

Spacecraft Components



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Table of Contents

Chapter 1 - Astrionics

Chapter 2 - Avionics

Chapter 3 - Boilerplate (spaceflight)

Chapter 4 - Electrodynamic Tether

Chapter 5 - Gemini Space Suit

Chapter 6 - Heat Pipe

Chapter 7 - Magnetic Sail

Chapter 8 - Rocket Engine

Chapter 9 - Solar Sail

Chapter 10 - Space Shuttle External Tank

Chapter 11 - Spacecraft Magnetometer

Chapter 12 - Rocket Propellant

Chapter-1

Astrionics

Astrionics is the science and technology of the development and application of electronic systems, sub-systems, and components used in spacecraft. The electronic systems on board a spacecraft include attitude determination and control, communications, command and telemetry, and computer systems. Sensors refers to the electronic components on board a spacecraft.

For engineers one of the most important considerations that must be made in the design process is the environment in which the spacecraft systems and components must operate and endure. The challenges of designing systems and components for the space environment include more than the fact that space is a vacuum.

Attitude Determination and Control

Overview

One of the most vital roles electronics and sensors play in a mission and performance of a spacecraft is to determine and control its attitude, or how it is orientated in space. The orientation of a spacecraft varies depending on the mission. The spacecraft may need to be stationary and always pointed at Earth, which is the case for a weather or communications satellite. However, there may also be the need to fix the spacecraft about an axis and the have it spin. The attitude determination and control system, ACS, ensures the spacecraft is behaving correctly. Below are several ways in which ACS can obtain the necessary measurements to determine this.

Magnetometer

This device measures the strength of the Earth's magnetic field in one direction. For measurements on 3-axis the device would consist of three orthogonal magnetometers. Given the spacecraft's position, the magnetic field measurements can be compared to a known magnetic field which is given by the International Geomagnetic Reference Field model. Measurements made by magnetometers are affected by noise consisting of alignment error, scale factor errors, and spacecraft electrical activity. For near Earth

orbits, the error in the modeled field direction may vary from 0.5 degrees near the Equator to 3 degrees near the magnetic poles, where erratic auroral currents play a large role. The limitation of such a device is that in orbits far from Earth, the magnetic field is too weak and is actually dominated by the interplanetary field which is complicated and unpredictable.

Sun Sensors

This device works on the light entering a thin slit on top of a rectangular chamber that casts an image of a thin line on the bottom of the chamber, which is lined with a network of light-sensitive cells. These cells measure the distance of the image from a centerline and using the height of the chamber can determine the angle of refraction. The cells operate based on the photoelectric effect. Incoming photons are converted to electrons and therefore voltages, which are in turn converted into a digital signal. By placing two sensors perpendicular to each other the complete direction of the sun with respect to the sensor axes can be measured.

Digital Solar Aspect Detectors

Also known as DSADs, these devices are purely digital Sun sensors. They determine the angles of the Sun by determining which of the light-sensitive cells in the sensor is the most strongly illuminated. By knowing the intensity of light striking neighboring pixels, the direction of the centroid of the sun can be calculated to within a few arcseconds.

Earth Horizon Sensor

Static

Static Earth horizon sensors contain a number of sensors and sense infrared radiation from the Earth's surface with a field of view slightly larger than the Earth. The accuracy of determining the geocenter is 0.1 degrees in near-Earth orbit to 0.01 degrees at GEO. Their use is generally restricted to spacecraft with a circular orbit.

Scanning

Scanning Earth horizon sensors use a spinning mirror or prism and focus a narrow beam of light onto a sensing element usually called a bolometer. The spinning causes the device to sweep out the area of a cone and electronics inside the sensor detect when the infrared signal from Earth is first received and then lost. The time between is used to determine Earth's width. From this the roll angle can be determined. A factor that plays into the accuracy of such sensors is the fact the Earth is not perfectly circular. Another is that the sensor does not detect land or ocean, but infrared in the atmosphere which can reach certain intensities due to the season and latitude.

GPS

This sensor is simple in that using one signal many characteristics can be determined. A signal carries satellite identification, position, the duration of the propagated signal and clock information. Using a constellation of 36 GPS satellites, of which only four are needed, navigation, positioning, precise time, orbit, and attitude can be determined. One advantage of GPS is all orbits from Low Earth orbit to Geosynchronous orbit can use GPS for ACS.

Command and Telemetry

Overview

Another system which is vital to a spacecraft is the command and telemetry system, so much in fact, that it is the first system to be redundant. The communication from the ground to the spacecraft is the responsibility of the command system. The telemetry system handles communications from the spacecraft to the ground. Signals from ground stations are sent to command the spacecraft what to do, while telemetry reports back on the status of those commands including spacecraft vitals and mission specific data.

Command Systems

The purpose of a command system is to give the spacecraft a set of instructions to perform. Commands for a spacecraft are executed based on priority. Some commands require immediate execution; other may specify particular delay times that must elapse prior to their execution, an absolute time at which the command must be executed, or an event or combination of events that must occur before the command is executed. Spacecraft perform a range of functions based on the command they receive. These include: power to be applied to or removed from a spacecraft subsystem or experiment, alter operating modes of the subsystem, and control various functions of the spacecraft guidance and ACS. Commands also control booms, antennas, solar cell arrays, and protective covers. A command system may also be used to upload entire programs into the RAM of programmable, micro-processor based, onboard subsystems.

The radio-frequency signal that is transmitted from the ground is received by the command receiver and is amplified and demodulated. Amplification is necessary because the signal is very weak after traveling the long distance. Next in the command system is the command decoder. This device examines the subcarrier signal and detects the command message that it is carrying. The output for the decoder is normally non-return-to-zero data. The command decoder also provides clock information to the command logic and this tells the command logic when a bit is valid on the serial data line. The command bit stream that is sent to the command processor has a unique feature for spacecraft. Among the different types of bits sent, the first are spacecraft address bits. These carry a specific identification code for a particular spacecraft and prevent the intended command from being performed by another spacecraft. This is necessary due to the fact there are many satellites using the same frequency and modulation type.

The microprocessor receives inputs from the command decoder, operates on these inputs in accordance with a program that is stored in ROM or RAM, and then outputs the results to the interface circuitry. Because there is such a wide variety of command types and messages, most command systems are implemented using programmable microprocessors. The type of interface circuitry needed is based on the command sent by the processor. These commands include relay, pulse, level, and data commands. Relay commands activate the coils of electromagnetic relays in the central power switching unit. Pulse commands are short pulses of voltage or current that is sent by the command logic to the appropriate subsystem. A level command is exactly like a logic pulse command except that a logic level is delivered instead of a logic pulse. Data commands transfer data words to the destination subsystem.

Telemetry Systems

Commands to a spacecraft would be useless if ground control did not know what the spacecraft was doing. Types of information telemetry includes is: The status data concerning spacecraft resources, health, attitude and mode of operation. Scientific data gathered by onboard sensors: telescopes, spectrometers, magnetometers, accelerometers, electrometers, thermometers, etc.. Specific spacecraft orbit and timing data that may be used for guidance and navigation by ground, sea, or air vehicles. Images captured by onboard cameras (visible or infrared). Locations of other objects, either on the Earth or in space, that are being tracked by the spacecraft. Telemetry data that have been relayed from the ground or from another spacecraft in a satellite constellation. The telemetry system is responsible for acquisition from the sensors, conditioners, selectors, and/or converters, processing, including compression, format, and storage, and finally transmission, which includes encoding, modulating, transmitting and the antenna.

There are several unique features of the telemetry system design for spacecraft. One of these is the approach to the fact that for any given satellite in LEO, because it is traveling so fast, it may only be in contact with a particular station for ten to twenty minutes. This would require hundreds of ground stations to stay in constant communication which is not at all practical. One solution to this is onboard data storage. Data storage can accumulate data slowly throughout the orbit and dump it quickly when over a ground station. In deep space missions the recorder is often used the opposite way, to capture high-rate data and play it back slowly over data-rate-limited links. Another solution is data relay satellites. NASA has satellites in GEO called TDRS, Tracking and Data Relay Satellites, which relay commands and telemetry from LEO satellites. Prior to TDRS, astronauts could communicate with the Earth for only about 15% of the orbit, using 14 NASA ground stations around the world. With TDRS, coverage of low-altitude satellites is global, from a single ground station at White Sands, New Mexico.

Another unique feature of telemetry systems is autonomy. Spacecraft require the ability to monitor their internal functions and act on information without ground control interaction. The need for autonomy originates from problems such as insufficient ground coverage, communication geometry, too near the Earth-Sun line, where solar noise interferes with radio frequencies, or simply for security purposes. The reason autonomy is

important when talking about command and telemetry systems is that the telemetry system already has the capability to monitor the spacecraft functions and the command systems has the ability to give the necessary commands to reconfigure based on the needs of the action to be taken. There are three steps to this process:

1. The telemetry system must be able to recognize when one of the functions it's monitoring deviates beyond the normal ranges.
2. The command system must know how to interpret abnormal functions, so that it can generate a proper command response.
3. The command and telemetry systems must be capable of communicating with each other.

Sensors

Sensors can be classified under two categories: health sensors and payload sensors. Health sensors monitor the spacecraft or payload functionality and can include temperature sensors, strain gauges, gyros and accelerometers. Payload sensors would include radar imaging systems and IR cameras. While payload sensors represent some of the reason the mission exists, it is the health sensors that measure and control systems to ensure optimum operation.

Chapter-2

Avionics

Avionics is a portmanteau of "aviation" and "electronics". It comprises electronic systems for use on aircraft, artificial satellites and spacecraft, comprising communications, navigation and the display and management of multiple systems. It also includes the hundreds of systems that are fitted to aircraft to meet individual roles, these can be as simple as a search light for a police helicopter or as complicated as the tactical system for an Airborne Early Warning platform.

History

The term avionics is believed to have been coined by journalist Philip J. Klass. Avionics was pioneered in the 1970s, driven by military need rather than civil airliner development. Military aircraft had become flying sensor platforms, and making large amounts of electronic equipment work together had become the new challenge. Today, avionics as used in military aircraft almost always forms the biggest part of any development budget. Aircraft like the F-15E and the now retired F-14 have roughly 80 percent of their budget spent on avionics. Most modern helicopters now have budget splits of 60/40 in favour of avionics.

The civilian market has also seen a growth in cost of avionics. Flight control systems (fly-by-wire) and new navigation needs brought on by tighter airspaces, have pushed up development costs. The major change has been the recent boom in consumer flying. As more people begin to use planes as their primary method of transportation, more elaborate methods of controlling aircraft safely in these high restrictive airspaces have been invented.

Main categories

Aircraft avionics

The cockpit of an aircraft is a major location for avionic equipment, including control, monitoring, communication, navigation, weather, and anti-collision systems. The majority of aircraft drive their avionics using 14 or 28 volt DC electrical systems;

however, large, more sophisticated aircraft (such as airliners or military combat aircraft) have AC systems operating at 400 Hz, & 115 volt rather than the more common 50 and 60 Hz of North American home electrical devices. There are several major vendors of flight avionics, including Honeywell (which now owns Bendix/King, Baker Electronics, Allied Signal, etc..), Rockwell Collins, Thales Group, Garmin, Avidyne Corporation, and Narco Avionics.

Communications

Communications connect the flight deck to the ground, and the flight deck to the passengers. On board communications are provided by public address systems and aircraft intercoms.

The VHF aviation communication system works on the Airband of 118.000 MHz to 136.975 MHz. Each channel is spaced from the adjacent by 8.33 kHz. And VHF is also used for line of sight communication as, aircraft to aircraft, aircraft to atc for short distances. There are three VHF sys. Amplitude Modulation (AM) is used. The conversation is performed by simplex mode. Aircraft communication can also take place using HF (especially for trans-oceanic flights) or satellite communication.

Navigation

Navigation is the determination of position and direction on or above the surface of the Earth. Avionics can use satellite-based systems (such as GPS and WAAS), ground-based systems (such as VOR or LORAN), or any combination thereof. Older avionics required a pilot or navigator to plot the intersection of signals on a paper map to determine an aircraft's location; modern systems calculate the position automatically and display it to the flight crew on moving map displays.

Monitoring

Glass cockpits started to come into being with the Gulfstream G-IV private jet in 1985. Display systems display sensor data that allows the aircraft to fly safely. Much information that used to be displayed using mechanical gauges appears on electronic displays in newer aircraft.

Aircraft flight control systems

Airplanes and helicopters have means of automatically controlling flight. They reduce pilot workload at important times (like during landing, or in hover), and they make these actions safer by 'removing' pilot error. The first simple auto-pilots were used to control heading and altitude and had limited authority on things like thrust and flight control surfaces. In helicopters, auto stabilization was used in a similar way. The old systems were electromechanical in nature until very recently.

The advent of fly by wire and electro actuated flight surfaces (rather than the traditional hydraulic) has increased safety. As with displays and instruments, critical devices which were electro-mechanical had a finite life. With safety critical systems, the software is very strictly tested.

Collision-avoidance systems

To supplement air traffic control, most large transport aircraft and many smaller ones use a TCAS (Traffic Alert and Collision Avoidance System), which can detect the location of nearby aircraft, and provide instructions for avoiding a midair collision. Smaller aircraft may use simpler traffic alerting systems such as TPAS, which are passive (they do not actively interrogate the transponders of other aircraft) and do not provide advisories for conflict resolution.

To help avoid collision with terrain, (CFIT) aircraft use systems such as ground-proximity warning systems (GPWS), radar altimeter being the key element in GPWS. One of the major weaknesses of (GPWS) is the lack of "look-ahead" information as it only provides altitude above terrain "look-down". In order to overcome such weakness, modern aircraft use the Terrain Awareness Warning System (TAWS).

Weather systems

Weather systems such as weather radar (typically Arinc 708 on commercial aircraft) and lightning detectors are important for aircraft flying at night or in Instrument meteorological conditions, where it is not possible for pilots to see the weather ahead. Heavy precipitation (as sensed by radar) or severe turbulence (as sensed by lightning activity) are both indications of strong convective activity and severe turbulence, and weather systems allow pilots to deviate around these areas.

Lightning detectors like the Stormscope or Strikefinder have become inexpensive enough that they are practical for light aircraft. In addition to radar and lightning detection, observations and extended radar pictures (such as NEXRAD) are now available through satellite data connections, allowing pilots to see weather conditions far beyond the range of their own in-flight systems. Modern displays allow weather information to be integrated with moving maps, terrain, traffic, etc. onto a single screen, greatly simplifying navigation.

Aircraft management Systems

There has been a progression towards centralized control of the multiple complex systems fitted to aircraft, including engine monitoring and management. Health and Usage Monitoring Systems (HUMS) are integrated with aircraft management computers to allow maintainers early warnings of parts that will need replacement.

The Integrated Modular Avionics concept proposes an integrated architecture with application software portable across an assembly of common hardware modules. It has been used in Fourth generation jet fighters and the latests generation of Airliners.

Mission or tactical avionics

Military aircraft have been designed either to deliver a weapon or to be the eyes and ears of other weapon systems. The vast array of sensors available to the military is used for whatever tactical means required. As with aircraft management, the bigger sensor platforms (like the E-3D, JSTARS, ASTOR, Nimrod MRA4, Merlin HM Mk 1) have mission management computers.

Police and EMS aircraft also carry sophisticated tactical sensors.

Military communications

While aircraft communications provide the backbone for safe flight, the tactical systems are designed to withstand the rigours of the battle field. UHF, VHF Tactical (30-88 MHz) and SatCom systems combined with ECCM methods, and cryptography secure the communications. Data links like Link 11, 16, 22 and BOWMAN, JTRS and even TETRA provide the means of transmitting data (such as images, targeting information etc.).

Radar

Airborne radar was one of the first tactical sensors. The benefit of altitude providing range has meant a significant focus on airborne radar technologies. Radars include Airborne Early Warning (AEW), Anti-Submarine Warfare (ASW), and even Weather radar (Arinc 708) and ground tracking/proximity radar.

The military uses radar in fast jets to help pilots fly at low levels. While the civil market has had weather radar for a while, there are strict rules about using it to navigate the aircraft.

Sonar

Dipping sonar fitted to a range of military helicopters allows the helicopter to protect shipping assets from submarines or surface threats. Maritime support aircraft can drop active and passive sonar devices (Sonobuoys) and these are also used to determine the location of hostile submarines.

Electro-Optics

Electro-optic systems include Forward Looking Infrared (FLIR), and Passive Infrared Devices (PIDS). These are all used to provide imagery to crews. This imagery is used for everything from Search and Rescue through to acquiring better resolution on a target.

ESM/DAS

Electronic support measures and defensive aids are used extensively to gather information about threats or possible threats. They can be used to launch devices (in some cases automatically) to counter direct threats against the aircraft. They are also used to determine the state of a threat and identify it.

Aircraft Networks

The avionics systems in military, commercial and advanced models of civilian aircraft are interconnected using an avionics databus. Common avionics databus protocols, with their primary application, include:

- Aircraft Data Network (ADN): Ethernet derivative for Commercial Aircraft
- Avionics Full-Duplex Switched Ethernet (AFDX): Specific implementation of ARINC 664 (ADN) for Commercial Aircraft
- ARINC 429: Generic Medium-Speed Data Sharing for Private and Commercial Aircraft
- ARINC 629: Commercial Aircraft (Boeing 777)
- ARINC 708: Weather Radar for Commercial Aircraft
- ARINC 717: Flight Data Recorder for Commercial Aircraft
- IEEE 1394b: Military Aircraft
- MIL-STD-1553: Military Aircraft
- MIL-STD-1760: Military Aircraft
- TTP - Time-Triggered Protocol: Boeing 787 Dreamliner, Airbus A380, Fly-By-Wire Actuation Platforms from Parker Aerospace
- TTEthernet - Time-Triggered Ethernet: NASA Orion Spacecraft

Police and Air Ambulance

Police and EMS aircraft (mostly helicopters) are now a significant market. Military aircraft are often now built with a role available to assist in civil disobedience. Police helicopters are almost always fitted with video/FLIR systems to allow them to track suspects. They can also be fitted with searchlights and loudspeakers.

EMS and police helicopters will be required to fly in unpleasant conditions, this may require more aircraft sensors, some of which were until recently considered purely for military aircraft.

Chapter-3

Boilerplate (spaceflight)



Boilerplate version of Gemini spacecraft on display at Air Force Space & Missile Museum, Cape Canaveral, Florida October 15, 2004.

The term **boilerplate** in rocketry refers to a nonfunctional craft, system, or payload which is used to test various configurations and basic size, load, and handling characteristics. It is far less expensive to build multiple, full-scale, non-functional boilerplate spacecraft than it is to develop the full system (design, test, redesign, and launch). In this way, boilerplate spacecraft allow components and aspects of cutting-edge aerospace projects to

be tested while detailed contracts for the final project are being negotiated. During these tests, procedures are developed in mating boilerplates to rocket boosters along with emergency access and egress, maintenance support activities, and various transportation processes.

