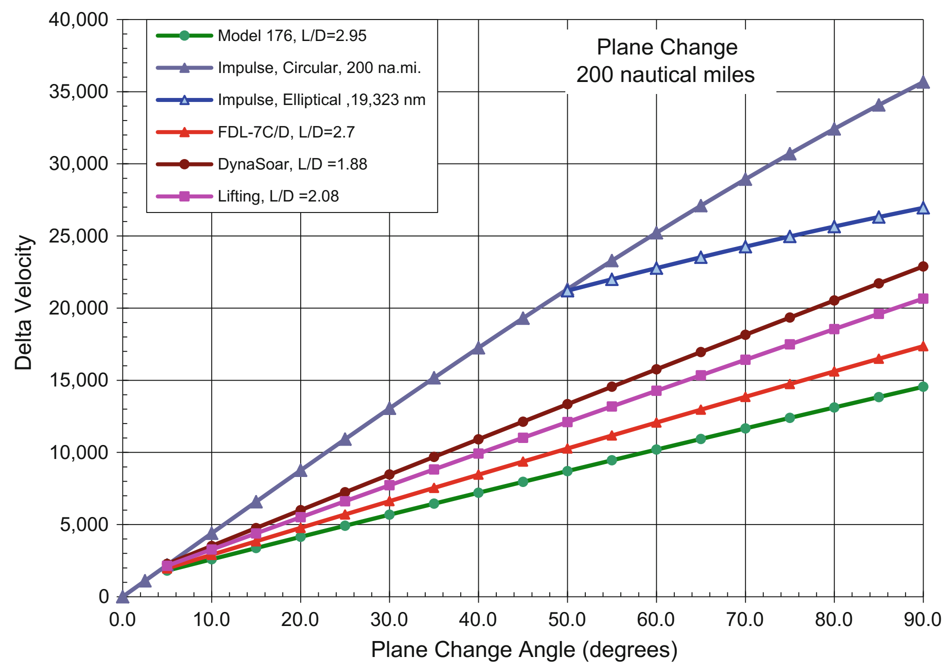


Fig. 5.14 Velocity increment as a function of turn method and plane angle change

the initial orbital altitude and speed. The incremental velocity required to execute the orbital turn is a function of the lift-to-drag ratio, as presented in Fig. 5.14 where it is compared to an orbital impulse turn.

The lift-to-drag ratio at Mach 22 varies from 1.88 to 2.95 for the four hypersonic gliders presented. This performance can be represented as a curve fit as follows:

$$\Delta V_{\text{turn}} = 1022 + C \cdot \left(\frac{L}{D}\right) - 0.0883 \cdot \left(\frac{L}{D}\right)^2 \text{ (ft/s)} \quad (5.25\text{a})$$

with

$$C = 2317.2 - 2545.6 \cdot \left(\frac{L}{D}\right) + 1040.9 \cdot \left(\frac{L}{D}\right)^2 - 144.45 \cdot \left(\frac{L}{D}\right)^3 \quad (5.25\text{b})$$

As shown in Fig. 5.14, the aerodynamic plane change requires significantly less energy compared to the impulse turn. For the *Model 176* hypersonic glider configuration, see Chap. 3, the incremental velocity required is about 40% of the impulse turn requirement. Even a rather modest *X-20 Dyna-Soar* lift-to-drag ratio of 1.88 offers a plane change requirement of around 60% of the incremental velocity required by an impulse turn. The Space Shuttle Orbiter had a lift-to-drag ratio of about 1.5, and the Russian Buran had about 1.7. For the Space Shuttle and Buran orbiter configurations, blunt wing leading edges and nose reduced their hypersonic lift-to-drag ratio well below two.

Figure 5.15 depicts a USAF Flight Dynamics Laboratory FDL-7 C/D glider making a plane change to rendezvous with another orbital vehicle seen in the distance (top right corner of image). In actuality, the rocket engines would be firing, but the artist omitted the engine plume to clarify the orientation of the maneuver. The hypersonic glider is generally a second stage of a TSTO vehicle sized as an automatic OMV, specifically for plane change maneuvers. The design payload is the same as for the space OMV, a 2268 kg (5000 lb) payload. A traditional in-space OMV (a space tug) cannot enter the Earth's atmosphere; hence, it is limited to space operations only. In contrast, the hypersonic glider has the capability to enter the atmosphere when needed to operate as an optional rescue vehicle. The hypersonic glider has an Earth's circumference glide range and can return to Earth without any prior preparation or waiting in orbit. With a payload bay of 36.5 m^3 (1289 ft^3) capacity, it can accommodate nine to twelve persons in pressure suits in an emergency situation.

Table 5.3 provides the specific impulse, I_{sp}, for each of the four OMV propulsion systems. For operation in the space environment, since there is no atmospheric drag, the ideal weight ratio equation applies, see Eq. (5.16). In contrast, the hypersonic glider does experience about an 8% reduction in I_{sp} due to atmosphere drag during the aerodynamic turn maneuver. Combining the incremental velocity and specific impulse data into weight ratio yields Fig. 5.16.

5.4.2 Mass Ratio Required for Orbital Inclination Change

Figure 5.16 presents the mass ratio for the four propulsion systems described in Table 5.4 and the four hypersonic gliders indicated in the column headings. Note that (1) with the hypergolic propellant, the mass ratio quickly becomes impracticable. The curve was terminated at a mass ratio of 10 and a 50° plane change. (2) With the hydrogen/oxygen rocket, the same mass ratio permits an 85° plane change. Extending the time for the plane change by transitioning to an elliptical transfer orbit and executing the plane change at 19,323 nm (35,786 km) GSO orbital altitude reduces the mass ratio to 6 for a 90° plane change. (3) The solar electric propulsion system and the nuclear electric propulsion system, when vehicle-integrated to perform aerodynamic plane changes, provide the only practicable mass ratios for an operational infrastructure. The mass ratios for a 90° orbital turn are between 11 and 5. The mass ratios for the 32° impulse turn orbital plane change are 4.53 for the hypergolic engine, 2.62 for oxygen/hydrogen, 1.15 for solar electric, and 1.05 for nuclear electric, as shown in Table 5.10. The acceleration specified for the chemical rocket-powered OMV is 0.5 g. For the electric thruster-powered OMV, the acceleration is 0.1 g.

The gross weight of the plane change OMVs is straightforward, and the sizing program balances the propellant required versus the capacity of the propellant tank that determines W_{OEW}. The sized OMVs, for each of the propulsion systems transporting a 5000 lb (2.268 t) satellite,

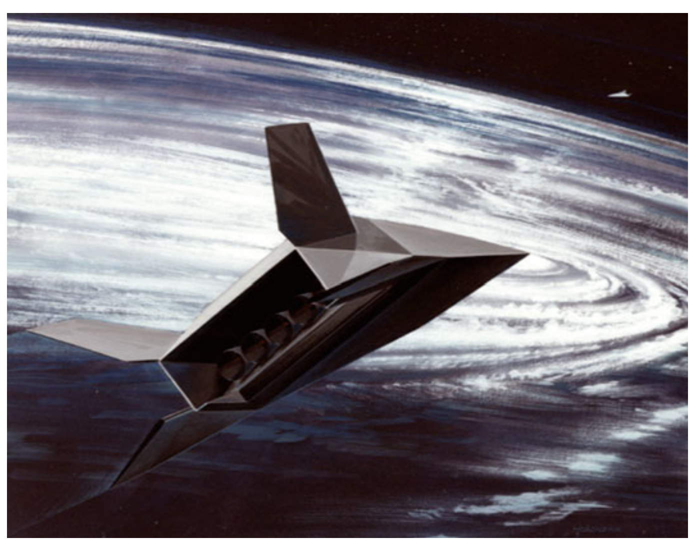


Fig. 5.15 A notional space glider based on the FDL-7 configuration performing an aerodynamic orbital plane change

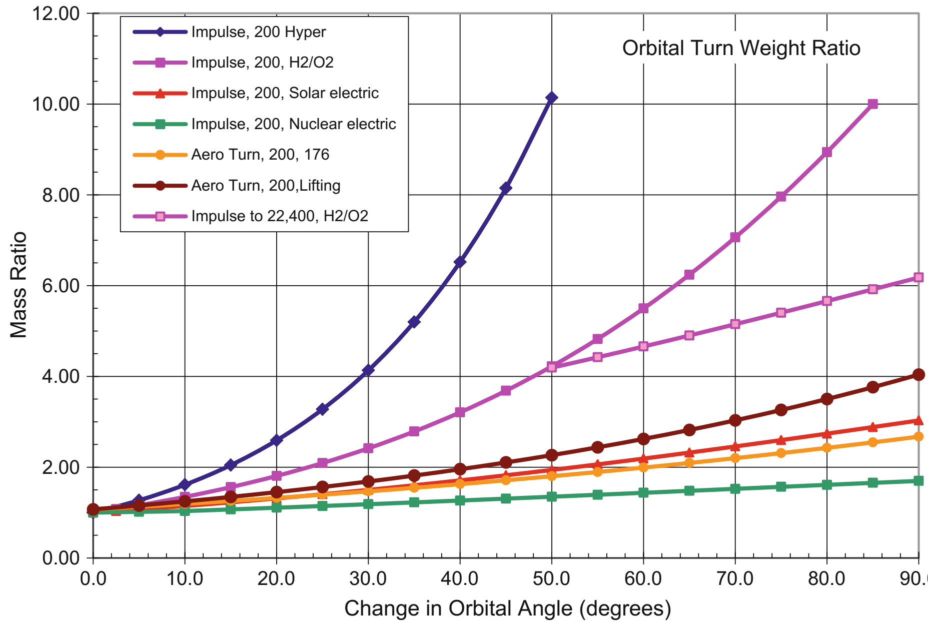


Fig. 5.16 Mass ratio requirements for orbital plane change

Table 5.10 Sized OMV for a 32° plane change at 200-km altitude for a 2,268 kg satellite

Propulsion	W_{GW} (t)	W_{ppl} (t)	W_{OWE} (t)	W_{OEW} (t)	W_R (–)	T (kN)
Hypergolic	13.83	10.78	3.05	0.786	4.529	67.8
H_2/O_2	7.82	4.80	3.02	0.716	2.619	38.3
Solar electric	5.38	1.91	3.47	1.20	1.147	10.6
Nuclear electric	3.82	0.397	3.42	1.15	1.050	7.49

are given in Table 5.10. The gross weight for a single mission is:

$$W_{GW} = W_R \cdot (W_{OEW_{OMV}} + W_{satellite}) \quad (5.26)$$

The propellant weight for the single mission is:

$$W_{ppl} = (W_R - 1) \cdot (W_{OEW_{OMV}} + W_{satellite}) \quad (5.27)$$

Note that the operational empty weight (W_{OEW}) of the OMV is essentially constant. It is larger for the electric propulsion configurations, because of the solar panels for the solar electric propulsion system and the radiators for the nuclear electric propulsion system, see Chap. 7. As in the case for the launchers, despite varying weight and thrust values, the primary differences are the weight and volume required for the carried propellant. The propellant mass for the hypergolic rocket is 27 times larger when compared with the nuclear electric rocket. Again, the propellant load is reduced with increasing I_{sp} of the propulsion system, and the resulting reduction in mass, subsequent engine thrust and propellant flow rate. Unlike the space launcher, where the payload is about 1/7 of the W_{OEW}, for the OMV the payload is larger than the W_{OEW}. The W_{OEW} differs from empty or dry weight in that all of the fluid lines are assumed filled, and any fluids or propellants trapped are included in the W_{OEW}. The operating weight empty, W_{OWE}, is the operating empty weight, W_{OEW}, plus the payload and crew, overall resembling the vehicle operationally ready but without the propellants loaded.

The ideal hypersonic glider for plane change maneuvers is usually a second stage of a TSTO vehicle sized as an automatic OMV specifically for plane change maneuvers. The design payload assumed here is 2.268 t (5000 lb). With a mass ratio of 1.603, the OMV is sized for a 32° plane change capability, the same as the impulse turn OMV. The size and mass characteristics are given in Table 5.11. At Mach 22, the glider has a hypersonic *L/D* of 2.70. At this speed, the glider is in orbit acting as a plane change OMV. An alternate design is shown with a design payload to accommodate the heaviest satellite in Table 5.5, which has a beginning-of-life (BOL) weight of 3650 kg. The vehicle scales as the square-cube law, as the ratio of masses, 1.609, is just slightly greater than the ratio of areas 1.354 raised to the 3/2 power, that is 1.576. As would be expected, the W_{OEW} ratio 1.362 scales with the area ratio.

Because the glider is a hypersonic glider and not just a space structure, it requires more resources to construct and operate. However, it is the only OMV with a true escape and rescue capability for an orbital facility crew. It might be better to design the glider to more demanding requirements,

Table 5.11 Hypersonic glider (FDL-7 C/D) for 32° plane change at 200-km altitude

$W_{pay\ satellite}$ (t)	W_{GW} (t)	W_{ppl} (t)	W_{OEW} (t)	W_{OWE} (t)	S_{plan} (m^2)	T (kN)
2.268	8.33	3.13	2.93	5.20	42.33	40.8
3.650	12.15	4.61	3.99	7.64	57.33	60.1

Table 5.12 Hypersonic glider (FDL-7 C/D) for variable-degree plane change at 200 km and 2.268 t satellite

Plane change	W_{MR} (–)	W_{ppl} (t)	W_{OEW} (t)	W_{OWE} (t)	W_{GW} (t)	S_{plan} (m^2)
90.0	3.228	14.69	4.33	6.60	21.29	59.59
62.0	2.313	7.57	3.49	5.76	13.13	49.29
32.0	1.603	3.13	2.93	5.20	8.33	42.33
32.0	1.603	3.47	3.49	5.76	9.23	49.29

so it can have a more versatile operational life. Table 5.12 gives the sizing of a hypersonic glider with a 2268 kg payload for three plane change capabilities. Increasing the plane change capability from 32° to 62° (+93.8%), the W_{OEW} increases just 19.1%. W_{OEW} and W_{dry} determine the cost of the spacecraft. W_{GW} determines the operational cost. In this case, the W_{GW} is 57% larger. Designing for a larger plane change capability like 62°, but operating at a 32° plane change, has only a minimal increase in the resources required over a spacecraft specifically designed for a 32° plane change, see the last two rows of Table 5.11. It would be practicable and highly desirable to design for greater operational capability. Since the hypersonic gliders are designed to operate with hydrogen/oxygen propellants, the availability of engines is not critical; a number of engines from either the USA or Russia are suitable.