Boilerplate spacecraft are most commonly used to test manned spacecraft; for example, in the early 1960s, NASA performed many tests of **boilerplates**. Such boilerplates were made for Apollo spacecraft atop Saturn I rockets, and Mercury spacecraft atop Atlas rockets (for example Big Joe 1). Space Shuttle Enterprise was used as both a ground test boilerplate spacecraft and an atmospheric flight test vehicle. The development of NASA's Project Constellation used boilerplate Orion spacecraft atop an Ares I rocket for initial testing.

Mercury boilerplates

Mercury boilerplates were manufactured "in-house" by Langley AFB technicians prior to McDonnell Aircraft Company building the Mercury spacecraft. The boilerplate capsules were designed and used to test spacecraft recovery systems, and escape tower and rocket motors. Formal tests were done on the test pad at Langley and at Wallops Island using the Little Joe rockets.

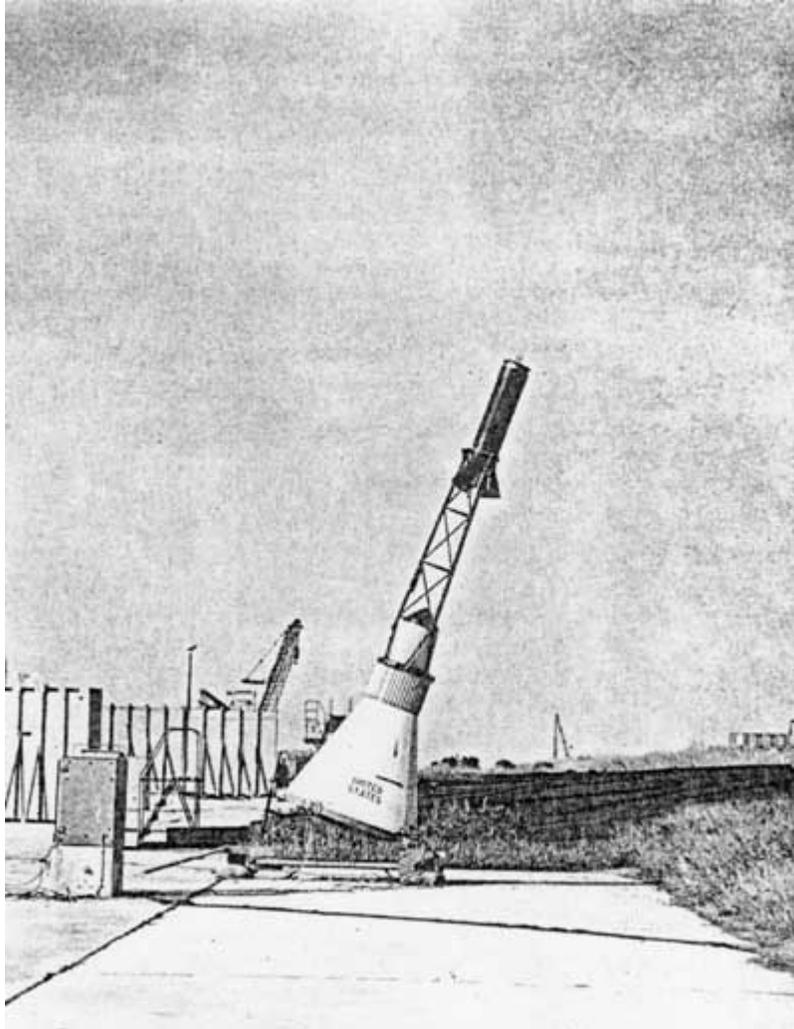
A summary of Mercury boilerplates can be found at A Field Guide to American Spacecraft. The term *boilerplate* has been wrongly used since the Little Joe series of 7 boosters. There was only one boilerplate capsule and it was called as such, as its conical section was made of steel at the Norfolk Naval shipyard. It was used in a beach abort test, and then subsequently used in the LJ1A flight. The term was used thereafter for the *prototype* capsules, which in their own right were nearly as complicated as the orbital capsules.

Notable events

- 1959 July 22 - First successful pad abort flight test with a functional escape tower attached to a Mercury Boilerplate.
- 1959 July 28 - A Mercury Boilerplate with instruments to measure sound pressure levels and vibrations from the Little Joe test rocket and Grand Central abort rocket/escape tower.
- 1959 September 9 - A Big Joe Atlas boilerplate Mercury (BJ-1) was successfully launched and flown from Cape Canaveral. This test flight was to determine the performance of the heat shield and heat transfer to the boilerplate, to observe flight dynamics of boilerplate during re-entry into the South Atlantic, to perform and evaluate capsule floatation and recovery system procedures, and to evaluate the entire capsule and rocket characters and system controls.
- 1960 May 9 - Beach Abort test with a Little Joe booster was successful.
- 1961 February 25 - A successful drop test of the Mercury Boilerplate spacecraft fitted with impact skirt, straps and cables, and a heat shield.

- 1961 March 24 - A successful Mercury-Redstone BD (MR-3) launched occurred with an apogee of 181 km (112 mi); first sub-orbital unmanned flight.

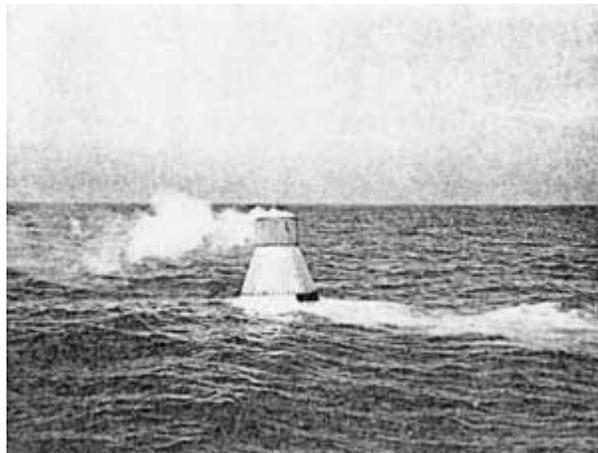
Photos



Mercury Beach Abort test



Mercury parachute test

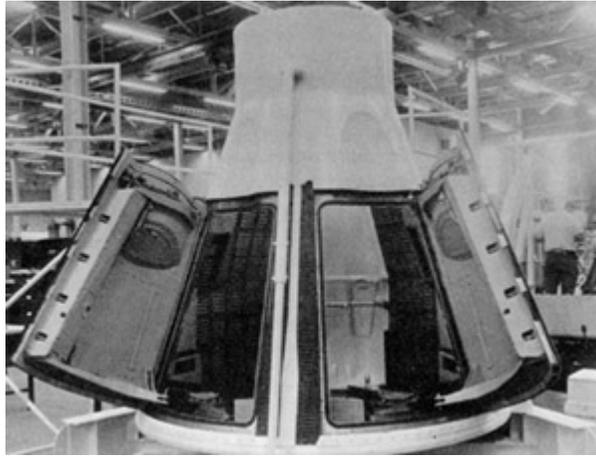


Mercury flotation test

Gemini boilerplates

There were seven Gemini boilerplates: BP-1, 2, 3, 3A, 4, 5, and 201. Gemini Boilerplate 3A had functional doors and had multi-uses for testing watertightness, flotation collars, and egress procedures.

Photos



McDonnell plant, St. Louis, Missouri



Flotation and rescue test



Flotation and egress test



Boilerplate on the USS Hornet Museum

Apollo boilerplates

NASA created a variety of Apollo boilerplates. A list of them can be found in Apollo Section of A Field Guide to American Spacecraft.

Launch escape system tests (LES)

Apollo boilerplates were used in the Launch Escape System (LES) for tests of the jettison tower rockets and procedures:

- BP-6 with Pad Abort Test-1 - LES pad abort test from launch pad; with photo.
- BP-23A with Pad Abort Test-2 - LES pad abort test of near Block-I CM; with photo.
- BP-23 with Mission A-002 Test Flight - LES test of canards, Oct.29-Nov.5, 1964.
- BP-27 with LES-015 - Dynamic tests.

Boilerplate tests

- BP-1 - Water impact tests
- BP-2 - Flotation tests storage
- BP-3 - Parachute tests
- BP-6,-6B, - PA-1, later Parachute drop test vehicle, and LES pad abort flight test to demonstrate launch escape system's (LES) pad abort(PA) performance at White Sands Missile Range.
- BP-9 with Mission A-105(SA-10) Test Flight, Micro Meteoroid Dynamic Test; not recovered.
- BP-12 with Mission A-001 Test Flight, now at former NASA Facility, Downey, CA to test the LES transonic abort flight performance at White Sands Missile Range.
- BP-13 with Mission A-101(SA-6) Test Flight, not recovered
- BP-14 with environmental control system tests, Oct. 22-29, 1964, consisted of Command Module 14, Service Module 3, Launch Escape System 14, and Saturn Launch Adapters.
- BP-15 with Mission A-102(SA-7) Test Flight, not recovered.
- BP-16 with Mission A-103 Test Flight, another Micro Meteoroid test, not recovered.
- BP-19A - VHF antenna, parachute drop tests; now at the Columbia Memorial Space Center (former NASA Facility, Downey, CA)
- BP-22 with Mission A-003 Test Flight; boilerplate on display at Johnson Space Center, Houston, TX
- BP-23 - LES high-dynamic-pressure abort flight performance tests at White Sands Missile Range.
- BP-23A - LES pad abort flight performance tests with Canard, BPC, and major sequencing changes at White Sands Missile Range, now displayed with SA-500D at the U.S. Space & Rocket Center, Huntsville, Alabama.



BP-29 at Barringer Crater (used for flotation tests)

- BP-25 Command Module (CM) - Water recovery test, at Fort Worth Museum of Transportation
- BP-26 with Mission A-104(SA-8) Test Flight - another Micro Meteoroid test.
- BP-27 CM and Service Module with LES-16 - Stack and engine gimbal test. Now on display atop the vertical Saturn V at the U.S. Space & Rocket Center, Huntsville, Alabama.
- BP-28A - Impact tests
- BP-29 - Uprighting drop tests at Downey, CA, Oct. 30, 1964, on display at Barringer Crater, Arizona
- BP-30 - Swing arm tests; currently on display at Kennedy Space Center's Apollo/Saturn V Center

Specific Apollo BP units

BP-1101A

BP-1101A was used in numerous tests to develop spacecraft recovery equipment and procedures. Specifically, 1101A tested the air bags as part of the "up-righting" procedure

when the Apollo lands upside down in the water. The sequence of the bags inflating caused the capsule to roll and up-right itself.

This McDonnell boilerplate is now on loan to the Wings Over the Rockies Air and Space Museum, Denver, Colorado, from the Smithsonian. BP-1101A has an external painted marking of AP.5. Examination of the interior in 2006 revealed large heavy steel ingots. After further research, a new paint scheme was applied in June 2007.



BP1101A AP5. Front view, Wings Museum, 2006.



BP1101A AP5. Side view.



**APOLLO COMMAND
MODULE BOILERPLATE**
BP-1101A

Before the engineers of the National Aeronautics and Space Administration (NASA) ever thought of sending man to the moon, they knew they had to be safe. To do this required an exact knowledge of the instruments and their machine could endure at the first landing on

isolation, protection, heatshields, and structures were of major importance. It quickly became apparent that testing the actual Apollo capsule in the highest priority flight American Aviation was tested with one "boilerplate" made up of the scraps for the many different tests would need to be performed.

The boilerplate was a simple metal model created specifically for the boilerplate simulated the size, weight, shape, and center of a normal Apollo command module. Some parts made for landings, while others were designed to test the feasibility of the future V-2 systems. These "junk" tests were conducted at the White Sands Test Facility. More than 60 different boilerplate mock-ups were constructed during the Apollo program.

Wings Over the Rockies boilerplate was acquired through the Smithsonian Institution. Before coming to Wings, it was in the custody of the Smithsonian Institution.

The most tests BP-1101A was used for being not come to but some evidence suggests that it was used for launch tests. It is also possible that BP-1101A was used for its systems to be used in the Apollo 5 launch. An uncrewed launch Apollo 5 was the first flight of the Apollo Lunar Module, which would later carry astronauts to the moon's surface.



WORLD TECHNOLOGIES
SMITHSONIAN INSTITUTION

Display plaque.



New paint scheme June 2007.

BP-1102A

BP-1102 was used for water egress trainer for all Apollo flights, including by the crew of Apollo 11, the first lunar landing mission. It was also adapted for mock-up interior components and used by astronauts to practice routine and emergency exits from the spacecraft.

It was then modified again where the interior was set up to be configured either as Apollo/Soyuz or a five-man rescue vehicle as once proposed for Skylab. With these two conversions, astronauts could train for those special missions. It was finally transferred from NASA to the Smithsonian in 1977, and is displayed now at the Hazy Center with the flotation collar and bags that were attached to Columbia at the end of its historic mission.

BP-1220/1228 Series

The purpose of this series design was to simulate the weight and other external physical characteristics of the Apollo command module. These prototypes were in the 9000 lb range for both laboratory water tanks and ocean tests. The experiments tested floatation

collars, collar installations, and buoyancy characteristics. The Navy trained their recovery personnel for ocean collar installation and shipboard retrieval procedures. These boilerplates rarely had internal equipment.

BP-1224

BP-1224 was a Component level Flammability Test Program to test for design decisions on selection and application of nonmetallic materials. Boilerplate configuration comparisons with Command Service Module 2TV-1 and 101 were performed by North American. The NASA Review Board decided on February 5, 1967, that the boilerplate configuration had determined a reasonable "worst case" configuration, after more than 1,000 tests were performed.

BP-1227

This was lost in the North Sea in early 1970, recovered by a Hungarian vessel, transferred to the Soviet Union, and returned to the US in September 1970 by the USCGC Southwind (WAGB-280). It is now located in Grand Rapids, Michigan as a time capsule. The only certainties about this capsule are that it was returned to the United States at Murmansk early in September 1970 during a visit by the USCG Southwind who returned it to the Naval Air Station, Norfolk, Virginia. There it remained until title was passed to the Smithsonian in April 1976 when it was passed on to Grand Rapids, Michigan to serve as a time capsule. Two official sources – the US Navy and the US Coastguard – both say that it was lost by an ARRS unit training in recovery procedures. Where it was lost is not certain, but the most likely location is the Bay of Biscay where it was recovered by a Russian trawler. A contemporary account of its return quotes a NASA spokesman as saying, “... as far as NASA can determine the object... the Navy lost two years ago.”. When BP-1227 was lost ranges from 1968 to 1970 depending on which account one reads. This uncertainty may be due to a Russian account that claims there is an agreement between the Russians and the US to keep details secret until 2021.

Space Shuttle OV-101 in boilerplate configuration



Enterprise on Launch Pad 39A

First in March 1978 at the Marshall Space Flight Center and then again in June 1979, the Space Shuttle *Enterprise* was fitted together with an external tank and two inert solid rocket motors in a test-bed or boilerplate configuration. The STS-1 preliminary mission test program consisted of vibration tests in a horizontal mode at the Marshall Center, and then in a vertical launch configuration on Launch Pad 39A at Kennedy Space Center, Florida. In 1985, the boilerplate configuration was used to test the Air Force shuttle facilities at Vandenberg Air Force Base, including a full mating on the SLC-6 launch pad.

Orion boilerplate

Ares boosters

NASA's future space flights to the Moon are being planned for 2015. These flights will be based upon the Orion spacecraft and its Ares booster. The Shuttles are planned to be retired in 2010. The Orion boilerplates will be used between 2008 and 2014 using the Ares I booster and the heavy-lift launches Ares V, both of which are slated to launch initially from NASA's Pad 39B site at the Kennedy Space Center in Florida.

Development

The construction of the first Orion boilerplate, will be a basic mockup prototype to test the assembling sequences and launch procedures at NASA's Langley Research Center while Lockheed aerospace engineers assemble the first rocket motors for the spacecraft's escape tower. The first boilerplate will go to Dryden Flight Research Center at Edwards, California, for integration of Lockheed's avionics and NASA's developmental flight instrumentation prior to shipment to New Mexico's White Sands Missile Range for the first Orion pad abort test (PA-1) in 2009. On November 20, 2008 a complete test of the abort rockets took place in Utah. PA-1 is the first of the six test events in Orion Abort Flight Test subproject. Lockheed Martin Corp. was awarded the contract to build Orion on Aug. 31, 2006.

Other boilerplates will be used to test thermal, electromagnetic, audio, mechanical vibration conditions and research studies. These tests for the Orion spacecraft will be done at Plum Brook Station in the agency's Ohio-based Glenn Research Center. The first boilerplate Orions will launched/tested as early as 2008.

Pathfinder

On March 2, 2009, the LAS Pathfinder began its transfer from the Langley Research Center to the White Sands Missile Range, for first PA-1 launch tests. Pathfinder is the combination of the Orion boilerplate and the LAS module.

Post-landing Orion Recovery Test (PORT)

On March 23, 2009 a Navy-built Orion boilerplate began the PORT Tests in Navy test facilities and then sea testing near the Kennedy Space Center.

Photos



Orion full size boilerplate getting its first coat of paint.



Painted at Dryden Research Center.



Ready for testing.



Navy-built, 18,000-pound Orion mock-up in a test pool at the Naval Surface Warfare Center's Carderock Division in West Bethesda, Md.

Project Constellation

The Orion-Ares configuration is known as a part of NASA's *Project Constellation*. This project's plan is to send humans to the Moon, Mars, and other destinations in the solar system. Its base components will consist of the Launch Abort System, the Crew Module, the Service Module, and the Spacecraft Adapter.

Chapter-4

Electrodynamic Tether



Medium close-up view, captured with a 70mm camera, shows Tethered Satellite System deployment.