We now have determined both, (a) the quantity of launcher propellant required to deliver the OMV propellant to LEO and (b) the OMV propellant required in each of three orbital maneuver missions. At this point, we can determine for each of the four space propulsion systems the total mass units of propellant (launcher and OMV) required per unit mass of the satellite.

5.4.3 Propellant Delivery Ratio for Orbital Inclination Change

For the OMV impulse turn, Fig. 5.17 shows the dramatic reduction in the total propellant mass (launcher and OMV) required per unit mass of the satellite by advancing the performance of the critical component, the launcher propulsion system. Incorporating a LACE system into an existing hydrogen/oxygen rocket, the propellant required to deliver one mass unit of propellant to LEO is reduced by 56%. Proceeding to a Mach 12 ram/scramjet produces another 50% reduction in the required propellant to deliver one mass unit of propellant to LEO. Clearly, instead of the 228.2 mass units of propellant required to deliver one mass unit of propellant to LEO, the LACE launcher propulsion system reduces that number to 99.6, and a Mach 12 ram/scramjet launcher propulsion system further reduces that to 50.0 propellant mass units. However, the real advances occur when *both*, the launcher *and* the OMV propulsion systems, are improved.

Similar to Fig. 5.11, Fig. 5.18 focuses on the electric propulsion system for the OMV and the more efficient launcher propulsion systems. In this case, the propellant/satellite weight ratio required to deliver one mass unit of propellant to LEO is between 4.5 and 2. Subsequently, delivering propellant to LEO is no longer impracticable as the cost of propellants burnt is comparable with that of delivering a unit mass of payload in a commercial transport aircraft.

In contrast, utilizing conventional hypergolic space rockets and conventional expendable launch rockets for delivering significant quantities of orbital maneuver propellant to LEO is still prohibitive (228.2 kg of propellant per kilogram of LEO propellant delivered). Clearly, substantial improvements are enabled when (a) using the hydrogen/oxygen propulsion system in space, (b) when using the hydrogen/oxygen rocket in combination with the airbreathing propulsion system for the launcher, and (c) with the application of electric propulsion in space and the hydrogen/oxygen rocket and airbreathing propulsion for the launcher, that ratio can be reduced to a figure of about 3 or maybe 2. In short, the orbital tanker is now competitive with a KC-135 or the more modern KC-46 for refueling missions.

Since the hypersonic glider is part of a VTHL TSTO launch system, the first stage is used only once to launch the glider. After reaching orbital altitude and inclination, the space propellant tankers are used to replenish its operational propellants. Table 5.13 gives the propellant to satellite weight ratio for a *FDL-7C/D* hypersonic glider and two satellite weights. The *Model 176* hypersonic glider would have an even smaller value of this ratio, while the *X-20 Dyna-Soar* and the lifting body would have a larger value of the ratio. This table corresponds to the values in Table 5.11.

The hypersonic glider is capable of larger and less expensive plane changes. As we have seen in Table 5.12, the increase in capability is possible for a reasonable investment in vehicle size. This table corresponds to the values in

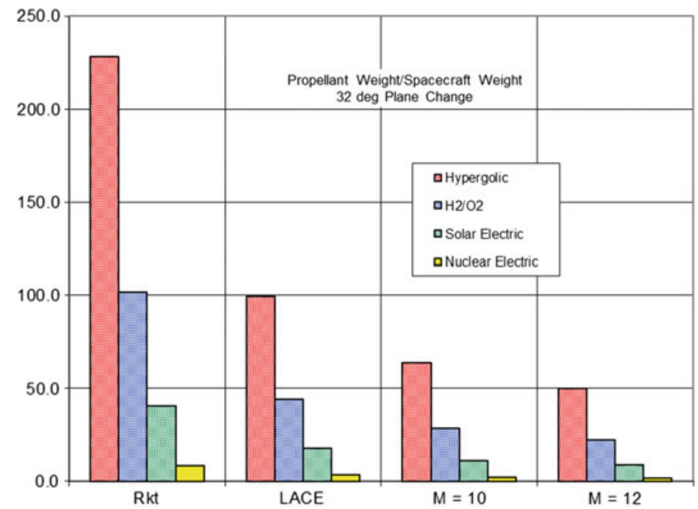


Fig. 5.17 Ratio of total propellant weight to satellite weight

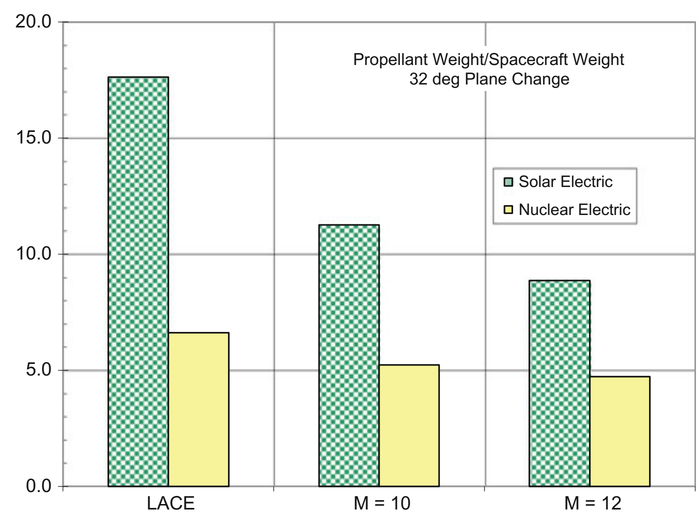


Fig. 5.18 Ratio of total propellant weight to satellite weight for solar and nuclear electric propulsion

Table 5.13 Ratio of total propellant weight to satellite weight for an FDL-7C/D hypersonic glider with a 32° plane change capability and two satellite weights

$W_{\text{Satellite}}$ (kg)	Launcher propulsion			
	Rocket (–)	LACE (–)	M = 10 (–)	M = 12 (–)
3650	60.6	26.5	16.9	13.3
2268	73.5	32.1	20.5	16.1

Table 5.11 for three levels of design for the plane change hypersonic glider. As shown in Table 5.12, the last row in Table 5.14 is for the 62° orbital plane change design spacecraft operating a 32° plane change.

Observations pertaining the OMV results are as follows:

- It is clear that the better propulsion system of the orbital tanker results in reduced resources required to transport the propellant to LEO.
- There is a clear advantage for an airbreathing launcher when considering sustained space operations.
- Compared to the impulse turn OMV, the hypersonic glider requires less total propellant to accomplish its mission, requiring only about 65% of the impulse turn OMV propellant, see Table 5.15.

In summary, for performing orbital plane changes, hypersonic gliders have clear advantages. Even for the

Table 5.14 Ratio of total propellant weight to satellite weight for FDL-7C/D hypersonic glider and three plane change angles for four launcher propulsion systems

Plane change (degree)	Launcher propulsion			
	Rocket (–)	LACE (–)	M = 10 (–)	M = 12 (–)
90	310.9	135.7	86.7	68.2
62	160.2	70.0	44.7	35.1
32	66.2	28.9	18.5	14.5
32[a]	73.5	32.1	20.5	16.1

[a]Sized for 62° plane change operated over a 32° plane change

Table 5.15 Ratio of total propellant weight to satellite weight for the FDL-7C/D hypersonic glider compared to the hydrogen/oxygen propellant OMV designed for a 32° plane change for four launch propulsion systems

Plane change	Launcher propulsion			
	Rocket (–)	LACE (–)	M = 10 (–)	M = 12 (–)
Hypersonic glider	66.2	28.9	18.5	14.5
H_2/O_2 OMV	101.7	44.4	28.3	22.2

hypersonic glider designed for a 62° plane change, while flying a 32° plane change (last row of Table 5.14) requires less propellant compared to an impulse OMV. The hypersonic gliders require less propellant to be lifted to orbit and offer an escape and rescue capability not available with impulse turn OMVs.

5.5 Representative Space Transfer Vehicles

Each OMV has approximately the same W_{OEW} as indicated in Tables 5.10, 5.11 and 5.12. However, each OMV has a different configuration that is determined by the characteristics of the individual propulsion system, as depicted in Fig. 5.19.

The two chemical rocket-powered OMVs are similar and conventional. Although having different gross weights, they are similarly sized. The satellite attaches to an equipment module mounted on the front end of the propellant tank, where the guidance and control systems and all subsystems are housed, see Fig. 5.20. There would be a stowed communications antenna and solar panels for power in the equipment module (not shown).

The solar electric propulsion system does require much larger solar panels than shown. Current communications satellites have solar panels in the 25–30 m (82–98 ft) total span for thrusters with less than 1/10th the thrust required for the solar electric OMV. Some of the limitations of this system are the current low thrust levels, the continuously degrading solar panel output (aging), and the unwieldy size of the solar panels for such a vehicle. Nuclear electric has the same problem as the solar electric, in that current thrusters have less than 1/10th the thrust required for the nuclear electric OMV. This system does have the advantage that the power output is sufficient and constant. The nuclear electric OMV requires large radiators to dissipate the rejected thermal energy from the reactor to space. Their exact size depends on the nuclear system chosen and the thermodynamic cycle to power the electric generators. The nuclear reactor will be a space-designed reactor and not based on Earth-based nuclear power stations. A most likely candidate is some type of gas-cooled reactor.

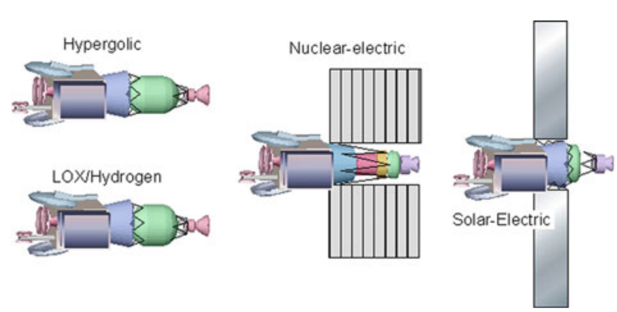


Fig. 5.19 Relative size and general configuration of OMVs

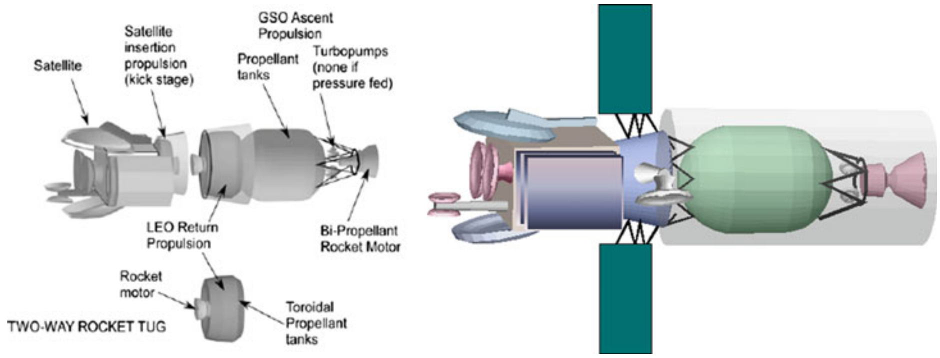


Fig. 5.20 LEO-GSO-LEO two-way OMV with shield

A round-trip operational OMV that travels from LEO to GSO and returns to LEO is shown in Fig. 5.20. The solar panels are just sufficient to power the system electronics and other electrical subsystems. A communication link to Earth and space-based ground stations is indicated. Because the intended life expectancy is multiple years, and recalling the damage one of these authors (P.A. Czysz) witnessed on the LDEF (Long Duration Exposure Facility) satellite, a shield over the tank structure and engine is necessary, as shown in phantom in Fig. 5.20. The equipment module can be made robust enough not to require a separate shield. As with the MIR orbital station, the solar panels on an operational OMV will probably have to be replaced within its lifetime.

The orbital plane change OMV can change the orbital plane by an (a) impulse turn in orbit or an (b) aerodynamic turn in the upper atmosphere. The impulse plane change OMV is very similar to the OMV as shown in Fig. 5.20 and is shown on the left side of Fig. 5.21. The aerodynamic plane change OMV is shown in the right side of Fig. 5.21. Both are sized for a 32° plane change with a 2268 kg (5000 lb) satellite. The impulse plane change OMV cannot enter the Earth's atmosphere, therefore limiting it to space operations. The hypersonic glider OMV has the capability to enter the atmosphere to operate as a rescue vehicle. The glider has a glide range equal to the Earth's circumference and can return to Earth without any prior preparation or waiting in orbit. With a payload bay of 36.5 m^3 (1289 ft^3) capacity, it can accommodate nine to twelve persons in pressure suits in an emergency situation.

5.6 Operational Considerations

Given the characteristics of the OMVs, the question is how to build an operational infrastructure in addition to the OMVs. The next five subsections will attempt to put the needs for an operational infrastructure into perspective. In fact, one of the most critical issues, if not the most critical, is the orbital propellant resources required to sustain an operational infrastructure. The availability of an infrastructure architecture and infrastructure hardware is important, but without propellant all grinds to a standstill. The infrastructure will probably be configured in some type of constellation, distributing resources over the infrastructure shell around the Earth. Since nowadays resources are scarce, the operators of the infrastructure must be a frugal group, not wasting any reusable resource or hardware. This consideration hints at private entrepreneurs rather than to the traditional space agencies as the main players. And, finally, having populated the infrastructure with human beings that

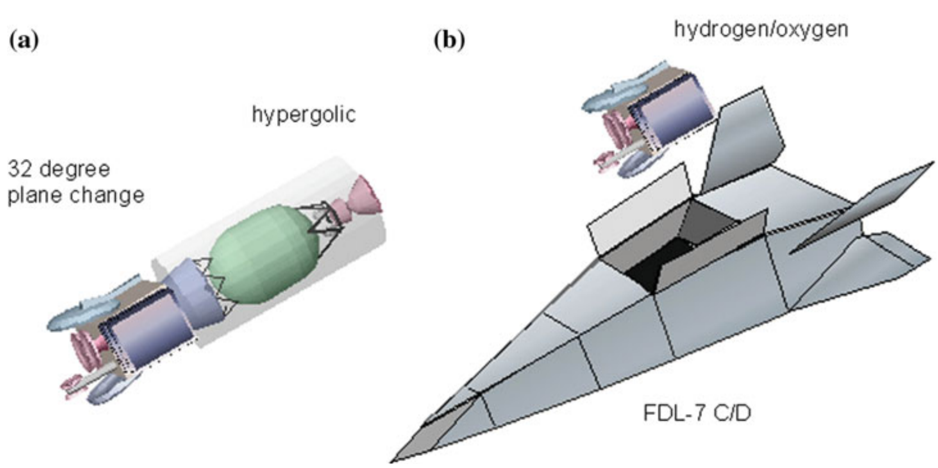


Fig. 5.21 OMV for **a** impulse turn and **b** hypersonic glider for aerodynamic turn

are not pilots, but workers with identified tasks, and tourists hoping to see and experience space, a viable and readily available rescue and return capability is a necessity.