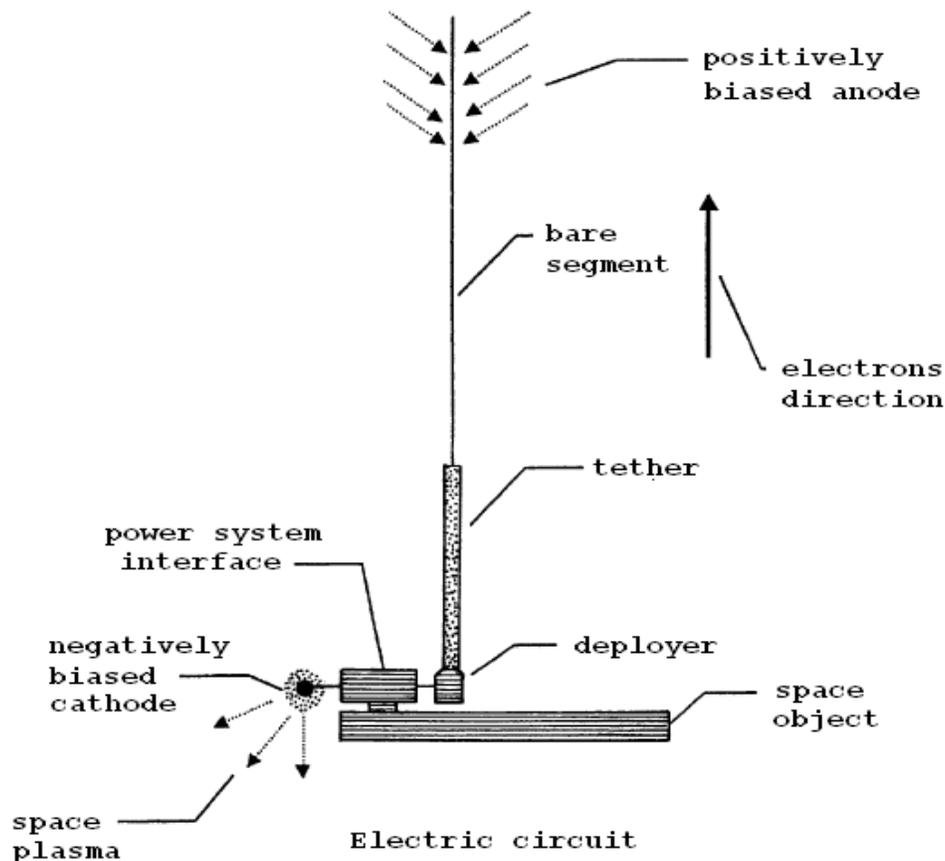
Electrodynamic tethers are long conducting wires, such as one deployed from a tether satellite, which can operate on electromagnetic principles as generators, by converting their kinetic energy to electrical energy, or as motors, converting electrical energy to kinetic energy. Electric potential is generated across a conductive tether by its motion through the Earth's magnetic field. The choice of the metal conductor to be used in an

electrodynamic tether is determined by a variety of factors. Primary factors usually include high electrical conductivity, and low density. Secondary factors, depending on the application, include cost, strength, and melting point.

Tether propulsion

As part of a *tether propulsion* system, crafts can use long, strong conductors (though not all tethers are conductive) to change the orbits of spacecraft. It has the potential to make space travel significantly cheaper. It is a simplified, very low-budget magnetic sail. It can be used either to accelerate or brake an orbiting spacecraft. When direct current is pumped through the tether, it exerts a force against the magnetic field, and the tether accelerates the spacecraft.

Tethers as generators



A space object, i.e. a satellite in Earth orbit, or any other space object either natural or man made, is physically connected to the tether system. The tether system comprises a deployer from which a conductive tether having a bare segment extends upward from space object. The positively biased anode end of tether collects electrons from the

ionosphere as space object moves in direction across the Earth's magnetic field. These electrons flow through the conductive structure of the tether to the power system interface, where it supplies power to an associated load, not shown. The electrons then flow to the negatively biased cathode where electrons are ejected into the space plasma, thus completing the electric circuit.

An electrodynamic tether is attached to an object, the tether being oriented at an angle to the local vertical between the object and a planet with a magnetic field. When the tether cuts the planet's magnetic field, it generates a current, and thereby converts some of the orbiting body's kinetic energy to electrical energy. As a result of this process, an electrodynamic force acts on the tether and attached object, slowing their orbital motion. The tether's far end can be left bare, making electrical contact with the ionosphere. Functionally, electrons flow from the space plasma into the conductive tether, are passed through a resistive load in a control unit and are emitted into the space plasma by an electron emitter as free electrons. In principle, compact high-current tether power generators are possible and, with basic hardware, 10 to 25 kilowatts appears to be attainable.

Voltage and current

NASA has conducted several experiments with Plasma Motor Generator (PMG) tethers in space. An early experiment used a 500 meter conducting tether. In 1996, NASA conducted an experiment with a 20,000-meter conducting tether. When the tether was fully deployed during this test, the orbiting tether generated a potential of 3,500 volts. This conducting single-line tether was severed after five hours of deployment. It is believed that the failure was caused by an electric arc generated by the conductive tether's movement through the Earth's magnetic field.

When a tether is moved at a velocity (v) at right angles to the Earth's magnetic field (\mathbf{B}), an electric field is observed in the tether's frame of reference. This can be stated as:

$$\mathbf{E} = v * \mathbf{B} = v\mathbf{B}$$

The direction of the electric field (\mathbf{E}) is at right angles to both the tether's velocity (v) and magnetic field (\mathbf{B}). If the tether is a conductor, then the electric field leads to the displacement of charges along the tether. Note that the velocity used in this equation is the orbital velocity of the tether. The rate of rotation of the Earth, or of its core, is not relevant. In this regard.

Voltage across conductor

With a long conducting wire of length L , an electric field E is generated in the wire. It produces a voltage V between the opposite ends of the wire. This can be expressed as:

$$V = \mathbf{E} \cdot \mathbf{L} = EL \cos \tau = vBL \cos \tau$$

where the angle τ is between the length vector (L) of the tether and the electric field vector (E), assumed to be in the vertical direction at right angles to the velocity vector (v) in plane and the magnetic field vector (B) is out of the plane.

Current in conductor

An electrodynamic tether can be described as a type of thermodynamically "open system". Electrodynamic tether circuits cannot be completed by simply using another wire, since another tether will develop a similar voltage. Fortunately, the Earth's magnetosphere is not "empty", and, in near-Earth regions (especially near the Earth's atmosphere) there exist highly electrically conductive plasmas which are kept partially ionized by solar radiation or other radiant energy. The electron and ion density varies according to various factors, such as the location, altitude, season, sunspot cycle, and contamination levels. It is known that a positively charged bare conductor can readily remove free electrons out of the plasma. Thus, to complete the electrical circuit, a sufficiently large area of uninsulated conductor is needed at the upper, positively charged end of the tether, thereby permitting current to flow through the tether.

However, it is more difficult for the opposite (negative) end of the tether to eject free electrons or to collect positive ions from the plasma. It is plausible that, by using a very large collection area at one end of the tether, enough ions can be collected to permit significant current through the plasma. This was demonstrated during the Shuttle orbiter's TSS-1R mission, when the shuttle itself was used as a large plasma contactor to provide over an ampere of current. Improved methods include creating an electron emitter, such as a thermionic cathode, plasma cathode, plasma contactor, or field electron emission device. Since both ends of the tether are "open" to the surrounding plasma, electrons can flow out of one end of the tether while a corresponding flow of electrons enters the other end. In this fashion, the voltage that is electromagnetically induced within the tether can cause current to flow through the surrounding space environment, completing an electrical circuit through what appears to be, at first glance, an open circuit.

Tether current

The amount of current (I) flowing through a tether depends on various factors. One of these is the circuit's total resistance (R). The circuit's resistance consist of three components:

1. the effective resistance of the plasma,
2. the resistance of the tether, and
3. a control variable resistor.

In addition, a parasitic load is needed. The load on the current may take the form of a charging device which, in turn, charges reserve power sources such as batteries. The batteries in return will be used to control power and communication circuits, as well as drive the electron emitting devices at the negative end of the tether. As such the tether

can be completely self-powered, besides the initial charge in the batteries to provide electrical power for the deployment and startup procedure.

The charging battery load can be viewed as a resistor which absorbs power, but stores this for later use (instead of immediately dissipating heat). It is included as part of the "control resistor". The charging battery load is not treated as a "base resistance" though, as the charging circuit can be turned off at anytime. When off, the operations can be continued without interruption using the power stored in the batteries.

Challenges

One complication to these techniques is that if the tether rotates, the direction of current must reverse (such as is the case in alternating currents of alternators). Others include pendular motion instability and electrical surges.

Pendular motion instability

Electrodynamic tethers deployed along the local vertical ('hanging tethers') may suffer from dynamical instability. Pendular motion causes the tether vibration amplitude to build up under the action of electromagnetic interaction. As the mission time increases, this behavior can compromise the performance of the system. Over a few weeks, electrodynamic tethers in Earth orbit might build up vibrations in many modes, as their orbit interacts with irregularities in magnetic and gravitational fields.

One plan to control the vibrations is to actively vary the tether current to counteract the growth of the vibrations. Electrodynamic tethers can be stabilized by reducing their current when it would feed the oscillations, and increasing it when it opposes oscillations. Simulations have demonstrated that this can control tether vibration. This approach requires sensors to measure tether vibrations, which can either be an inertial navigation system on one end of the tether, or satellite navigation systems mounted on the tether, transmitting their positions to a receiver on the end.

Another proposed method is to utilise spinning electrodynamic tethers instead of hanging tethers. The gyroscopic effect provides passive stabilisation, avoiding the instability.

Surges

As mentioned earlier, conductive tethers have failed from unexpected current surges. Unexpected electrostatic discharges have cut tethers, damaged electronics, and welded tether handling machinery. It may be that the Earth's magnetic field is not as homogeneous as some engineers have believed.

Chapter-5

Gemini Space Suit



Gemini G3C Suit



G4C with ELSS on chest and ESP on back

The **Gemini space suit** is a space suit worn by astronauts for launch, in-flight activities (including EVAs) and landing. It was designed by NASA based on the X-15 high-altitude pressure suit, and has been used since Gemini, in various forms, by the U.S. Air Force and NASA on U-2 and SR-71 Blackbird high-altitude flights and on the Space Shuttle. All Gemini spacesuits were developed and manufactured by the David Clark Company in Worcester, Massachusetts.

G3C and G4C suits

The G3C and G4C suits were the primary spacesuits worn for all but the Gemini 7 mission. The G3C consisted of six layers of nylon (the innermost containing a rubberized

nylon "bladder") and nomex, with a link net retaining layer and an outer layer of white nomex fabric. It had removable combat-style boots, also made of nomex fabric, along with a full-pressure helmet (containing a set of earphones and microphones) and gloves detachable by improved locking rings that allow easy rotation of the wrists. On Gemini 3 the G3C suit was worn by both Gus Grissom and John W. Young and was the only flight to use this suit.

The G4C suit was identical to the G3C suit, but came in two different styles. Both had additional layers of mylar insulation for temperature control (+250°F in direct sunlight and -250°F in shadow), but the commander's suit retained the removable boots, while the pilot's version had integrated boots and a detachable sun visor which clipped onto the helmet. The G4C suit was worn by all crews from Gemini 4 to its end and it was in this suit that Ed White made the first American spacewalk in 1965. Except for one modification, the incorporation of additional layers in the legs, as well as an outer layer of "Chromel-R" fabric on the Gemini 9A spacesuit worn by Eugene Cernan (he was to test the Astronaut Maneuvering Unit, which used hydrogen peroxide as its fuel), and the replacement of the plexiglas helmet faceplate with one made of polycarbonate plastic.

G5C suits



G5C space suit

For the 14-day Gemini 7 mission, both Frank Borman and Jim Lovell wore modified G3C suits, but incorporating several changes:

- Replacement of the pressure helmet and neck ring with a zippered hood incorporating a clear, fixed polycarbonate visor, with the astronauts wearing modified Navy-style aviator crash helmets that incorporated the communication equipment (microphones and earphones).
- Additional zippers for in-flight adjustment, along with provisions for complete removal of the suit.

This configuration is the G5C suit. During the mission, Lovell was the first person to take his pressure suit off, which was achieved with great difficulty due to his size. Borman later was able to get his suit off and biomedical data collected during the flight revealed that astronauts would be more comfortable during the flight when wearing flight suits during "non-critical" phases of the mission. This led to the wearing of such flight suits from Apollo 7 to the present day. This suit somewhat resembles the current Soyuz Sokol pressure suits worn by Russian-launched ISS crews.

After Gemini

After the ending of Gemini, the Gemini space suit was chosen by NASA for the initial Apollo "Block I" testing phase of the program. Since no dockings or EVAs were scheduled for the first manned Apollo flight, and with NASA searching for a suitable lunar EVA suit for the "Block II" phase of the program (a competition between ILC Dover, Hamilton Sundstrand, and David Clark), NASA decided to use the G3C as a base for the new "A1C" suit. Using the base G3C suit, but with a white nomex cover layer from the G4C and G5C suits, the new A1C suits had new electrical and environmental disconnects, a protective shell over the plexiglas visor (plexiglas was used instead of polycarbonate due to its expense), and with a launch escape system in place of ejector seats, a yellow-colored U-shaped inflatable "Mae West" PFD replaced the pilot parachute and its harness. Unfortunately, the Apollo 1 accident, which killed astronauts Grissom, White, and Roger Chaffee resulted in NASA cancelling the "Block I" program and starting with the flight of Apollo 7, all Apollo equipment, including spacesuits, was tested using the "Block II" configuration. This included the docking equipment on the Apollo CM and the ILC A7L space suit.

Since Apollo, the Gemini spacesuit was looked at for the Manned Orbiting Laboratory program, but it has since been used as the baseline for all high-altitude pressure suits worn by U.S. Air Force, and later by NASA as its current ACES pressure suit. On June 11, 2008, the David Clark Company was selected by the Houston, Texas-based Oceaneering International as a subcontractor for the manufacture of the new Constellation Space Suit system, in which its "Operation One" configuration, resembles the current ACES suit, but functions in the same manner as that of the Gemini suit.



G4C with ELSS on chest and AMU on back



View of G4C Suits entry zipper

Specifications

The suit family system included both parachute and flotation systems. For EVAs Gemini 4 used the Ventilation Control Module (VCM), for Gemini 8-12 the Extravehicular Life Support System (ELSS) was used. The ELSS was also designed to optionally supply autonomous life support though they never made it to EVA, two oxygen supply packs were developed for it, one was the Extravehicular Support Package (ESP)(provide an hour's worth of life support) carried aboard Gemini 8 but not used and the other for the Astronaut Maneuvering Unit (provide an hour's worth of life support) carried on-board Gemini 9 and to fly while tethered but was not used. The AMU was also meant to be launched and flown on-board Gemini 12, and to fly untethered from the Gemini spacecraft but was scrubbed two months before the mission.

Name: Gemini G3C Spacesuit

Manufacturer: David Clark Company

Missions: Gemini 3, Gemini 6 (Schirra), Gemini 8 (Armstrong)

Function: Intra-vehicular activity (IVA) and Ejection

Operating Pressure: 3.7 psi (25.5 kPa)

Suit Weight: 23.5 lb (10.7 kg)

Primary Life Support: Vehicle Provided

Backup Life Support: Vehicle Provided



G4C space suits

Name: Gemini G4C Spacesuit

Manufacturer: David Clark Company

Missions: Gemini 4-6, 8-12

Function: Intra-vehicular activity (IVA), Ejection and orbital Extra-vehicular activity (EVA)

Operating Pressure: 3.7 psi (25.5 kPa)

Suit Weight: 34 lb (15.4 kg)

Ventilation Control Module (VCM) weight (Gemini 4): 7.75 lb (3.52 kg)

Extravehicular Life Support System (ELSS) weight (Gemini 8-12): 47 lb (21.3 kg)

Primary Life Support: Vehicle Provided

Backup Life Support (Gemini 4): 9 minutes

Backup Life Support (Gemini 5,6,8-12): 30 minutes

Name: Gemini G5C Spacesuit

Manufacturer: David Clark Company

Missions: Gemini 7

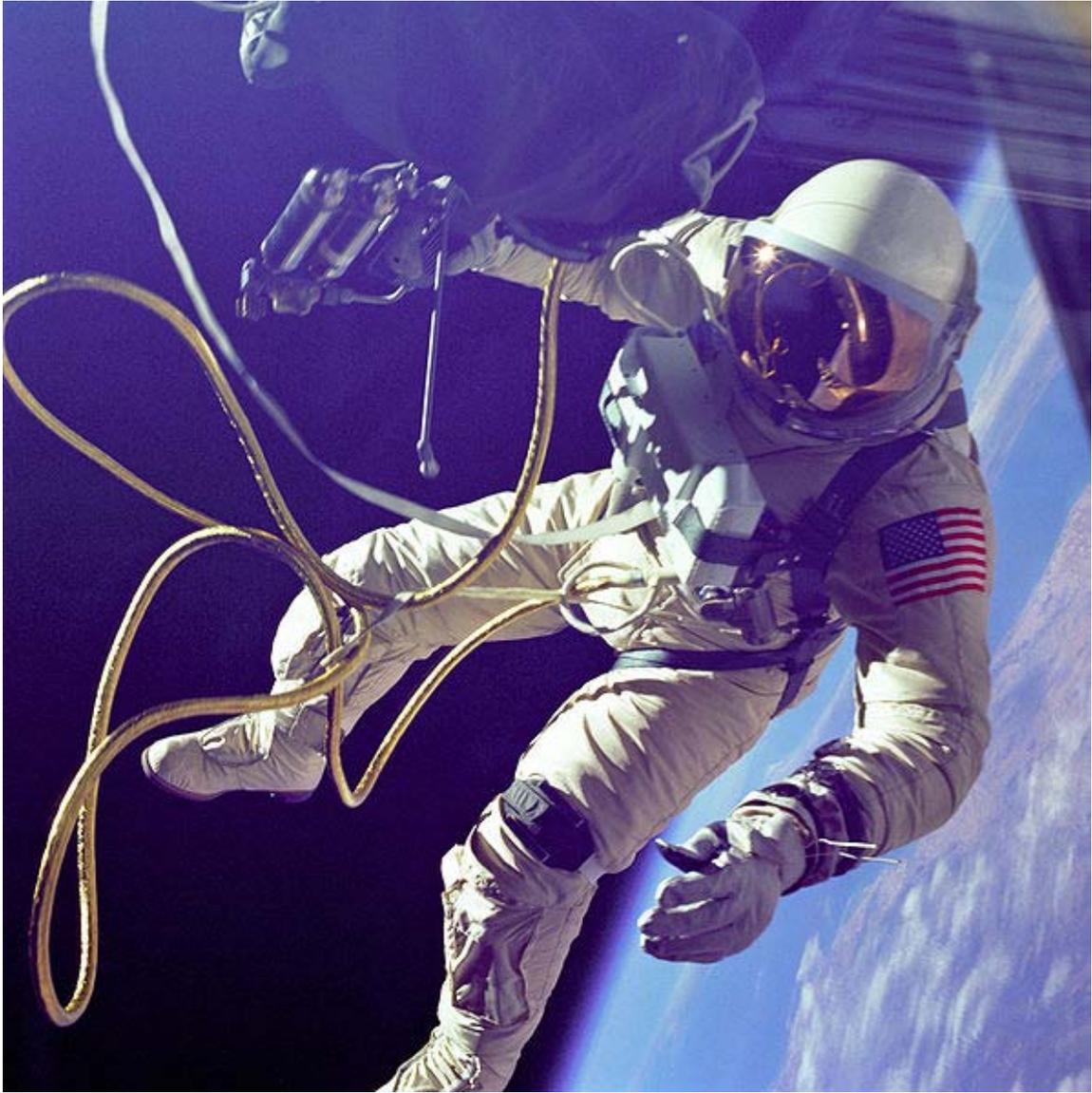
Function: Intra-vehicular activity (IVA) and Ejection

Operating Pressure: 3.7 psi (25.5 kPa)