5.6.1 Missions Per Propellant Delivery

It is worth repeating that the critical issue is the orbital propellant resources required to sustain an operational infrastructure. As the results given with the previous subchapters have shown, the existing expendable rocket launcher systems and hypergolic propellant space rockets force a level of launcher performance and activity that make any but limited space operations impractical, and this is witnessed by the current status quo. Figures 5.10 and 5.11 with Figs. 5.17 and 5.18 show that the expendable rocket launcher and hypergolic rocket OMV spend over 200 kg of propellant to deliver 1 kg of OMV propellant to LEO. The solution anticipated is to use airbreathing launchers and nuclear electric-powered OMVs. Then, the requirement reduces to a figure on the order of 2 or 3 to deliver propellant to LEO and on the order of 5 to deliver to LEO propellant required for orbital plane changes. It would appear that the operational infrastructure envisioned by the late Dr. Gaubatz in Fig. 5.1 must wait for the deployment of the correct propulsion systems for both, the synergistic "twins" consisting of the operational space launcher and the operational OMV.

The next critical issue is the following: Given the propellant is delivered to LEO in 19 t (41,895 lb) increments, how many missions can the OMVs complete from a single delivery? Figure 5.22 and Table 5.16 give the number of missions for the impulse OMVs executing two different missions, and the aerodynamic turn mission for the FDL-7 C/D hypersonic glider with a lift-to-drag ratio of 2.7.

Although heavier than the impulse OMV, the efficiency of the aerodynamic plane change maneuver permits the hypersonic glider OMV to have 45% greater mission capability from the same orbital tanker propellant load. Solar electric and nuclear electric are not appropriate propulsion systems for vehicles that fly in the upper atmosphere, because of the solar panels and radiators associated with those systems.

5.6.2 Orbital Structures

The concept of space ways depicted in Fig. 5.1 is dependent on the capability to manufacture space structures as standard items on a limited production line, comparable to the aircraft assembly line. Although the USA, Japan, and Europe have manufactured individual modules for the International Space Station (ISS) over its more than 10-year construction time (and a similar length characterize the Chinese *Tiangong-1*), these are one-of-a-kind items, hand-built at great expense. The only nation known to manufacture space structures with standardized components on a limited production line is the former Soviet Union.

Figure 5.23 shows one picture of one of a number of orbital station major components being manufactured in a factory in the Moscow area. In this picture, the orbital station module is being integrated with its *Proton* launcher, at the manufacturing plant for immediate detection of interface problems that can be addressed during the manufacturing process, not later on the launch stand. Each of the modules and components has different functions, but, such as automobiles and aircraft, each has been tailored to a specific mission based on installed equipment and a common structural core. The costs and time to manufacture the components have been minimized using this approach.

The organization of the manufacturing line, and the use of standardized components that can be gleaned from the plant pictures, is quite impressive. The pictures of this plant are now more than 35 years old. It is not known whether the

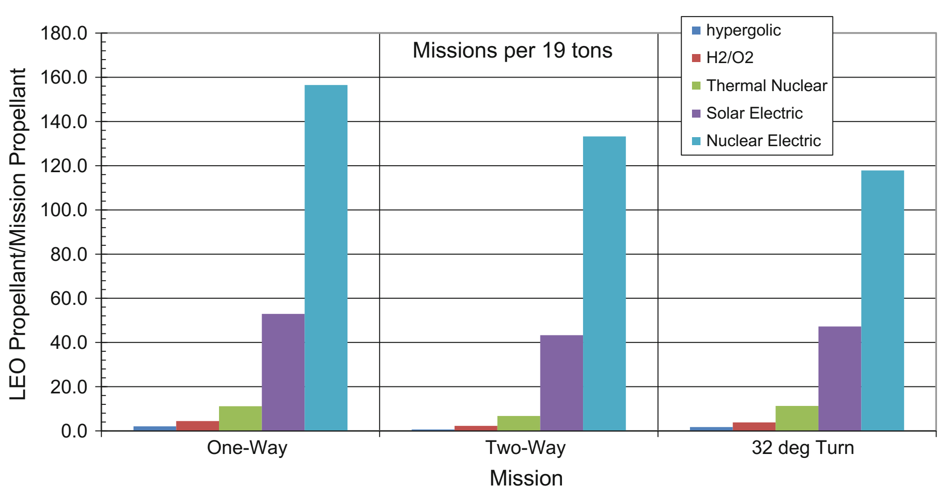


Fig. 5.22 Orbital maneuver missions per 19 t propellant payload for five different OMV propulsion systems

Table 5.16 Number of orbital missions per 19 t (metric ton) propellant payload for 2268 kg satellite payload for the OMV

Mission	Launcher propulsion				
	Hypergolic (–)	H_2/O_2 (–)	Solar electric (–)	Thermal nuclear (–)	Nuclear electric (–)
Impulse OMV LEO to GSO and return	0.71	2.3	4.3	6.8	133
Impulse OMV 32° plane change	1.7	3.8	4.7	11	118
Hypersonic glider 32° plane change		5.5			

Fig. 5.23 Large orbital station in final assembly and integration with its Proton booster. Moscow factory, circa 1989

plant or manufacturing capability remains in present Russia. This is the only plant of its kind known to the authors, and it should be the model for manufacturing components for an operational space infrastructure instead of relying on building single, one-of-a-kind custom components. One of the very important observations of the Russian approach to space payloads is that the payload and delivery stage are integrated as a part of the manufacturing process and not left to cause future delays on the launch pad. Note the Proton booster on the right-hand side of the photograph. In this context, a technology revolution is the advent of additive manufacturing, facilitated by low- or micro-gravity and vacuum. This last makes plasma torches easier and more convenient to use.

5.6.3 Orbital Constellations

One of the Senior Capstone Design course project teams at Saint Louis University, USA, looked at the near-Earth infrastructure postulated by the late Dr. William Gaubatz. The project topic was to analyze what would constitute the first step in the development of that infrastructure. The title of the project was *Space-Based Satellite Service Infrastructure* (Shekleton et al. 2002). Among other results found was that as the number of structures in space continually increases, the need for a space-based service infrastructure continues to grow.

The increasing human presence in space calls for creative support and rescue capabilities that will make space an "easier" and safer frontier. In addition, over 2200 functioning unmanned satellites are currently populating Earth's orbits. These include a variety of commercial, military, weather, and research satellites, many of which require servicing or ultimately removal from orbit as space debris at some point in time. As a first step, the student team determined that significant space facilities are necessary to achieve support of an initial "catalyst" infrastructure. As shown in Fig. 5.24, there is a requirement for distributed facilities (Shekleton et al. 2002).

The primary facility identified is a twin propellant tank arrangement with living quarters, repair shop, and a parts storage straddling the two propellant tanks. A much larger and modified version of the "elliptical" Space Cruiser is shown in Fig. 5.25; this vehicle has been identified as the

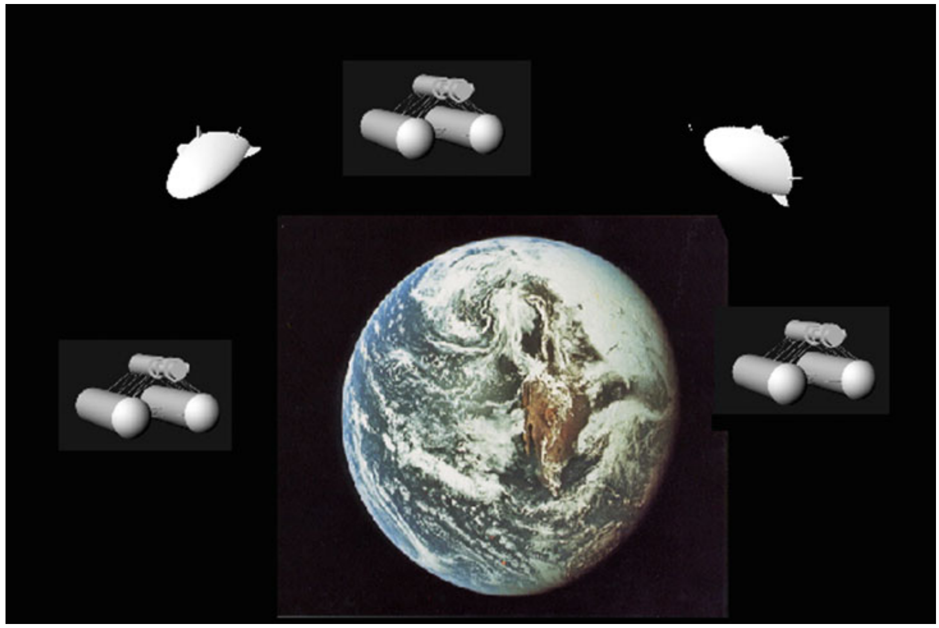


Fig. 5.24 Student design team results in terms of orbital systems hardware

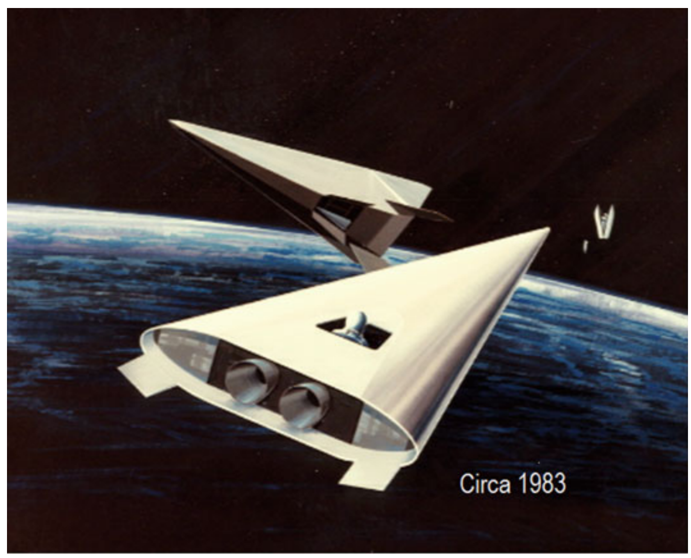


Fig. 5.25 "Bud" Redding "elliptical" Space Cruiser launched from a transatmospheric vehicle to accomplish a satellite repair

primary OMV. The elliptical cross-sectional hypersonic glider has been modified to a captured shock cross section (waverider) based on the work of Mark J. Lewis when at the University of Maryland (Lewis 1993), see Fig. 5.24. The initial concept of the "waverider," first implemented with North American's supersonic B-70 Valkyrie strategic bomber, can be attributed to the developments by (Eggers and Syvertson 1956).

The OMVs are deployed from the service facilities on an as-needed basis for non-routine maintenance and repair and on a scheduled basis for operational satellites and facilities. The gliders have limited facilities as habitats but have sufficient provisions for 3- to 5-day deployments away from the main service facility. The space station has not been chosen as a support base, because of the large quantity of propellant stored and the large inventory of spare parts and repair facilities required. One of the service facilities could be in orbital proximity to the space station if that is operationally required. The propellant storage could accommodate about 100 t of propellant or up to five propellant tanker payloads. The propellant tanks are segregated to accommodate hypergolic and hydrogen/oxygen propellants separately. Cryogenic propellants will need cryo-coolers and much energy to operate them (cooling needs about 20–100 times the amount of energy to be extracted). The hypersonic gliders are capable of escape and rescue missions for up to 15 persons. This constellation has been considered the foundation on which to build an operational space infrastructure.

5.6.4 Docking with Space Facilities and the ISS

Examining Fig. 5.1, we see a variety of space structures (facilities) that are unique to each facility's function, in time an aspect that is probably the norm for space facilities. In reality, we are just beginning because there is no existing space infrastructure with the exception of the ISS and, on a limited basis, the Chinese Tiangong-1 station. At best, there are specific missions to specific orbital assets (such as to Hubble, before the retirement of the US Space Shuttle). As published in the aerospace literature, the European (Columbus Laboratory) and Japanese (Japanese Experimental Module, Kimbo) laboratory modules for the International Space Station needed over 5 years to complete at large financial cost (Baker 2012). The high-maintenance and consequently high-cost Space Shuttle retired in July 2011, shortly after the completion of the ISS assembly.

Existing orbital facilities are expensive, and visiting vehicles must conform to standards and requirements based on vehicle and facility idiosyncrasies. For now, there is no consistent set of standards and requirements in sync with the commercial industries. Eventually, the transportation vehicles will have to provide the requirements for the orbital baseline infrastructure, including the definition of transportation cycles in analogy to how airports are defining air transportation. Commercial platform markets include transportation-related support services, habitation, and in-space service industry support.

The most economical space facility ever flown was the US Skylab (Anon 2012). It was a Saturn S-IVB stage modified for habitation and launched empty. Instead of being the prototype of future space structures for the initial phase of infrastructure building, it was summarily and unwittingly permitted to decay from orbit and burn up in the atmosphere. Skylab was placed into a 435-km (235 nm) orbit at an inclination of 50° (Furniss 2001). Skylab was in orbit from May 14, 1973, to July 11, 1979, (6 years, 5 months, and 25 days). It was launched empty and was sent crews via a Saturn rocket and an Apollo capsule. There were three missions to crew Skylab: Skylab 2 for 28 days, Skylab 3 for 59 days, and the final Skylab 4 for 84 days, for a total of 171 days occupied. The last crew departed Skylab on February 8, 1974, just 8 months and 26 days after being put into orbit. Clearly, Skylab remained unused for over 5 years. Unfortunately, there was no mechanism to maintain Skylab in orbit, and on July 11, 1979, it entered the atmosphere over Australia. Again, instead of being a prototype for an economical first step toward an orbital station, it was a one-of-a-kind experiment only. The next philosophical path taken was then to create an "optimum" space station, the "perfect" creation of NASA that took almost 26 years before another American astronaut crewed a US orbital station. In that time period, the former Soviet Union placed seven orbital stations into orbit, ending with the orbital station MIR. Note that since the retirement of the Space Shuttle in 2011, the USA is still devoid of a US man-rated space launch system to LEO.