Suit Weight: 16 lb (7.2 kg)

Primary Life Support: Vehicle Provided

Backup Life Support: Vehicle Provided



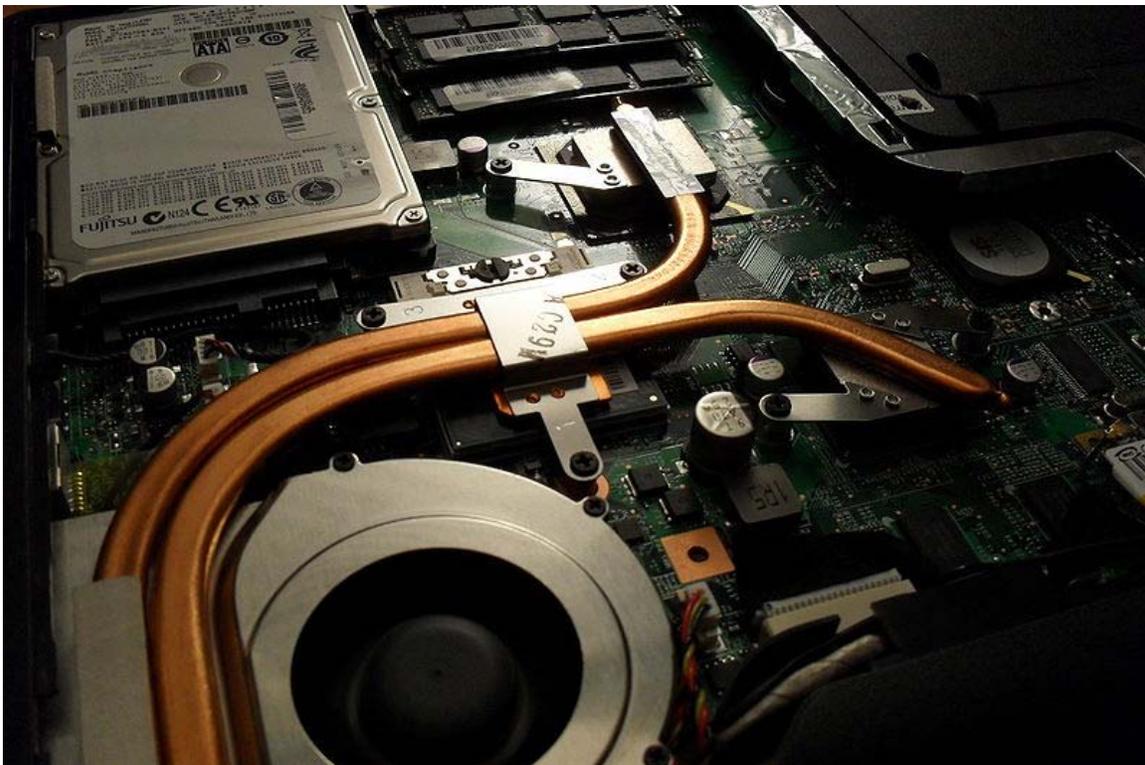
G4C with VCM on chest



G4C with ELSS on chest

Chapter-6

Heat Pipe



A laptop heat pipe system

A **heat pipe** is a heat transfer mechanism that combines the principles of both thermal conductivity and phase transition to efficiently manage the transfer of heat between two solid interfaces.

At the hot interface within a heat pipe, which is typically at a very low pressure, a liquid in contact with a thermally conductive solid surface turns into a vapor by absorbing heat from that surface. The vapor condenses back into a liquid at the cold interface, releasing the latent heat. The liquid then returns to the hot interface through either capillary action or gravity action where it evaporates once more and repeats the cycle. In addition, the

internal pressure of the heat pipe can be set or adjusted to facilitate the phase change depending on the demands of the working conditions of the thermally managed system.

Structure, design and construction

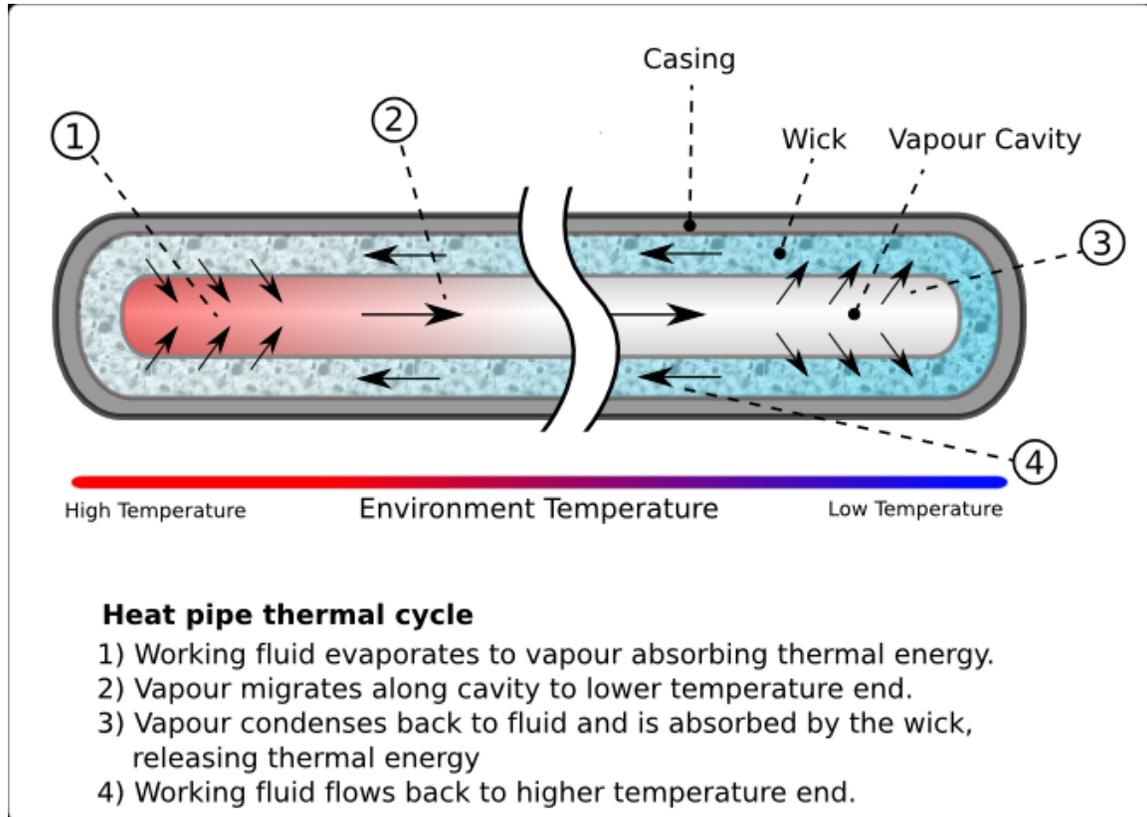
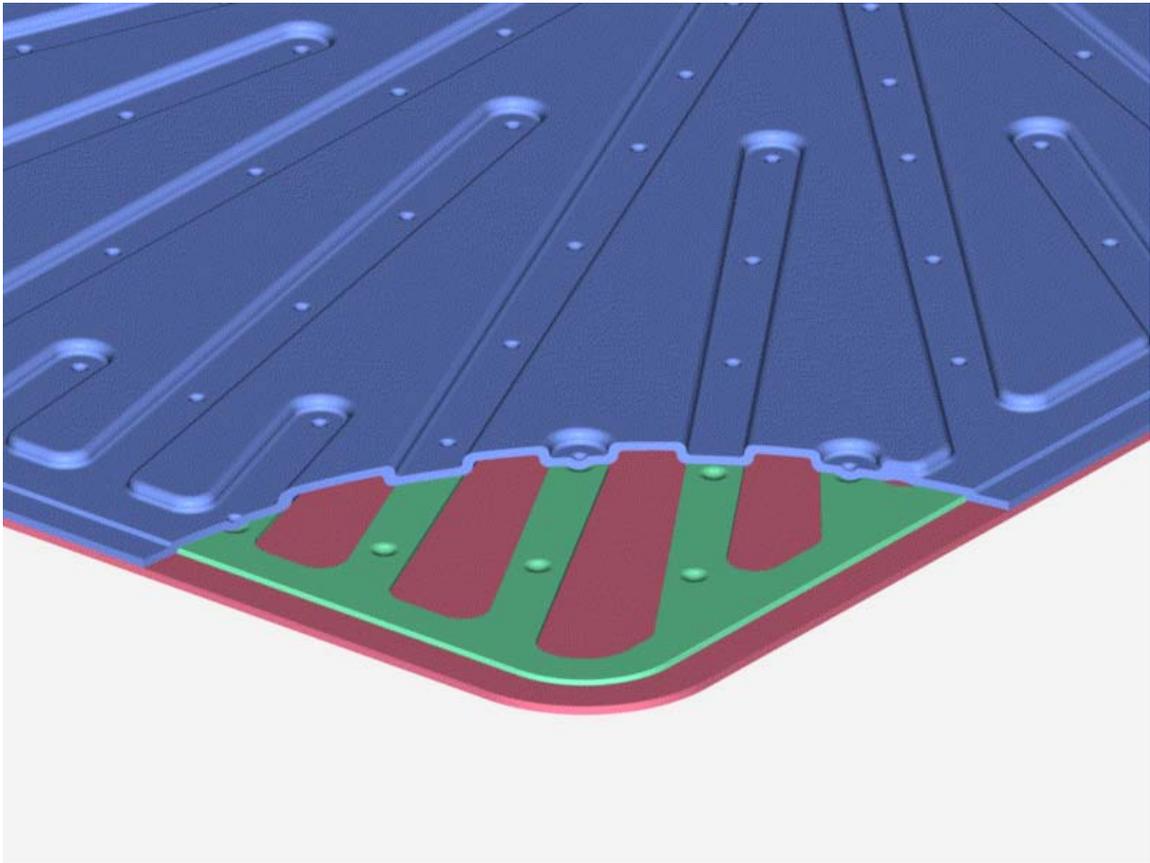
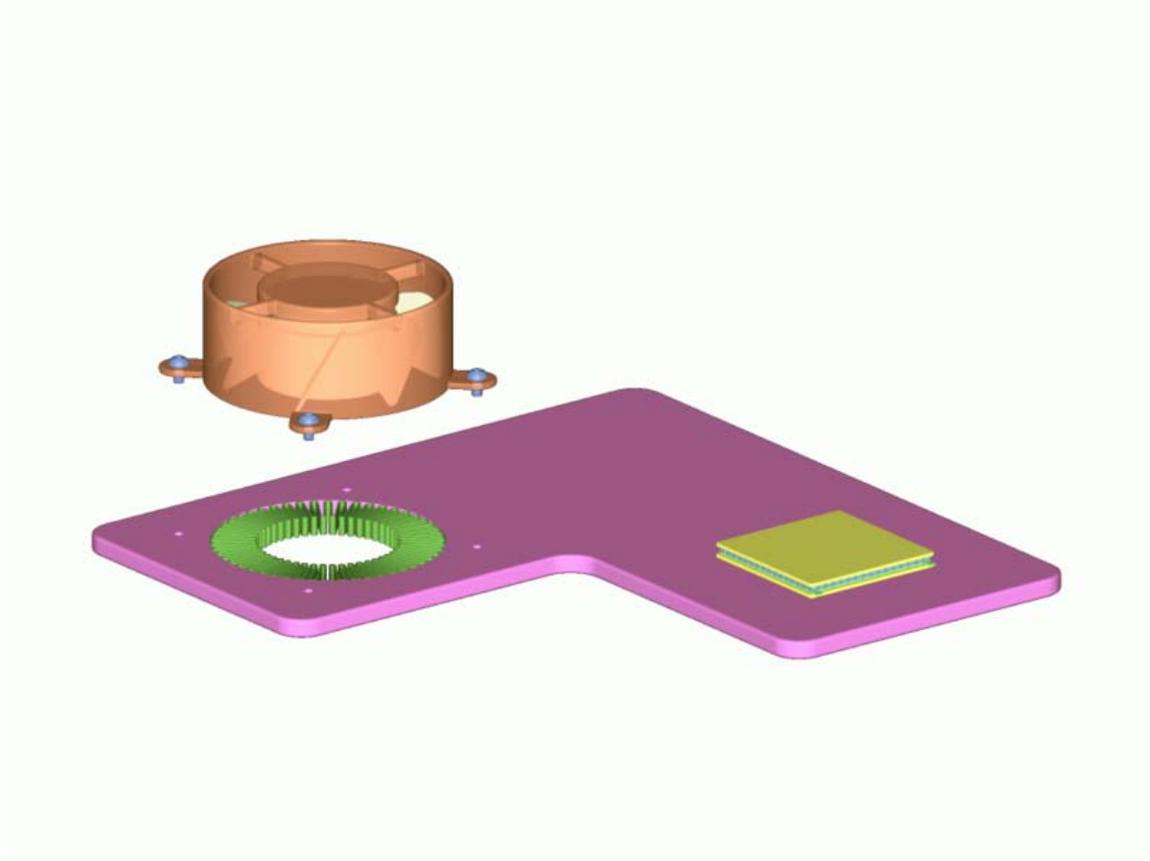


Diagram showing components and mechanism for a heat pipe containing a wick



Cut-away view of a 500 μm thick flat heat pipe, with a thin planar capillary (aqua colored)



Thin flat heat pipe (heat spreader) with remote heat sink and fan

A typical heat pipe consists of a sealed pipe or tube made of a material with high thermal conductivity such as copper or aluminium at both hot and cold ends. A vacuum pump is used to remove all air from the empty heat pipe, and then the pipe is filled with a fraction of a percent by volume of *working fluid* (or coolant) chosen to match the operating temperature. Examples of such fluids include water, ethanol, acetone, sodium, or mercury. Due to the partial vacuum that is near or below the vapor pressure of the fluid, some of the fluid will be in the liquid phase and some will be in the gas phase. The use of a vacuum eliminates the need for the working gas to diffuse through any other gas and so the bulk transfer of the vapor to the cold end of the heat pipe is at the speed of the moving molecules. In this sense, the only practical limit to the rate of heat transfer is the speed with which the gas can be condensed to a liquid at the cold end.

Inside the pipe's walls, an optional wick structure exerts a capillary pressure on the liquid phase of the working fluid. This is typically a sintered metal powder or a series of grooves parallel to the pipe axis, but it may be any material capable of exerting capillary pressure on the condensed liquid to wick it back to the heated end. The heat pipe may not need a wick structure if gravity or some other source of acceleration is sufficient to overcome surface tension and cause the condensed liquid to flow back to the heated end.

A heat pipe is not a thermosiphon, because there is no siphon. Thermosiphons transfer heat by single-phase convection.

Heat pipes contain no mechanical moving parts and typically require no maintenance, though non-condensing gases (that diffuse through the pipe's walls, result from breakdown of the working fluid, or exist as impurities in the materials) may eventually reduce the pipe's effectiveness at transferring heat. This is significant when the working fluid's vapour pressure is low.

The materials chosen depend on the temperature conditions in which the heat pipe must operate, with coolants ranging from liquid helium for extremely low temperature applications (2–4 K) to mercury (523–923 K) & sodium (873–1473 K) and even indium (2000–3000 K) for extremely high temperatures. The vast majority of heat pipes for low temperature applications use some combination of ammonia (213–373 K), alcohol (methanol (283–403 K) or ethanol (273–403 K)) or water (303–473 K) as working fluid. Since the heat pipe contains a vacuum, the working fluid will boil and hence take up latent heat at well below its boiling point at atmospheric pressure. Water, for instance, will boil at just above 273 K (0 degrees Celsius) and so can start to effectively transfer latent heat at this low temperature.

The advantage of heat pipes over many other heat-dissipation mechanisms is their great efficiency in transferring heat. They are a fundamentally better heat conductor than an equivalent cross-section of solid copper (a heat sink alone, though simpler in design and construction, does not take advantage of the principle of matter phase transition). Some heat pipes have demonstrated a heat flux of more than 230 MW/m², nearly four times the heat flux at the surface of the sun.

Active control of heat flux can be effected by adding a variable volume liquid reservoir to the evaporator section. Variable conductance heat pipes employ a large reservoir of inert immiscible gas attached to the condensing section. Varying the gas reservoir pressure changes the volume of gas charged to the condenser which in turn limits the area available for vapor condensation. Thus a wider range of heat fluxes and temperature gradients can be accommodated with a single design.

A modified heat pipe with a reservoir having no capillary connection to the heat pipe wick at the evaporator end can also be used as a thermal diode. This heat pipe will transfer heat in one direction, acting as an insulator in the other.

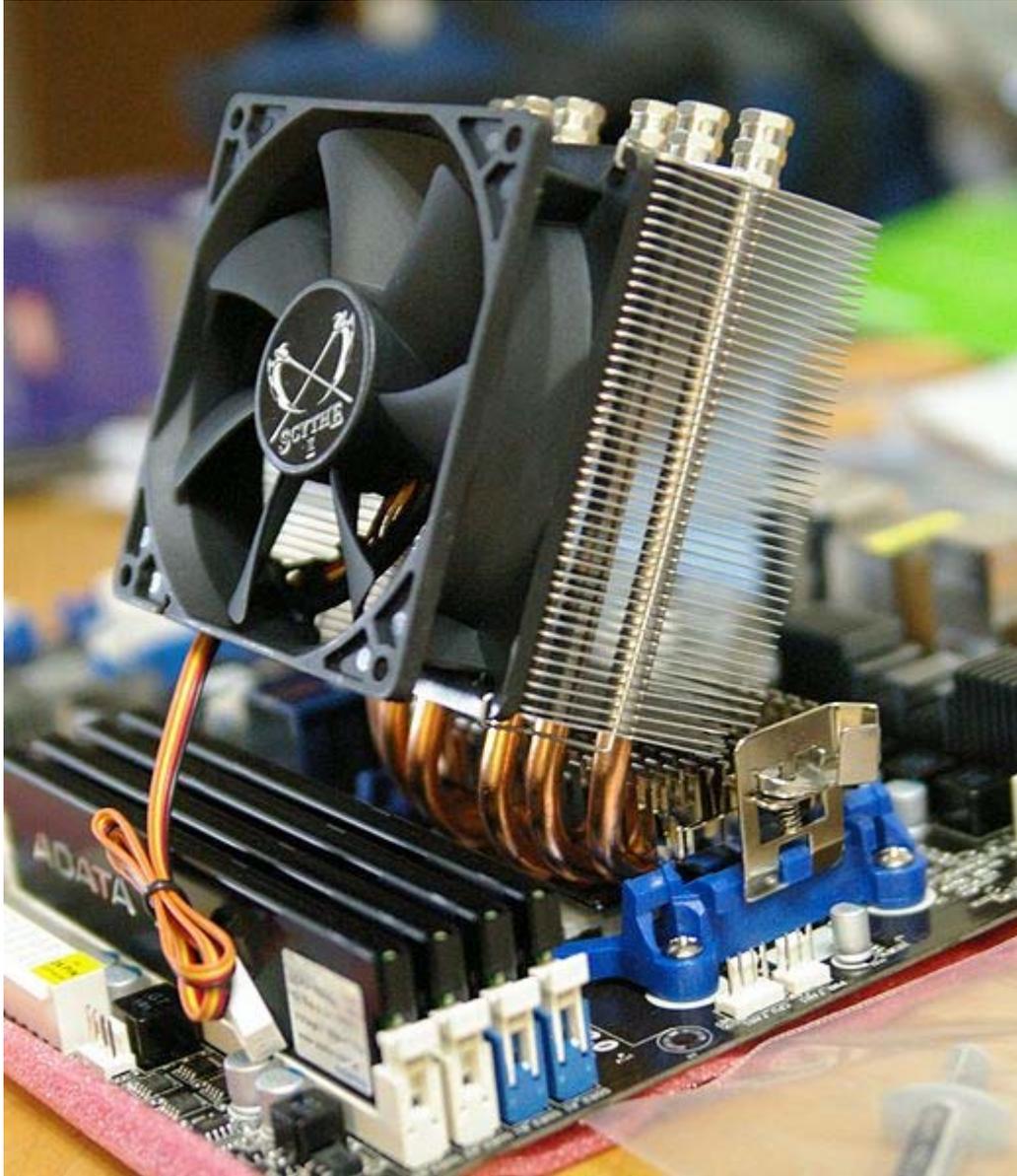
Vapor Chamber or Flat heat pipes

Thin planar heat pipes (heat spreaders) have the same primary components as tubular heat pipes. These components are a hermetically sealed hollow vessel, a working fluid, and a closed-loop capillary recirculation system.