There exists an analogous situation involving the STS (Space Transportation System, or Space Shuttle) which retired in 2011. The Space Shuttle external tank is a giant cylinder 154 ft (46.7 m) in length and 27.5 ft (8.4 m) in diameter containing 73,600 ft^3 (2083 m^3) of propellants. That is about 369,600 lb (167.63 t) at a 6:1 oxygen/hydrogen ratio by mass. The lithium–aluminum external tank weighs 58,250 lb dry. Each Space Shuttle mission discarded the external tank after it had achieved 99% of full orbital velocity. This means significant energy had been invested in the external tank, only about 260 ft/s (79 m/s) short of orbital velocity. With a very small investment, the external tank (ET) could have been placed into orbit and become the initial building block for orbital facilities other than the International Space Station, at a fraction of the cost.

At one time, the government was encouraging organizations to put this empty space asset to a useful application (Commerce Business Daily 1988). One of the individuals taking this seriously was Thomas Taylor, CEO of Global Outpost. He and his company championed the salvage of the external tank for over two decades (Taylor 1980, 1998;

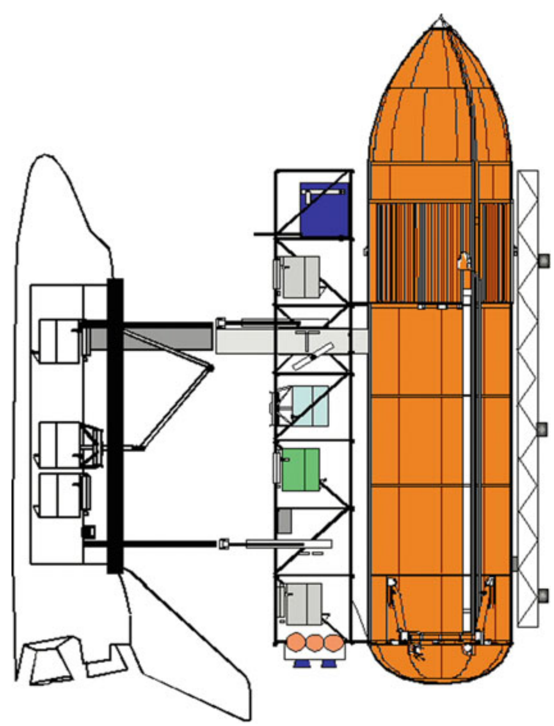


Fig. 5.26 An orbital infrastructure station fabricated from discarded Space Shuttle main propellant tanks with a Space Shuttle docked for resupply

Gimarc 1985). Global Outpost developed a salvage method using the Space Shuttle with NASA assistance. The organization had won the right to "five ET's in orbit at no cost" and had worked out a salvage procedure with NASA (Anon 1993). The concepts shown in Figs. 5.26, 5.27 and 5.28 are based on concepts developed by Thomas Taylor and Global Outpost Inc.

At the time, there were several possibilities for the empty external tank:

(1) The external tank could be used as it was intended to be used, as a hydrogen/oxygen propellant storage facility, using the orbital refueling launchers to supply propellants on a scheduled basis. The tank could accommodate 8.8 of the 19 t propellant deliveries by the orbital propellant tanker.
(2) The aft dome of the external tank could be cut to provide a 10.3 ft (3.14 m) diameter hole permitting the use of 55,000 ft^3 (1557 m^3) of the interior as a hangar for the OMVs.
(3) Just as with the Saturn S-IVB stage, the external tank could be launched, with some modifications so that at least one external tank could accommodate a human habitat. This modification is the basis for the sketches in Figs. 5.26, 5.27, and 5.28.
(4) An inflatable habitation structure is possible using the NASA TransHab or Bigelow Aerospace 8.0 m (26.25 ft) diameter and 8.2 m (26.90 ft) long inflatable structure (Kennedy et al. 2000). A fabricated volume would be transported uninflated in a sustained-use space launcher described in Chap. 3 and inflated on orbit for sustain use. The habitat is capable of resisting high-speed particle impact and providing environmental controlled life-support interior. A first prototype was launched to the ISS on April 16, 2016, was successfully connected and inflated, and is being evaluated throughout 2016 and into 2018, when a decision by NASA on its viability will be made.

Habitation requires cargo and passenger services. Each new industry will require cargo in both directions. The change from one type of transportation to another has always evolved into major commercial centers of industry, such as harbors and airports. Emerging commercial space ways have to expand the capabilities around the Earth and then to the Moon. Transportation is and will remain *the* major catalyst. The cost reduction stimulates the accelerated growth and expansion. Harbors start small, grow, and reach out to their customers with docks and wharfs; the space harbor will be no exception.

The external tank modified for crewed habitation and equipment and parts storage facility as conceived by Taylor (1980) is shown with the NASA Space Shuttle Orbiter docked with the crew transfer structure deployed between the orbiter air lock module and the external tank, see Fig. 5.26. This mission would have been for an equipment/parts resupply mission, for crew rotation, or as a mission adjunct. However, since the Shuttle had a limited useful operational life, its retirement opened the pathway toward a sustained flight rate spacecraft. The one actually designed for that purpose (for the USAF MOL in 1964) was the Air Force Flight Dynamics Laboratory *FDL-7C/D* and the McDonnell Douglas derivative, the *Model 176*. The modified external tank shown in Fig. 5.27 is shown docked with the crew transfer structure deployed between the *FDL-7C/D* or MDC *Model 176* air lock module and the external tank. As before, this could have been an equipment/parts resupply mission, crew rotation, or as a mission adjunct.

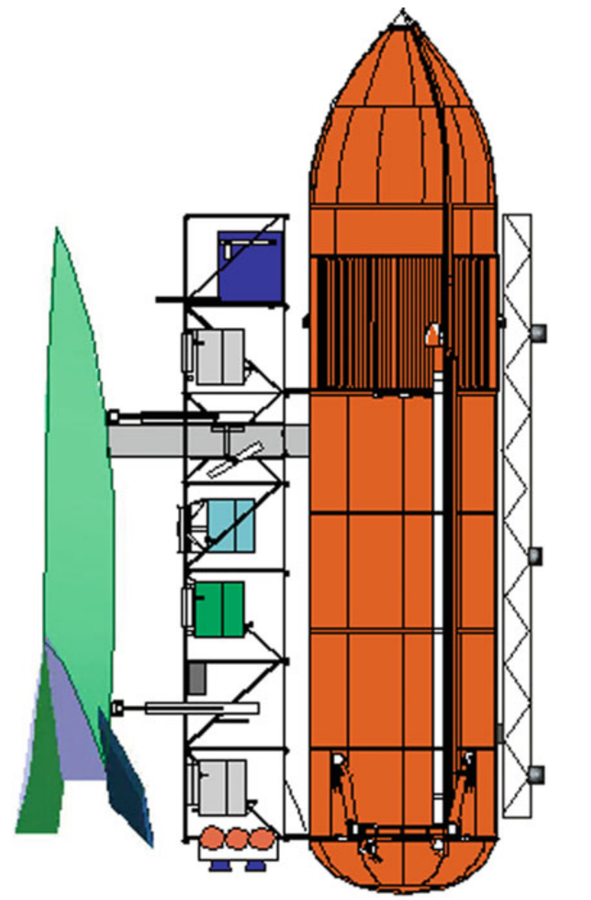


Fig. 5.27 An orbital infrastructure station fabricated from discarded Space Shuttle main propellant tanks with a hypersonic glider resupply spacecraft analogous to MDC *Model 176*

The concept of the Space Cruiser has been introduced in Chap. 2, see Fig. 2.26. This vehicle enables the external tank (ET) to take on the role of a maintenance, repair, and orbital transfer center, much as that developed by the Parks College design team (Shekleton et al. 2002). The Space Cruiser dates back over 40 years. The authors first were aware of the concept when one of the authors (P.A. Czysz) was manager of the McDonnell Douglas Aerospace Vehicle

Group in 1983. The late Mr. Redding visited the author and briefed him on the Space Cruiser concept. As originally conceived in 1980, the Space Cruiser is a low-angle conically shaped hypersonic glider similar to the McDonnell Douglas Model 122 (BGRV) experimental hypersonic vehicle that was flown in 1966 (Hallion 2005).

As initially conceived, the Space Cruiser length is 26 ft but can be folded to a 13.5 ft length, see Figs. 5.25 and 5.28. Redding adapted the design to incorporate an aft plug nozzle cluster configuration and storable propellants to create 13.3 kN (3000 lb) of thrust. The 4453 kg (10,000 lb) vehicle is to perform a variety of missions using the 8 ft^3 forward payload bay and the 4 ft^3 aft payload bay. The Space Cruiser is capable of atmospheric entry and uses a small drogue parachute at Mach 1 followed by a multi-reefed parafoil to land safely on any flat surface. The Space Cruiser has been intended to be operated by a pilot in a space suit (Griswold et al. 1982). In 1983, Redding modified the configuration to an elliptical cross section aimed at expanding the propellant quantity, as shown in a 1983 McDonnell Douglas Corporation transatmospheric vehicle (TAV) artist illustration, see Fig. 2.25 (Redding et al. 1983; Redding 1984). Mr. Redding formed an organization shortly before his death to preserve the work on the Space Cruiser and seek future development, the In-Space Operations Corporation (IOC).

In Fig. 5.28, the external tank modified for crewed habitation and an equipment and parts storage facility is shown with several space maneuvering vehicles docked to the support structure. From the top-right, there is a round-trip to GSO rocket transfer vehicle, see also Fig. 5.20; the center-right shows a solar electric orbital transfer vehicle, see also Fig. 5.19. At the bottom-right, there is a folded Space Cruiser with a satellite for transfer to another facility. At top-left, there is a hypersonic glider aerodynamic plane change vehicle, and at bottom-left, a full length Space Cruiser is shown docked. The Space Cruisers shown in Fig. 5.28 are 2.4 times larger than the original Space Cruiser (62 ft or 18.9 m in length). They have 13.5 times more volume and greater capability because the propellants are now cryogenic hydrogen and oxygen with magnetic refrigerators to all to eliminate propellant losses. These, like all the orbital maneuver vehicles, are automatic control vehicles that can carry crewmembers when necessary. In this figure, the salvaged external tank was thought to be an operations center for orbital maneuver vehicles necessary to move satellites and provide on-site repair and maintenance and non-functioning satellite removal.

With the Shuttle demise, all these potential developments ceased and the 2008 financial crisis reduced the forecasts of commercial use of space. As of this writing, conventional rockets are the only form of launchers, although emphasis is on cost reduction by automatic landing and reuse of first stages fueled by kerosene or liquid methane. Blue Origin and Space-X are two companies involved in these developments.

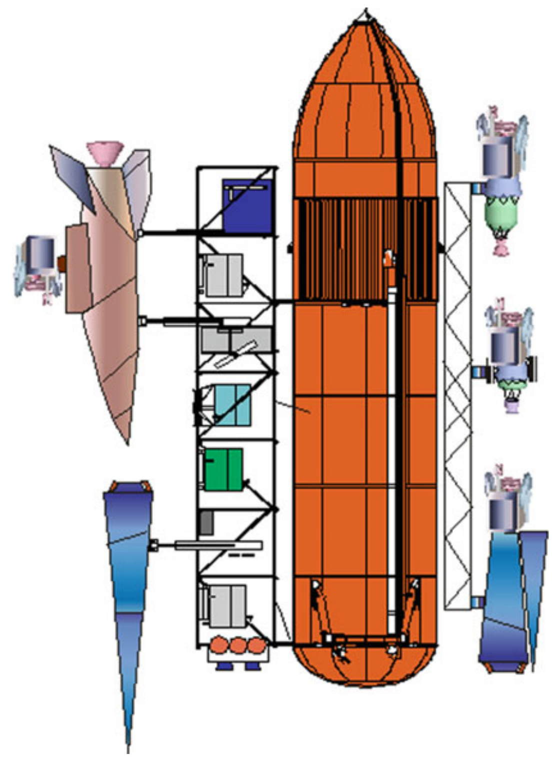


Fig. 5.28 An orbital infrastructure station fabricated from discarded Shuttle main propellant tanks with docked In-Space Operations Corporation (IOC) Space Cruiser, a hypersonic orbital plane change vehicle, and OMVs

5.6.5 Emergency Rescue Vehicle

Whether it is the orbital facility support vehicle, the hypersonic glider aerodynamic plane change vehicle, or the Space Cruiser, these vehicles can serve as an immediately available escape and rescue vehicle in case of an emergency. With these vehicles recovering in the Continental United States (CONUS), or Continental Europe (CONEU), they are capable to reenter to reach these locations without waiting in orbit for the correct orbital position due to their superior inherent extended crossrange and downrange hypersonic glide performance.

The orbital facility support vehicle has the capability to accommodate nine to thirteen crew, depending on the medical circumstance (litter patients or ambulatory). This means that with a fleet of these vehicles, the space facilities need not be only partially manned; the high-performance return vehicles provide safe return for the fully crewed facilities. These vehicles have been designed in the past to be able to generate 75–90 flights a year and to be launched in less than 24 h. This implementation scheme presented provides a true capability to build an operational infrastructure as envisioned by the late Dr. William Gaubatz in Fig. 5.1.

5.7 Observations and Recommendations

This chapter has demonstrated the very large resources required to support the delivery of propellant for an operational infrastructure if conventional rocket launchers are used with conventional hypergolic rockets for space operations. It is required that sustained-use airbreathing launchers and nuclear space propulsion be developed into an operational system *if* an operational space infrastructure is ever to exist.

The key to achieving an initial operating capability with an infrastructure is not to throw away valuable (reusable) assets in lieu of very costly and long delivery-time optimum solutions that have little tolerance or durability when encountering off-design conditions and unexpected events. Some of the missed opportunities include usage of the salvaged Space Shuttle main external tank (ET) that could have been put to use as identified by Thomas Taylor. The authors observe the following:

(1) The emerging partly reusable launch vehicles by Space-X and Blue Origin in the USA, and those being designed by Airbus in EU, will probably bring more cost-effective transportation and commercial ventures to LEO.
(2) Salvaged hardware in orbit will provide commercial opportunities and transportation markets in LEO.
(3) Human-operated commercial services in orbit will emerge as the lower-cost transportation options emerge.
(4) The transportation node in LEO is important to the commercial world, because the mode of transportation changes in LEO.
(5) The cost for countries interested in positioning on the trade routes of the future is lower than ever and will be commercial.
(6) A new method of cooperation between government and the private sector must be found.
(7) In the mid- to far-term, access to asteroids rich in ice ("dark comets") may provide water and oxygen for crew support and propellants for conventional and nuclear rockets, see Sect. 7.6.
(8) Space Tourism will serve as a stepping-stone or *primer* supporting (1) through (7).

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