Compared to a one-dimensional tubular heat pipe, the width of a two-dimensional heat pipe allows an adequate cross section for heat flow even with a very thin device. These

thin planar heat pipes are finding their way into “height sensitive” applications, such as notebook computers, and surface mount circuit board cores. It is possible to produce flat heat pipes as thin as 0.5 mm (thinner than a credit card).

Heat transfer



A heat sink (aluminium) with heat pipe (copper)

Heat pipes employ evaporative cooling to transfer thermal energy from one point to another by the evaporation and condensation of a working fluid or coolant. Heat pipes rely on a temperature difference between the ends of the pipe, and cannot lower temperatures at either end beyond the ambient temperature (hence they tend to equalise the temperature within the pipe).

When one end of the heat pipe is heated the working fluid inside the pipe at that end evaporates and increases the vapour pressure inside the cavity of the heat pipe. The latent heat of evaporation absorbed by the vaporisation of the working fluid reduces the temperature at the hot end of the pipe.

The vapour pressure over the hot liquid working fluid at the hot end of the pipe is higher than the equilibrium vapour pressure over condensing working fluid at the cooler end of the pipe, and this pressure difference drives a rapid mass transfer to the condensing end where the excess vapour condenses, releases its latent heat, and warms the cool end of the pipe. Non-condensing gases (caused by contamination for instance) in the vapour impede the gas flow and reduce the effectiveness of the heat pipe, particularly at low temperatures, where vapour pressures are low. The velocity of molecules in a gas is approximately the speed of sound and in the absence of non condensing gases, this is the upper velocity with which they could travel in the heat pipe. In practice, the speed of the vapour through the heat pipe is dependent on the rate of condensation at the cold end.

The condensed working fluid then flows back to the hot end of the pipe. In the case of vertically-oriented heat pipes the fluid may be moved by the force of gravity. In the case of heat pipes containing wicks, the fluid is returned by capillary action.

When making heat pipes, there is no need to create a vacuum in the pipe. One simply boils the working fluid in the heat pipe until the resulting vapour has purged the non condensing gases from the pipe and then seals the end.

An interesting property of heat pipes is the temperature over which they are effective. Initially, it might be suspected that a water charged heat pipe would only work when the hot end reached the boiling point (100 °C) and steam was transferred to the cold end. However, the boiling point of water is dependent on absolute pressure inside the pipe. In an evacuated pipe, water will boil just slightly above its melting point (0 °C). The heat pipe will operate, therefore, when the hot end is just slightly warmer than the melting point of the working fluid. Similarly, a heat pipe with water as a working fluid can work well above the boiling point (100 °C), if the cold end is low enough in temperature to condense the fluid.

The main reason for the effectiveness of heat pipes is the evaporation and condensation of the working fluid. The heat of vaporization greatly exceeds the sensible heat capacity. Using water as an example, the energy needed to evaporate one gram of water is equivalent to the amount of energy needed to raise the temperature of that same gram of water by 540 °C (hypothetically, if the water was under extremely high pressure so it didn't vaporize or freeze over this temperature range). Almost all of that energy is rapidly transferred to the "cold" end when the fluid condenses there, making a very effective heat transfer system with no moving parts.

Origins and research in the United States

The general principle of heat pipes using gravity (commonly classified as two phase thermosiphons) dates back to the steam age. The modern concept for a capillary driven heat pipe was first suggested by R.S. Gaugler of General Motors in 1942 who patented the idea. The benefits of employing capillary action were independently developed and first demonstrated by George Grover at Los Alamos National Laboratory in 1963 and subsequently published in the Journal of Applied Physics in 1964. Grover noted in his notebook:

"Heat transfer via capillary movement of fluids. The "pumping" action of surface tension forces may be sufficient to move liquids from a cold temperature zone to a high temperature zone (with subsequent return in vapor form using as the driving force, the difference in vapor pressure at the two temperatures) to be of interest in transferring heat from the hot to the cold zone. Such a closed system, requiring no external pumps, may be of particular interest in space reactors in moving heat from the reactor core to a radiating system. In the absence of gravity, the forces must only be such as to overcome the capillary and the drag of the returning vapor through its channels."

Between 1964 and 1966, RCA was the first corporation to undertake research and development of heat pipes for commercial applications (though their work was mostly funded by the US government). During the late 1960s NASA played a large role in heat pipe development by funding a significant amount of research on their applications and reliability in space flight following from Grover's suggestion. NASA's attraction to heat pipe cooling systems was understandable given their low weight, high heat flux, and zero power draw. Their primary interest however was based on the fact that the system wouldn't be adversely affected by operating in a zero gravity environment. The first application of heat pipes in the space program was in thermal equilibration of satellite transponders. As satellites orbit, one side is exposed to the direct radiation of the sun while the opposite side is completely dark and exposed to the deep cold of outer space. This causes severe discrepancies in the temperature (and thus reliability and accuracy) of the transponders. The heat pipe cooling system designed for this purpose managed the high heat fluxes and demonstrated flawless operation with and without the influence of gravity. The developed cooling system was the first description and usage of variable conductance heat pipes to actively regulate heat flow or evaporator temperature.

Corporate R&D

Publications in 1967 and 1968 by Feldman, Eastman, & Katzoff first discussed applications of heat pipes to areas outside of government concern and that did not fall under the high temperature classification such as: air conditioning, engine cooling, and electronics cooling. These papers also made the first mentions of flexible, arterial, and flat plate heat pipes. 1969 publications introduced the concepts of the rotational heat pipe with its applications to turbine blade cooling and the first discussions of heat pipe applications to cryogenic processes.

Starting in the 1980s Sony began incorporating heat pipes into the cooling schemes for some of its commercial electronic products in place of both forced convection and passive finned heat sinks. Initially they were used in tuners & amplifiers, soon spreading to other high heat flux electronics applications. During the late 1990s increasingly hot microcomputer CPUs spurred a threefold increase in the number of U.S. heat pipe patent applications. As heat pipes transferred from a specialized industrial heat transfer component to a consumer commodity most development and production moved from the U.S. to Asia. Modern CPU heat pipes are typically made from copper and use water as the working fluid.

Applications



Alaska pipeline support legs cooled by heat pipes to keep permafrost frozen.

Grover and his colleagues were working on cooling systems for nuclear power cells for space craft, where extreme thermal conditions are found. Heat pipes have since been used extensively in spacecraft as a means for managing internal temperature conditions.

Heat pipes are extensively used in many modern computer systems, where increased power requirements and subsequent increases in heat emission have resulted in greater demands on cooling systems. Heat pipes are typically used to move heat away from components such as CPUs and GPUs to heat sinks where thermal energy may be dissipated into the environment.

Solar Thermal

Heat pipes are also being widely used in solar thermal water heating applications in combination with evacuated tube solar collector arrays. In these applications, distilled water is commonly used as the heat transfer fluid inside a sealed length of copper tubing that is located within an evacuated glass tube and oriented towards the sun.

In solar thermal water heating applications, an evacuated tube collector can deliver up to 40% more efficiency compared to more traditional "flat plate" solar water heaters. Evacuated tube collectors eliminate the need for anti-freeze additives to be added as the vacuum helps prevent heat loss. These types of solar thermal water heaters are frost protected down to more than -3 °C and are being used in Antarctica to heat water.

Pipelines over permafrost

Heat pipes are used to dissipate heat on the Trans-Alaska Pipeline System. Without them residual ground heat remaining in the oil, as well as that produced by friction and turbulence in the moving oil would conduct down the pipe's support legs. This would likely melt the permafrost on which the supports are anchored. This would cause the pipeline to sink and possibly sustain damage. To prevent this each vertical support member has been mounted with 4 vertical heat pipes.

Cooking

Heat pipes have been designed to speed the cooking of roasts. The pipe is poked through the roast. One end of the pipe extends into the oven where it draws heat to the middle of the roast.

Ventilation heat recovery

In heating, ventilation and air-conditioning systems, HVAC, heat pipes are positioned within the supply and exhaust air streams of an air handling system, or in the exhaust gases of an industrial process, in order to recover the heat energy.

The device consists of a battery of multi-row finned heat pipe tubes located within both the supply and exhaust air streams. Within the exhaust air side of the heat pipe, the

refrigerant evaporates, taking its heat from the extract air. The refrigerant vapour moves towards the cooler end of the tube, within the supply air side of the device, where it condenses and gives up its heat. The condensed refrigerant returns by a combination of gravity and capillary action in the wick. Thus heat is transferred from the exhaust air stream through the tube wall to the refrigerant, and then from the refrigerant through the tube wall to the supply air stream.

Because of the characteristics of the device, better efficiencies are obtained when the unit is positioned upright with the supply air side mounted over the exhaust air side, this allows the liquid refrigerant to flow quickly under gravity back to the evaporator. Generally, gross heat transfer efficiencies of up to 75% are claimed by manufacturers.

Limitations

Heat pipes must be tuned to particular cooling conditions. The choice of pipe material, size and coolant all have an effect on the optimal temperatures in which heat pipes work.

When heated above a certain temperature, all of the working fluid in the heat pipe will vaporize and the condensation process will cease to occur; in such conditions, the heat pipe's thermal conductivity is effectively reduced to the heat conduction properties of its solid metal casing alone. As most heat pipes are constructed of copper (a metal with high heat conductivity), an overheated heatpipe will generally continue to conduct heat at around 1/80 of the original conductivity.

In addition, below a certain temperature, the working fluid will not undergo phase change, and the thermal conductivity will be reduced to that of the solid metal casing. One of the key criteria for the selection of a working fluid is the desired operational temperature range of the application. The lower temperature limit typically occurs a few degrees above the freezing point of the working fluid.

Most manufacturers cannot make a traditional heat pipe smaller than 3mm in diameter due to material limitations (though 1.6mm thin sheets can be fabricated). Experiments have been conducted with micro heat pipes, which use piping with sharp edges, such as triangular or rhombus-like tubing. In these cases, the sharp edges transfer the fluid through capillary action, and no wick is necessary.

Chapter-7

Magnetic Sail

A **magnetic sail** or **magsail** is a proposed method of spacecraft propulsion which would use a static magnetic field to deflect charged particles radiated by the Sun as a plasma wind, and thus impart momentum to accelerate the spacecraft. A magnetic sail could also thrust directly against planetary and solar magnetospheres.

Principles of operation and design

The solar wind is a tenuous stream of plasma that flows outwards from the Sun: near the Earth's orbit, it contains several million protons and electrons per cubic meter and flows at 400 to 600 kilometres per second (250 to 370 mi/s). The magnetic sail introduces a magnetic field into this plasma flow, perpendicular to the motion of the charged particles, which can deflect the particles from their original trajectory: the momentum of the particles is then transferred to the sail, leading to a thrust on the sail. One advantage of magnetic or solar sails over (chemical or ion) reaction thrusters is that no reaction mass is depleted or carried in the craft.

In typical magnetic sail designs, the magnetic field is generated by a loop of superconducting wire. Because loops of current-carrying conductors tend to be forced outwards towards a circular shape by their own magnetic field, the sail could be deployed simply by unspooling the conductor and applying a current through it.

For a sail in the solar wind at 1 AU away from the Sun, the field strength required to resist the dynamic pressure of the solar wind is 50 nT (Template:Convert/nT). Zubrin's proposed magnetic sail design would create a bubble of space of 100 km in diameter (62 mi) where solar-wind ions are substantially deflected using a hoop 50 km (31 mi) in radius. The minimum weight of such a coil is constrained by material strength limitations at roughly 40 tonnes and it would generate 70 newtons (16 lbf) of thrust, giving a mass/thrust ratio of 600 kg/N. It is not clear how such a coil would be cooled.

The operation of magnetic sails using plasma wind is analogous to the operation of solar sails using the radiation pressure of photons emitted by the Sun. Although solar wind particles have rest mass and photons do not, sunlight has thousands of times more

momentum than the solar wind. Therefore, a magnetic sail must deflect a proportionally larger area of the solar wind than a comparable solar sail to generate the same amount of thrust. However, it need not be as massive as a solar sail because the solar wind is deflected by a magnetic field instead of a large physical sail. Conventional materials for solar sails weigh around 7 grams per square metre (0.0014 lb/sq ft), giving a thrust of $1\text{e-}5\text{ N/m}^2$ at 1 AU. This gives a mass/thrust ratio of at least 700 kg/N, similar to a magnetic sail, neglecting other structural components.

The solar and magnetic sails have a thrust that falls off as the square of the distance from the Sun.

When close to a planet with a strong magnetosphere, e.g. Earth or a gas giant, the magsail could generate more thrust by interacting with the magnetosphere instead of the solar wind, and may therefore be more efficient.

Mini-magnetospheric plasma propulsion

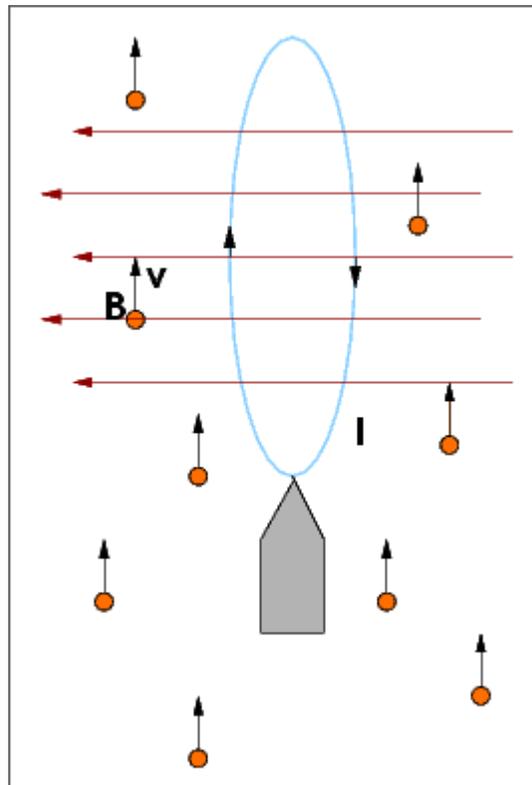
In order to reduce the size and weight of the magnet of the magnetic sail, it may be possible to *inflate* the magnetic field using a plasma in the same way that the plasma around the earth stretches out the Earth's magnetic field in the magnetosphere. In this approach, called **mini-magnetospheric plasma propulsion** (M2P2), currents running through the plasma augment and partially replace the currents in the coil. This is expected to be especially useful far from the Sun, where the increased effective size of a M2P2 sail compensates for the reduced dynamic pressure of the solar wind. The original NASA design proposes a spacecraft containing a can-shaped electromagnet into which a plasma is injected. The plasma pressure stretches the magnetic field and inflates a bubble of plasma around the spacecraft. The current in the plasma in this case augments and partially replaces currents in the coils. The plasma then generates a kind of miniaturized magnetosphere around the spacecraft, analogous to the magnetosphere that surrounds the earth. The protons and electrons which make up the solar wind are deflected by this magnetosphere and the reaction accelerates the spacecraft. The thrust of the M2P2 device would be steerable to some extent, potentially allowing the spacecraft to 'tack' into the solar wind and allowing efficient changes of orbit.

In the case of the (M2P2) system the spacecraft releases gas to create the plasma needed to maintain the somewhat leaky plasma bubble. The M2P2 system therefore has an effective *specific impulse* which is the amount of gas consumed per newton of thrust. This is a figure of merit usually used for rockets, where the fuel is actually reaction mass. Robert Winglee, who originally proposed the M2P2 technique, calculates a *specific impulse* of 200 kN·s/kg (roughly 50 times better than the space shuttle main engine). These calculations suggest that the system requires on the order of a kilowatt of power per newton of thrust, considerably lower than electric thrusters, and that the system generates the same thrust anywhere within the heliopause because the sail spreads automatically as the solar wind becomes less dense. However, this technique is less well understood than the simpler magnetic sail and issues of how large and heavy the

magnetic coil would have to be or whether the momentum from the solar wind can be efficiently transferred to the spacecraft are under dispute.

The expansion of the magnetic field using plasma injected has been successfully tested in a large vacuum chamber on Earth, but the development of thrust was not part of the experiment. A beam-powered variant, MagBeam, is also under development.

Modes of operation



A magnetic sail in a wind of charged particles. The sail generates a magnetic field, represented by red arrows, which deflects the particles into the page. The force on the sail is out of the page.

In a plasma wind

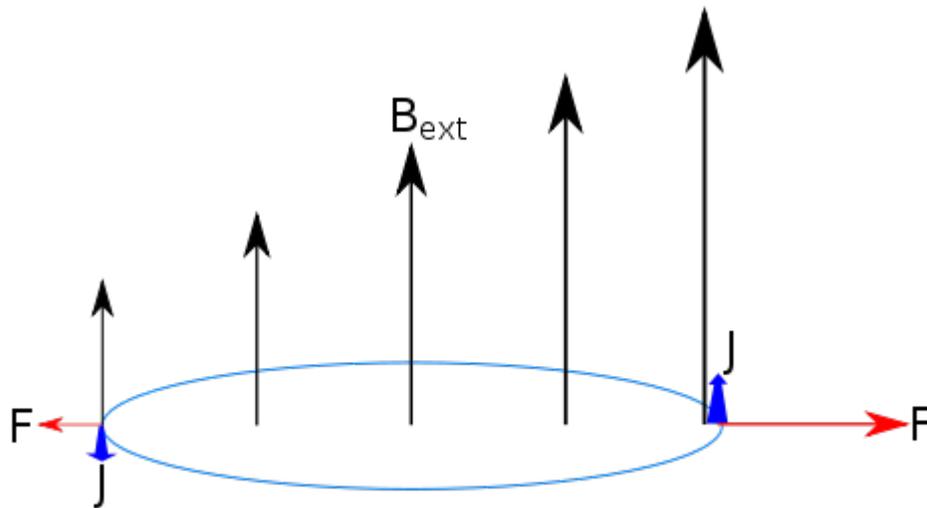
When operating away from planetary magnetospheres, a magnetic sail would force the positively charged protons of the solar wind to curve as they passed through the magnetic field. The change of momentum of the protons would thrust against the magnetic field, and thus against the field coil.

Just as with solar sails, magnetic sails can "tack." If a magnetic sail orients at an angle relative to the solar wind, charged particles are deflected preferentially to one side and the magnetic sail is pushed laterally. This means that magnetic sails could maneuver to most orbits.

In this mode, the amount of thrust generated by a magnetic sail falls off with the square of its distance from the Sun as the flux density of charged particles reduces. Solar weather also has major effects on the sail. It is possible that the plasma eruption from a severe solar flare could damage an efficient, fragile sail.

A common misconception is that a magnetic sail cannot exceed the speed of the plasma pushing it. As the speed of a magnetic sail increases, its acceleration becomes more dependent on its ability to tack efficiently. At high speeds, the plasma wind's direction will seem to come increasingly from the front of the spacecraft. Advanced sailing spacecraft might deploy field coils as "keels," so the spacecraft could use the difference in vector between the solar magnetic field and the solar wind, much as sailing yachts do.

Inside a planetary magnetosphere



A magnetic sail in a spatially-varying magnetic field. Because the vertical external field B_{ext} is stronger on one side than the other, the leftward force on the left side of the ring is smaller than the rightward force on the right side of the ring, and the net force on the sail is to the right.

Inside a planetary magnetosphere, a magnetic sail can thrust against a planet's magnetic field, especially in an orbit that passes over the planet's magnetic poles, in a similar manner to an electrodynamic tether.

The range of maneuvers available to a magnetic sail inside a planetary magnetosphere are more limited than in a plasma wind. Just as with the more familiar small-scale magnets used on Earth, a magnetic sail can only be attracted towards the magnetosphere's poles or repelled from them, depending on its orientation.

When the magnetic sail's field is oriented in the opposite direction to the magnetosphere it experiences a force inward and toward the nearest pole, and when it is oriented in the same direction as the magnetosphere it experiences the opposite effect. A magnetic sail

oriented in the same direction as the magnetosphere is not stable, and will have to prevent itself from being flipped over to the opposite orientation by some other means.

The thrust that a magnetic sail delivers within a magnetosphere decreases with the fourth power of its distance from the planet's internal magnetic dynamo.

This limited maneuvering capability is still quite useful. By varying the magnetic sail's field strength over the course of its orbit, a magnetic sail can give itself a "perigee kick" raising the altitude of its orbit's apogee.

Repeating this process with each orbit can drive the magnetic sail's apogee higher and higher, until the magnetic sail is able to leave the planetary magnetosphere and catch the solar wind. The same process in reverse can be used to lower or circularize the apogee of a magsail's orbit when it arrives at a destination planet.

In theory, it is possible for a magnetic sail to launch directly from the surface of a planet near one of its magnetic poles, repelling itself from the planet's magnetic field. However, this requires the magnetic sail to be maintained in its "unstable" orientation. A launch from Earth requires superconductors with 80 times the current density of the best known high-temperature superconductors.

Interstellar travel

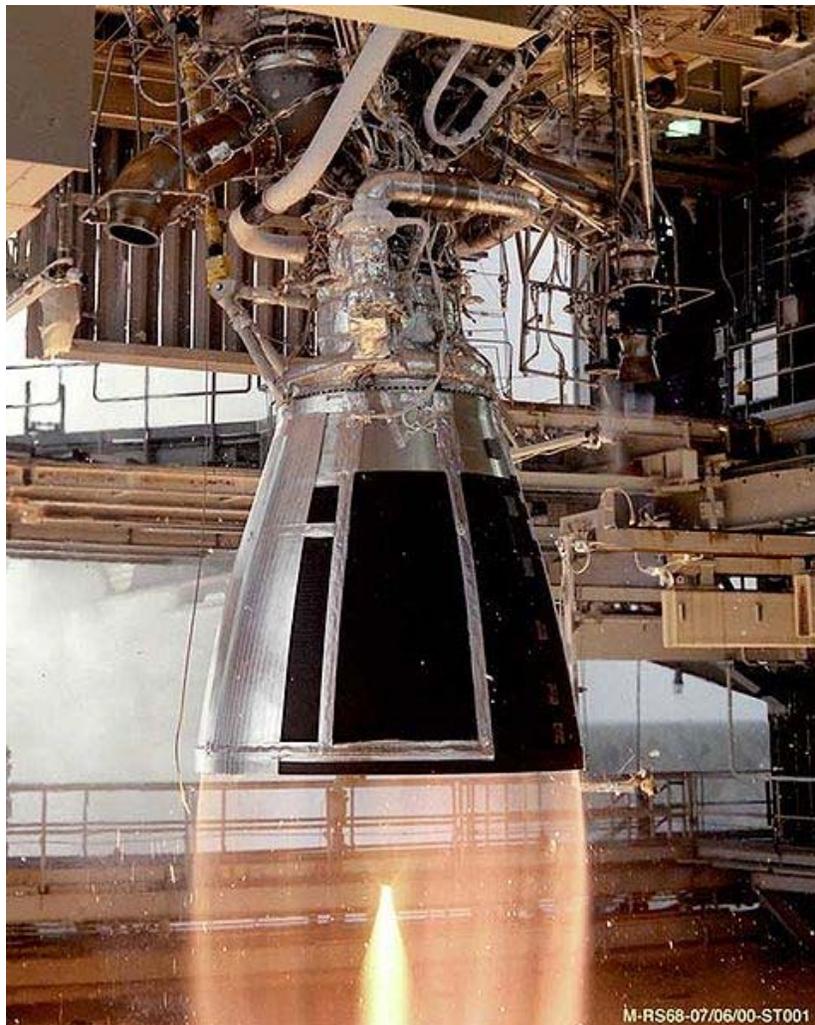
Interstellar space contains very small amounts of hydrogen. A fast-moving sail would ionize this hydrogen by accelerating the electrons in one direction and the oppositely-charged protons in the other direction. The energy for the ionization and cyclotron radiation would come from the spacecraft's kinetic energy, slowing the spacecraft. The cyclotron radiation from the acceleration of particles would be an easily detected howl in radio frequencies.

Thus, in interstellar spaceflight outside the heliopause of a star a magnetic sail could act as a parachute to decelerate a spacecraft. This removes any fuel requirements for the deceleration half of an interstellar journey, which would benefit interstellar travel enormously. The magsail was first proposed for this purpose in 1985 by Robert Zubrin and Dana Andrews, predating other uses, and evolved from a concept of the Bussard ramjet which used a magnetic scoop to collect interstellar material.

Magnetic sails could also be used with beam-powered propulsion by using a high-power particle accelerator to fire a beam of charged particles at the spacecraft. The magsail would deflect this beam, transferring momentum to the vehicle. This would provide much higher acceleration than a solar sail driven by a laser, but a charged particle beam would disperse in a shorter distance than a laser due to the electrostatic repulsion of its component particles. This dispersion problem could potentially be resolved by accelerating a stream of sails which then in turn transfer their momentum to a magsail vehicle, as proposed by Jordin Kare.

Chapter-8

Rocket Engine



RS-68 being tested at NASA's Stennis Space Center. The nearly transparent exhaust is due to this engine's exhaust being mostly superheated steam (water vapor from its propellants, hydrogen and oxygen)



Viking 5C rocket engine

A **rocket engine**, or simply "rocket," is a jet engine that uses only propellant mass for forming its high speed propulsive jet. Rocket engines are reaction engines and obtain thrust in accordance with Newton's third law. Since they need no external material to form their jet, rocket engines can be used for spacecraft propulsion as well as terrestrial uses, such as missiles. Most rocket engines are internal combustion engines, although non-combusting forms also exist.

Rocket engines as a group have the highest exhaust velocities, are by far the lightest, and are the least energy efficient of all types of jet engines.

Terminology

Chemical rockets are rockets powered by exothermic chemical reactions of the propellant.

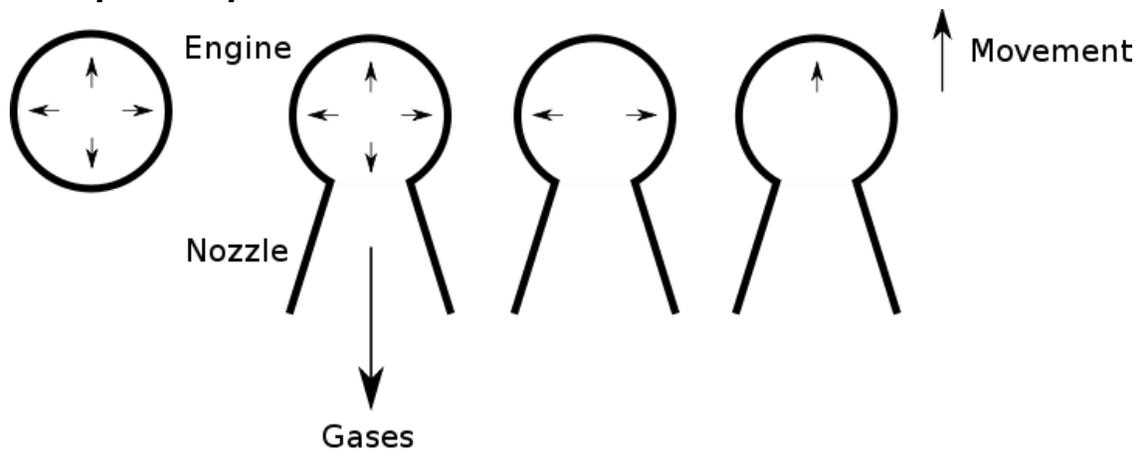
Rocket motor (or **solid-propellant rocket motor**) is a synonymous term with rocket engine that usually refers to solid rocket engines.

Liquid rockets (or **liquid-propellant rocket engine**) use one or more liquid propellants that are held in tanks prior to burning.

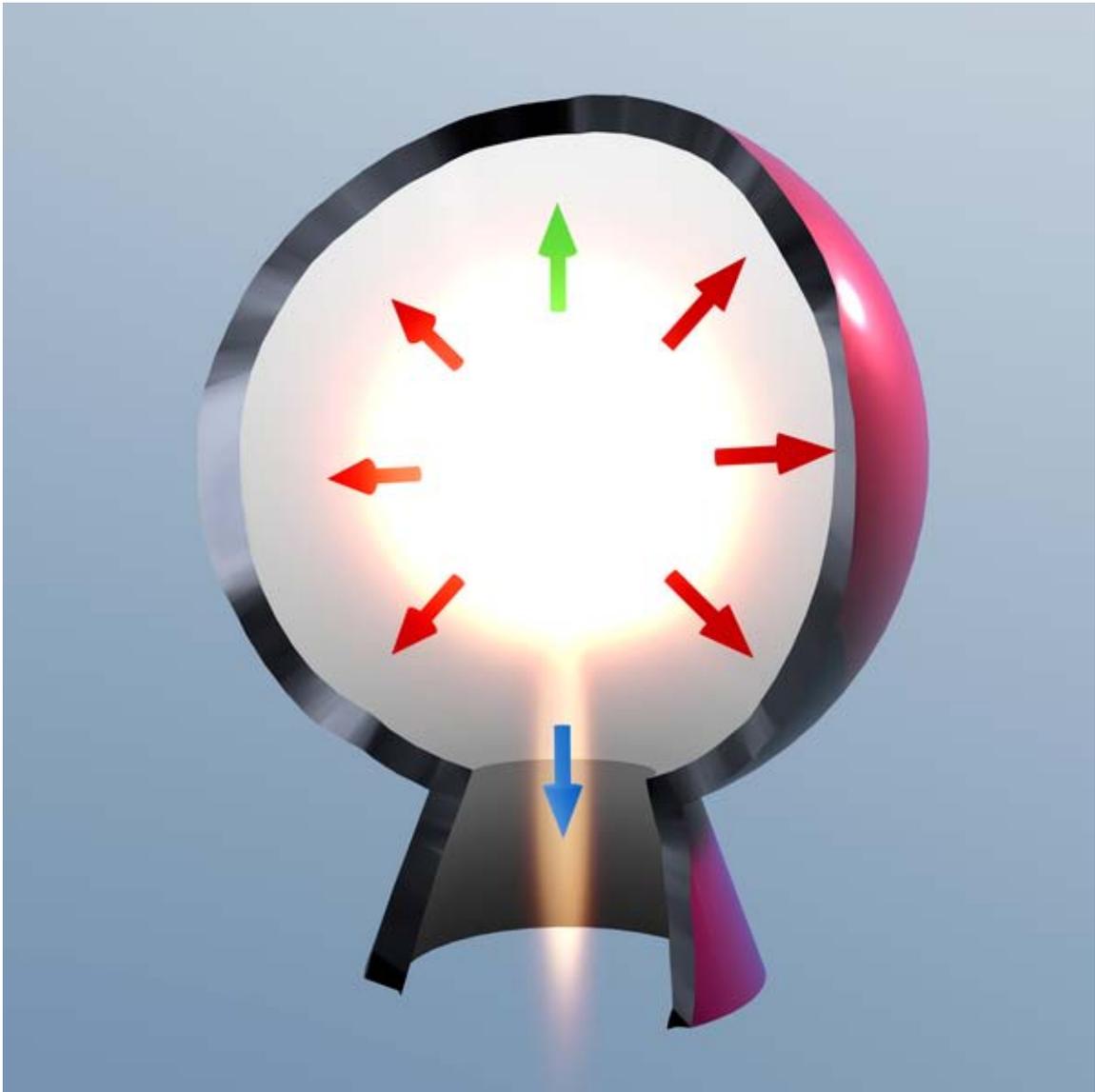
Hybrid rockets have a solid propellant in the combustion chamber and a second liquid or gas propellant is added to permit it to burn.

Thermal rockets are rockets where the propellant is inert, but is heated by a power source such as solar or nuclear power or beamed energy.

Principle of operation



How rocket engines work



Rocket engines give part of their thrust due to unopposed pressure on the combustion chamber

Rocket engines produce thrust by the expulsion of a high-speed fluid exhaust. This fluid is nearly always a gas which is created by high pressure (10-200 bar) combustion of solid or liquid propellants, consisting of fuel and oxidiser components, within a combustion chamber.

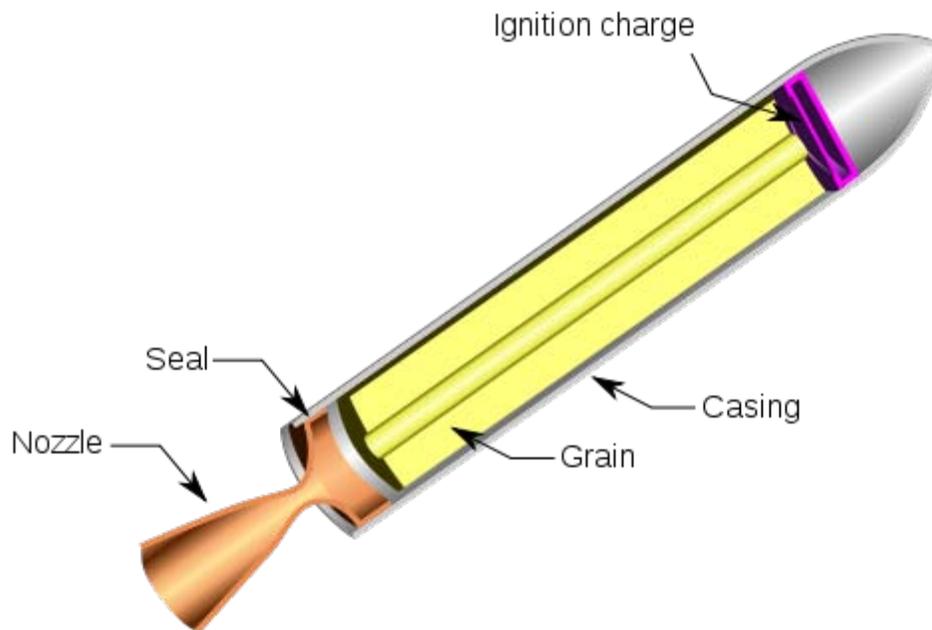
The fluid exhaust is then passed through a supersonic propelling nozzle which uses heat energy of the gas to accelerate the exhaust to very high speed, and the reaction to this pushes the engine in the opposite direction.

In rocket engines, high temperatures and pressures are highly desirable for good performance as this permits a longer nozzle to be fitted to the engine, which gives higher exhaust speeds, as well as giving better thermodynamic efficiency.

Introducing propellant into a combustion chamber

Rocket propellant is mass that is stored, usually in some form of propellant tank, prior to being ejected from a rocket engine in the form of a fluid jet to produce thrust.

Chemical rocket propellants are most commonly used, which undergo exothermic chemical reactions which produce hot gas which is used by a rocket for propulsive purposes. Alternatively, a chemically inert reaction mass can be heated using a high-energy power source via a heat exchanger, and then no combustion chamber is used.



A solid rocket motor.

Solid rocket propellants are prepared as a mixture of fuel and oxidizing components called 'grain' and the propellant storage casing effectively becomes the combustion chamber. Liquid-fueled rockets typically pump separate fuel and oxidiser components into the combustion chamber, where they mix and burn. Hybrid rocket engines use a combination of solid and liquid or gaseous propellants. Both liquid and hybrid rockets use *injectors* to introduce the propellant into the chamber. These are often an array of simple jets- holes through which the propellant escapes under pressure; but sometimes may be more complex spray nozzles. When two or more propellants are injected the jets usually deliberately collide the propellants as this breaks up the flow into smaller droplets that burn more easily.

Combustion chamber

For chemical rockets the combustion chamber is typically just a cylinder, and flame holders are rarely used. The dimensions of the cylinder are such that the propellant is able to combust thoroughly; different propellants require different combustion chamber sizes for this to occur. This leads to a number called L^* :

$$L^* = \frac{V_c}{A_t}$$

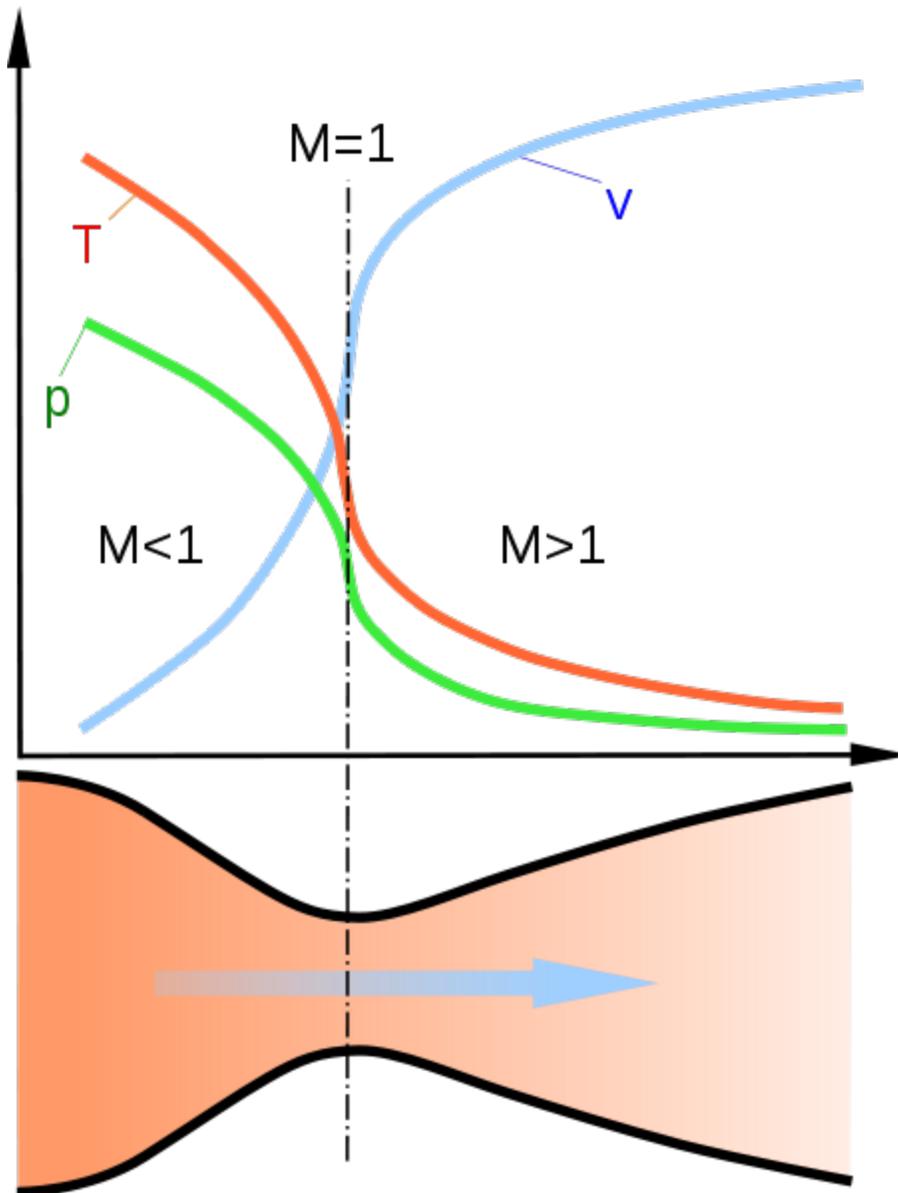
where:

- V_c is the volume of the chamber
- A_t is the area of the throat

L^* is typically in the range of 25–60 inches (0.63–1.5 m).

The combination of temperatures and pressures typically reached in a combustion chamber is usually extreme by any standards. Unlike in air-breathing jet engines, no atmospheric nitrogen is present to dilute and cool the combustion, and the temperature can reach true stoichiometric. This, in combination with the high pressures, means that the rate of heat conduction through the walls is very high.

Rocket nozzles



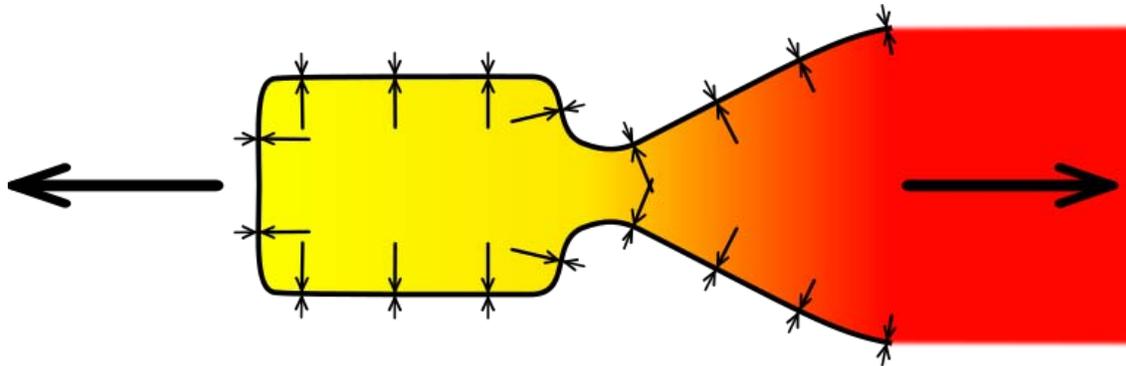
Typical temperatures (T) and pressures (p) and speeds (v) in a De Laval Nozzle

The large bell or cone shaped expansion nozzle gives a rocket engine its characteristic shape.

In rockets the hot gas produced in the combustion chamber is permitted to escape from the combustion chamber through an opening (the "throat"), within a high expansion-ratio 'de Laval' nozzle.

Provided sufficient pressure is provided to the nozzle (about 2.5-3x above ambient pressure) the nozzle *chokes* and a supersonic jet is formed, dramatically accelerating the gas, converting most of the thermal energy into kinetic energy.

The exhaust speeds vary, depending on the expansion ratio the nozzle is designed to give, but exhaust speeds as high as ten times the speed of sound of sea level air are not uncommon.



Rocket thrust is caused by pressures acting in the combustion chamber and nozzle. From Newton's third law, equal and opposite pressures act on the exhaust, and this accelerates it to high speeds.

About half of the rocket engine's thrust comes from the unbalanced pressures inside the combustion chamber and the rest comes from the pressures acting against the inside of the nozzle. As the gas expands (adiabatically) the pressure against the nozzle's walls forces the rocket engine in one direction while accelerating the gas in the other.

Propellant efficiency

For a rocket engine to be propellant efficient, it is important that the maximum pressures possible be created on the walls of the chamber and nozzle by a specific amount of propellant; as this is the source of the thrust. This can be achieved by all of:

- heating the propellant to as high a temperature as possible (using a high energy fuel, containing hydrogen and carbon and sometimes metals such as aluminium, or even using nuclear energy)
- using a low specific density gas (as hydrogen rich as possible)
- using propellants which are, or decompose to, simple molecules with few degrees of freedom to maximise translational velocity

Since all of these things minimise the mass of the propellant used, and since pressure is proportional to the mass of propellant present to be accelerated as it pushes on the engine, and since from Newton's third law the pressure that acts on the engine also reciprocally acts on the propellant, it turns out that for any given engine the speed that the propellant leaves the chamber is unaffected by the chamber pressure (although the thrust is

proportional). However, speed is significantly affected by all three of the above factors and the exhaust speed is an excellent measure of the engine propellant efficiency. This is termed *exhaust velocity*, and after allowance is made for factors that can reduce it, the **effective exhaust velocity** is one of the most important parameters of a rocket engine (although weight, cost, ease of manufacture etc. are usually also very important).

For aerodynamic reasons the flow goes sonic ("chokes") at the narrowest part of the nozzle, the 'throat'. Since the speed of sound in gases increases with the square root of temperature, the use of hot exhaust gas greatly improves performance. By comparison, at room temperature the speed of sound in air is about 340 m/s while the speed of sound in the hot gas of a rocket engine can be over 1700 m/s; much of this performance is due to the higher temperature, but additionally rocket propellants are chosen to be of low molecular mass, and this also gives a higher velocity compared to air.

Expansion in the rocket nozzle then further multiplies the speed, typically between 1.5 and 2 times, giving a highly collimated hypersonic exhaust jet. The speed increase of a rocket nozzle is mostly determined by its area expansion ratio—the ratio of the area of the throat to the area at the exit, but detailed properties of the gas are also important. Larger ratio nozzles are more massive but are able to extract more heat from the combustion gases, increasing the exhaust velocity.

Nozzle efficiency is affected by operation in the atmosphere because atmospheric pressure changes with altitude; but due to the supersonic speeds of the gas exiting from a rocket engine, the pressure of the jet may be either below or above ambient, and equilibrium between the two is not reached at all altitudes.

Back pressure and optimal expansion

For optimal performance the pressure of the gas at the end of the nozzle should just equal the ambient pressure: if the exhaust's pressure is lower than the ambient pressure, then the vehicle will be slowed by the difference in pressure between the top of the engine and the exit; on the other hand, if the exhaust's pressure is higher, then exhaust pressure that could have been converted into thrust is not converted, and energy is wasted.

To maintain this ideal of equality between the exhaust's exit pressure and the ambient pressure, the diameter of the nozzle would need to increase with altitude, giving the pressure a longer nozzle to act on (and reducing the exit pressure and temperature). This increase is difficult to arrange in a lightweight fashion, although is routinely done with other forms of jet engines. In rocketry a lightweight compromise nozzle is generally used and some reduction in atmospheric performance occurs when used at other than the 'design altitude' or when throttled. To improve on this, various exotic nozzle designs such as the plug nozzle, stepped nozzles, the expanding nozzle and the aerospike have been proposed, each providing some way to adapt to changing ambient air pressure and each allowing the gas to expand further against the nozzle, giving extra thrust at higher altitudes.

When exhausting into a sufficiently low ambient pressure (vacuum) several issues arise. One is the sheer weight of the nozzle- beyond a certain point, for a particular vehicle, the extra weight of the nozzle outweighs any performance gained. Secondly, as the exhaust gases adiabatically expand within the nozzle they cool, and eventually some of the chemicals can freeze, producing 'snow' within the jet. This causes instabilities in the jet and must be avoided.

On a De Laval nozzle, exhaust gas flow detachment will occur in a grossly over-expanded nozzle. As the detachment point will not be uniform around the axis of the engine, a side force may be imparted to the engine. This side force may change over time and result in control problems with the launch vehicle.

Thrust vectoring

Many engines require the overall thrust to change direction over the length of the burn. A number of different ways to achieve this have been flown:

- The entire engine is mounted on a hinge or gimbal and any propellant feeds reach the engine via low pressure flexible pipes or rotary couplings.
- Just the combustion chamber and nozzle is gimbled, the pumps are fixed, and high pressure feeds attach to the engine
- multiple engines (often canted at slight angles) are deployed but throttled to give the overall vector that is required, giving only a very small penalty
- fixed engines with vernier thrusters
- high temperature vanes held in the exhaust that can be tilted to deflect the jet

Overall rocket engine performance

Rocket technology can combine very high thrust (meganewtons), very high exhaust speeds (around 10 times the speed of sound in air at sea level) and very high thrust/weight ratios (>100) *simultaneously* as well as being able to operate outside the atmosphere, and while permitting the use of low pressure and hence lightweight tanks and structure.

Rockets can be further optimised to even more extreme performance along one or more of these axes at the expense of the others.

Specific impulse

The most important metric for the efficiency of a rocket engine is impulse per unit of propellant, this is called specific impulse (usually written I_{sp}). This is either measured as a speed (the *effective exhaust velocity* V_e in metres/second or ft/s) or as a time (seconds). An engine that gives a large specific impulse is normally highly desirable.

The specific impulse that can be achieved is primarily a function of the propellant mix (and ultimately would limit the specific impulse), but practical limits on chamber pressures and the nozzle expansion ratios reduce the performance that can be achieved.

Typical performances of common propellants

Propellant mix	Vacuum Isp (seconds)	Effective exhaust velocity (m/s)
liquid oxygen/ liquid hydrogen	455	4462
liquid oxygen/ kerosene (RP-1)	358	3510
nitrogen tetroxide/ hydrazine	305	2993

Net thrust

Below is an approximate equation for calculating the net thrust of a rocket engine:

$$F_n = \dot{m} V_e = \dot{m} V_{e-act} + A_e(P_e - P_{amb})$$

where:

\dot{m} = exhaust gas mass flow

V_e = effective exhaust velocity

V_{e-act} = actual jet velocity at nozzle exit plane

A_e = flow area at nozzle exit plane (or the plane where the jet leaves the nozzle if separated flow)

P_e = static pressure at nozzle exit plane

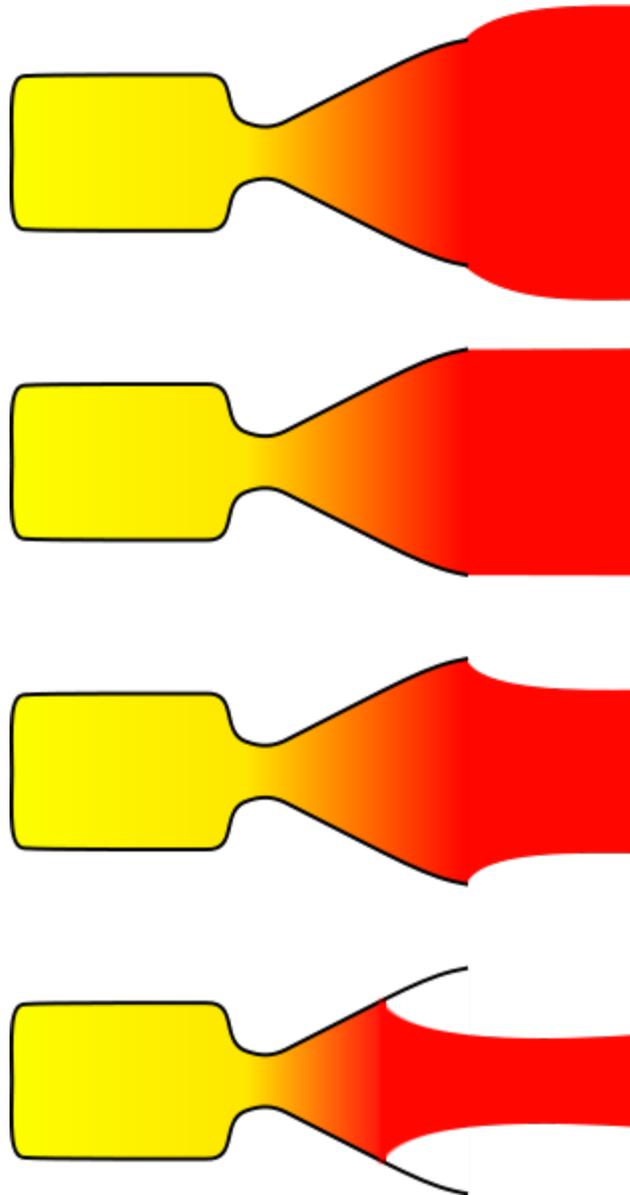
P_{amb} = ambient (or atmospheric) pressure

Since, unlike a jet engine, a conventional rocket motor lacks an air intake, there is no 'ram drag' to deduct from the gross thrust. Consequently the net thrust of a rocket motor is equal to the gross thrust (apart from static back pressure).

The $\dot{m} V_{e-act}$ term represents the momentum thrust, which remains constant at a given throttle setting, whereas the $A_e(P_e - P_{amb})$ term represents the pressure thrust term. At full throttle, the net thrust of a rocket motor improves slightly with increasing altitude, because as atmospheric pressure decreases with altitude, the pressure thrust term increases. At the surface of the Earth the pressure thrust may be reduced by up to 30%, depending on the engine design. This reduction drops roughly exponentially to zero with increasing altitude.

Maximum thrust for a rocket engine is achieved by maximizing the momentum contribution of the equation without incurring penalties from over expanding the exhaust.

This occurs when $P_e = P_{amb}$. Since ambient pressure changes with altitude, most rocket engines spend very little time operating at peak efficiency.



If the pressure of the exhaust jet varies from atmospheric pressure, nozzles can be said to be (top to bottom):

Underexpanded

Ambient

Overexpanded

Grossly overexpanded

If under or overexpanded then loss of efficiency occurs, grossly overexpanded nozzles lose less efficiency, but can cause mechanical issues with the nozzle. Rockets become progressively more underexpanded as they gain altitude. Note that almost all rocket engines will be momentarily grossly overexpanded during startup in an atmosphere.

Vacuum Isp

Due to the specific impulse varying with pressure, a quantity that is easy to compare and calculate with is useful. Because rockets choke at the throat, and because the supersonic exhaust prevents external pressure influences travelling upstream, it turns out that the pressure at the exit is ideally exactly proportional to the propellant flow \dot{m} , provided the mixture ratios and combustion efficiencies are maintained. It is thus quite usual to rearrange the above equation slightly:

$$F_{vac} = C_f \dot{m} c^*$$

and so define the *vacuum Isp* to be:

$$V_{evac} = C_f c^*$$

Where:

$$\begin{aligned} c^* &= \text{the speed of sound constant at the throat} \\ C_f &= \text{the thrust coefficient constant of the nozzle (typically about 2)} \end{aligned}$$

And hence:

$$F_n = \dot{m} V_{evac} - A_e P_{amb}$$

Throttling

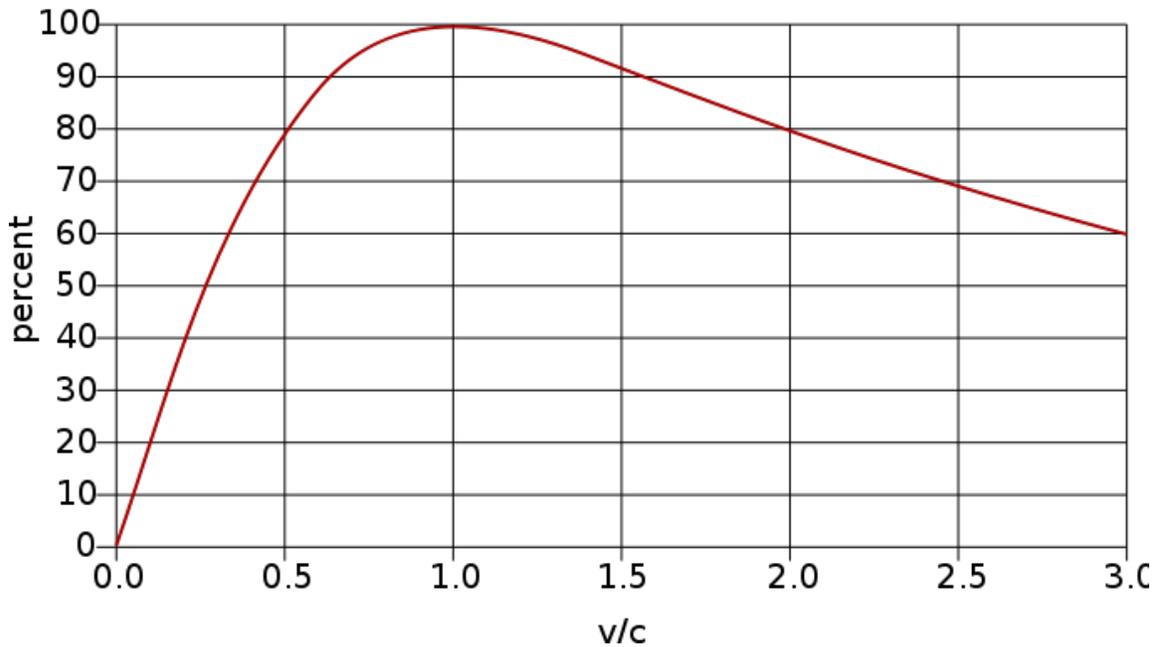
Rockets can be throttled by controlling the propellant combustion rate \dot{m} (usually measured in kg/s or lb/s). In liquid and hybrid rockets, the propellant flow entering the chamber is controlled using valves, in solid rockets it is controlled by changing the area of propellant that is burning and this can be designed into the propellant grain (and hence cannot be controlled in real-time).

Rockets can usually be throttled down to an exit pressure of about one-third of ambient pressure (often limited flow separation in nozzles) and up to a maximum limit determined only by the mechanical strength of the engine.

In practice, the degree to which rockets can be throttled varies greatly, but most rockets can be throttled by a factor of 2 without great difficulty; the typical limitation is combustion stability, as for example, injectors need a minimum pressure to avoid triggering damaging oscillations (chugging or combustion instabilities); but injectors can often be optimised and tested for wider ranges. Solid rockets can be throttled by using shaped grains that will vary their surface area over the course of the burn.

Energy efficiency

Propulsive efficiency



Rocket energy efficiency as a function of vehicle speed divided by effective exhaust speed

Rocket engine nozzles are surprisingly efficient heat engines for generating a high speed jet, as a consequence of the high combustion temperature and high compression ratio. Rocket nozzles give an excellent approximation to adiabatic expansion which is a reversible process, and hence they give efficiencies which are very close to that of the Carnot cycle. Given the temperatures reached, over 60% efficiency can be achieved with chemical rockets.

For a *vehicle* employing a rocket engine the energetic efficiency is very good if the vehicle speed approaches or somewhat exceeds the exhaust velocity (relative to launch); but at low speeds the energy efficiency goes to 0% at zero speed (as with all jet propulsion.)

Thrust to weight ratio

Rockets, of all the jet engines, indeed of essentially all engines, have the highest thrust to weight ratio. This is especially true for liquid rocket engines.

This high performance is due to the small volume of pressure vessels that make up the engine- the pumps, pipes and combustion chambers involved. The lack of inlet duct and the use of dense liquid propellant allows the pressurisation system to be small and

lightweight, whereas duct engines have to deal with air which has a density about one thousand times lower.

Jet or Rocket engine	Mass, kg	Jet or rocket thrust, kN	Thrust-to-weight ratio
RD-0410 nuclear rocket engine	2000	35.2	1.8
J-58 (SR-71 Blackbird jet engine)	2722	150	5.2
Concorde's Rolls-Royce/Snecma Olympus 593 turbojet with reheat	3175	169.2	5.4
RD-0750 rocket engine, three-propellant mode	4621	1413	31.2
RD-0146 rocket engine	260	98	38.5
Space Shuttle's SSME rocket engine	3177	2278	73.2
RD-180 rocket engine	5393	4152	78.6
F-1 (Saturn V first stage)	8391	7740.5	94.1
NK-33 rocket engine	1222	1638	136.8

Rocket thrusts are vacuum thrusts unless otherwise noted

Of the liquid propellants used, density is worst for liquid hydrogen. Although this propellant is marvellous in many ways, it has a very low density, about one fourteenth that of water. This makes the turbopumps and pipework larger and heavier, and this is reflected in the thrust-to-weight ratio of engines that use it (for example the SSME) compared to those that do not (NK-33).

Cooling

For efficiency reasons, and because they physically can, rockets run with combustion temperatures that can reach ~3500 K (~5800 °F)(~3227 °C).

Most other jet engines have gas turbines in the hot exhaust. Due to their larger surface area, they are harder to cool and hence there is a need to run the combustion processes at much lower temperatures, losing efficiency. In addition duct engines use air as an oxidant, which contains 80% largely unreactive nitrogen, which dilutes the reaction and lowers the temperatures. Rockets have none of these inherent disadvantages.

Therefore in rockets temperatures employed are very often far higher than the melting point of the nozzle and combustion chamber materials, two exceptions are graphite and tungsten (~1200 K for copper), however both are subject to oxidation if not protected. Indeed many construction materials can make perfectly acceptable propellants in their own right. It is important that these materials be prevented from combusting, melting or vaporising to the point of failure. This is sometimes somewhat facetiously termed an

'engine rich exhaust'. Materials technology could potentially place an upper limit on the exhaust temperature of chemical rockets.

Alternatively, rockets may use more common construction materials such as aluminium, steel, nickel or copper alloys and employ cooling systems that prevent the construction material itself becoming too hot. Regenerative cooling, where the propellant is passed through tubes around the combustion chamber or nozzle, and other techniques, such as curtain cooling or film cooling, are employed to give longer nozzle and chamber life. These techniques ensure that a gaseous thermal boundary layer touching the material is kept below the temperature which would cause the material to catastrophically fail.

In rockets, the heat fluxes that can pass through the wall are among the highest in engineering, fluxes are generally in the range of 1-200 MW/m². The strongest heat fluxes are found at the throat, which often sees twice that found in the associated chamber and nozzle. This is due to the combination of high speeds (which gives a very thin boundary layer), and although lower than the chamber, the high temperatures seen there.

In rockets the coolant methods include:

1. uncooled (used for short runs mainly during testing)
2. ablative walls (walls are lined with a material that is continuously vaporised and carried away).
3. radiative cooling (the chamber becomes almost white hot and radiates the heat away)
4. dump cooling (a propellant, usually hydrogen, is passed around the chamber and dumped)
5. regenerative cooling (liquid rockets use the fuel, or occasionally the oxidiser, to cool the chamber via a cooling jacket before being injected)
6. curtain cooling (propellant injection is arranged so the temperature of the gases is cooler at the walls)
7. film cooling (surfaces are wetted with liquid propellant, which cools as it evaporates)

In all cases the cooling effect that prevents the wall from being destroyed is caused by a thin layer of insulating fluid (a boundary layer) that is in contact with the walls that is far cooler than the combustion temperature. Provided this boundary layer is intact the wall will not be damaged.

Disruption of the boundary layer may occur during cooling failures or combustion instabilities, and wall failure typically occurs soon after.

With regenerative cooling a second boundary layer is found in the coolant channels around the chamber. This boundary layer thickness needs to be as small as possible, since the boundary layer acts as an insulator between the wall and the coolant. This may be achieved by making the coolant velocity in the channels as high as possible.

In practice, regenerative cooling is nearly always used in conjunction with curtain cooling and/or film cooling.

Liquid fueled engines are often run fuel rich, which results in lower temperature combustion. Cooler exhaust reduces heat loads on the engine allowing lower cost materials, a simplified cooling system, and a lower performance engine.

Mechanical issues

Rocket combustion chambers are normally operated at fairly high pressure, typically 10-200 bar (1 to 20 MPa, 150-3000 psi). When operated within significant atmospheric pressure, higher combustion chamber pressures give better performance by permitting a larger and more efficient nozzle to be fitted without it being grossly overexpanded.

However, these high pressures cause the outermost part of the chamber to be under very large hoop stresses – rocket engines are pressure vessels.

Worse, due to the high temperatures created in rocket engines the materials used tend to have a significantly lowered working tensile strength.

In addition, significant temperature gradients are set up in the walls of the chamber and nozzle, these cause differential expansion of the inner liner that create internal stresses.

Acoustic issues

In addition, the extreme vibration and acoustic environment inside a rocket motor commonly result in peak stresses well above mean values, especially in the presence of organ pipe-like resonances and gas turbulence.

Combustion instabilities

The combustion may display undesired instabilities, of sudden or periodic nature. The pressure in the injection chamber may increase until the propellant flow through the injector plate decreases; a moment later the pressure drops and the flow increases, injecting more propellant in the combustion chamber which burns a moment later, and again increases the chamber pressure, repeating the cycle. This may lead to high-amplitude pressure oscillations, often in ultrasonic range, which may damage the motor. Oscillations of ± 200 psi at 25 kHz were the cause of failures of early versions of the Titan II missile second stage engines. The other failure mode is a deflagration to detonation transition; the supersonic pressure wave formed in the combustion chamber may destroy the engine.

The combustion instabilities can be provoked by remains of cleaning solvents in the engine, reflected shock wave, initial instability after ignition, explosion near the nozzle that reflects into the combustion chamber, and many more factors. In stable engine

designs the oscillations are quickly suppressed; in unstable designs they persist for prolonged periods. Oscillation suppressors are commonly used.

Periodic variations of thrust, caused by combustion instability or longitudinal vibrations of structures between the tanks and the engines which modulate the propellant flow, are known as "pogo oscillations" or "pogo", named after the pogo stick.

Three different types of combustion instabilities occur:

Chugging

This is a low frequency oscillation at a few Hertz in chamber pressure usually caused by pressure variations in feed lines due to variations in acceleration of the vehicle. This can cause cyclic variation in thrust, and the effects can vary from merely annoying to actually damaging the payload or vehicle. Chugging can be minimised by using gas-filled damping tubes on feed lines of high density propellants.

Buzzing

This can be caused due to insufficient pressure drop across the injectors. It generally is mostly annoying, rather than being damaging. However, in extreme cases combustion can end up being forced backwards through the injectors – this can cause explosions with monopropellants.

Screeching

This is the most immediately damaging, and the hardest to control. It is due to acoustics within the combustion chamber that often couples to the chemical combustion processes that are the primary drivers of the energy release, and can lead to unstable resonant "screeching" that commonly leads to catastrophic failure due to thinning of the insulating thermal boundary layer. Such effects are very difficult to predict analytically during the design process, and have usually been addressed by expensive, time consuming and extensive testing, combined with trial and error remedial correction measures.

Screeching is often dealt with by detailed changes to injectors, or changes in the propellant chemistry, or vaporizing the propellant before injection, or use of Helmholtz dampers within the combustion chambers to change the resonant modes of the chamber.

Testing for the possibility of screeching is sometimes done by exploding small explosive charges outside the combustion chamber with a tube set tangentially to the combustion chamber near the injectors to determine the engine's impulse response and then evaluating the time response of the chamber pressure- a fast recovery indicates a stable system.

Exhaust noise

For all but the very smallest sizes, rocket exhaust compared to other engines is generally very noisy. As the hypersonic exhaust mixes with the ambient air, shock waves are formed. The Space Shuttle generates over 200 dB(A) of noise around its base.

The Saturn V launch was detectable on seismometers a considerable distance from the launch site. The sound intensity from the shock waves generated depends on the size of the rocket and on the exhaust velocity. Such shock waves seem to account for the characteristic crackling and popping sounds produced by large rocket engines when heard live. These noise peaks typically overload microphones and audio electronics, and so are generally weakened or entirely absent in recorded or broadcast audio reproductions. For large rockets at close range, the acoustic effects could actually kill.

More worryingly for space agencies, such sound levels can also damage the launch structure, or worse, be reflected back at the comparatively delicate rocket above. This is why so much water is typically used at launches. The water spray changes the acoustic qualities of the air and reduces or deflects the sound energy away from the rocket.

Generally speaking noise is most intense when a rocket is close to the ground, since the noise from the engines radiates up away from the plume, as well as reflecting off the ground. Also, when the vehicle is moving slowly, little of the chemical energy input to the engine can go into increasing the kinetic energy of the rocket (since useful power P transmitted to the vehicle is $P = F * V$ for thrust F and speed V). Then the largest portion of the energy is dissipated in the exhaust's interaction with the ambient air, producing noise. This noise can be reduced somewhat by flame trenches with roofs, by water injection around the plume and by deflecting the plume at an angle.

Testing

Rocket engines are usually statically tested at a test facility before being put into production. For high altitude engines, either a shorter nozzle must be used, or the rocket must be tested in a large vacuum chamber.

Safety

Rockets have a reputation for unreliability and danger; especially catastrophic failures. Contrary to this reputation, carefully designed rockets can be made arbitrarily reliable. In military use, rockets are not unreliable. However, one of the main non-military uses of rockets is for orbital launch. In this application, the premium is on minimum weight, and it is difficult to achieve high reliability and low weight simultaneously. In addition, if the number of flights launched is low, there is a very high chance of a design, operations or manufacturing error causing destruction of the vehicle. Essentially all launch vehicles are test vehicles by normal aerospace standards (as of 2006).

The X-15 rocket plane achieved a 0.5% failure rate, with a single catastrophic failure during ground test, and the SSME has managed to avoid catastrophic failures in over 350 engine-flights.

Chemistry

Rocket propellants require a high specific energy (energy per unit mass), because ideally all the reaction energy appears as kinetic energy of the exhaust gases, and exhaust velocity is the single most important performance parameter of an engine, on which vehicle performance depends.

Aside from inevitable losses and imperfections in the engine, incomplete combustion, etc., after specific reaction energy, the main theoretical limit reducing the exhaust velocity obtained is that, according to the laws of thermodynamics, a fraction of the chemical energy may go into rotation of the exhaust molecules, where it is unavailable for producing thrust. Monatomic gases like helium have only three degrees of freedom, corresponding to the three dimensions of space, $\{x,y,z\}$, and only such spherically symmetric molecules escape this kind of loss. A diatomic molecule like H_2 can rotate about either of the two axes perpendicular to the one joining the two atoms, and as the equipartition law of statistical mechanics demands that the available thermal energy be divided equally among the degrees of freedom, for such a gas in thermal equilibrium $3/5$ of the energy can go into unidirectional motion, and $2/5$ into rotation. A triatomic molecule like water has six degrees of freedom, so the energy is divided equally among rotational and translational degrees of freedom. For most chemical reactions the latter situation is the case. This issue is traditionally described in terms of the ratio, gamma, of the specific heat of the gas at constant volume to that at constant pressure. The rotational energy loss is largely recovered in practice if the expansion nozzle is large enough to allow the gases to expand and cool sufficiently, the function of the nozzle being to convert the random thermal motions of the molecules in the combustion chamber into the unidirectional translation that produces thrust. As long as the exhaust gas remains in equilibrium as it expands, the initial rotational energy will be largely returned to translation in the nozzle.

Although the specific reaction energy per unit mass of reactants is key, low mean molecular weight in the reaction products is also important in practice in determining exhaust velocity. This is because the high gas temperatures in rocket engines pose serious problems for the engineering of survivable motors. Because temperature is proportional to the mean *energy per molecule*, a given amount of energy distributed among more molecules of lower mass permits a higher exhaust velocity at a given temperature. This means low atomic mass elements are favoured. Liquid hydrogen (LH2) and oxygen (LOX, or LO2), are the most effective propellants in terms of exhaust velocity that have been widely used to date, though a few exotic combinations involving boron or liquid ozone are potentially somewhat better in theory if various practical problems could be solved.

It is important to note in computing the specific reaction energy, that the *entire mass of the propellants, including both fuel and oxidizer*, must be included. The fact that air-breathing engines are typically able to obtain oxygen "for free" without having to carry it along, accounts for one factor of why air-breathing engines are very much more propellant-mass efficient, and one reason that rocket engines are far less suitable for most ordinary terrestrial applications. Fuels for automobile or turbojet engines, utilize atmospheric oxygen and so have a much better effective energy output per unit mass of propellant that must be carried, but are similar per unit mass of fuel.

Computer programs that predict the performance of propellants in rocket engines are available.

Ignition

With liquid and hybrid rockets, immediate ignition of the propellant(s) as they first enter the combustion chamber is essential.

With liquid propellants (but not gaseous), failure to ignite within milliseconds usually causes too much liquid propellant to be within the chamber, and if/when ignition occurs the amount of hot gas created will often exceed the maximum design pressure of the chamber. The pressure vessel will often fail catastrophically. This is sometimes called a *hard start*.

Ignition can be achieved by a number of different methods; a pyrotechnic charge can be used, a plasma torch can be used, or electric spark plugs may be employed. Some fuel/oxidizer combinations ignite on contact (hypergolic), and non-hypergolic fuels can be "chemically ignited" by priming the fuel lines with hypergolic propellants (popular in Russian engines).

Gaseous propellants generally will not cause hard starts, with rockets the total injector area is less than the throat thus the chamber pressure tends to ambient prior to ignition and high pressures cannot form even if the entire chamber is full of flammable gas at ignition.

Solid propellants are usually ignited with one-shot pyrotechnic devices.

Once ignited, rocket chambers are self sustaining and igniters are not needed. Indeed chambers often spontaneously reignite if they are restarted after being shut down for a few seconds. However, when cooled, many rockets cannot be restarted without at least minor maintenance, such as replacement of the pyrotechnic igniter.

Plume physics



Armadillo aerospace's quad vehicle showing visible banding (shock diamonds) in the exhaust plume

Rocket plume varies depending on the rocket engine, design altitude, altitude, thrust and other factors.

Carbon rich exhausts from kerosene fuels are often orange in colour due to the black body radiation of the unburned particles, in addition to the blue Swan bands. Peroxide oxidiser based rockets and hydrogen rocket plumes contain largely steam and are nearly invisible to the naked eye but shine brightly in the ultraviolet and infrared. Plumes from solid rockets can be highly visible as the propellant frequently contains metals such as elemental aluminium which burns with an orange-white flame and adds energy to the combustion process.

Some exhausts, notably alcohol fuelled rockets, can show visible shock diamonds. These are due to cyclic variations in the plume pressure relative to ambient creating shock waves that form 'mach disks'.

The shape of the plume varies from the design altitude, at high altitude all rockets are grossly under-expanded, and a quite small percentage of exhaust gases actually end up expanding forwards.

Types of rocket engines

Physically powered

Type	Description	Advantages	Disadvantages
water rocket	Partially filled pressurised carbonated drinks container with tail and nose weighting	Very simple to build	Altitude typically limited to a few hundred feet or so (world record is 623 meters/2044 feet)
cold gas thruster	A non combusting form, used for vernier thrusters	Non contaminating exhaust	Extremely low performance
hot water rocket	Hot water is stored in a tank at high temperature/pressure and turns to steam in nozzle	Simple, fairly safe, under 200 seconds Isp	Low overall performance due to heavy tank

Chemically powered

Type	Description	Advantages	Disadvantages
Solid rocket	Ignitable, self sustaining solid fuel/oxidiser mixture ("grain") with central hole and nozzle	Simple, often no moving parts, reasonably good mass fraction, reasonable I_{sp} . A thrust schedule can be designed into the grain.	Once lit, extinguishing it is difficult although often possible, cannot be throttled in real time; handling issues from ignitable mixture, lower performance than liquid rockets, if grain cracks it can block nozzle with disastrous results, cracks burn and widen during burn. Refuelling grain harder than simply filling tanks, Lower specific Impulse than Liquid Rockets.
Hybrid rocket	Separate oxidiser/fuel, typically oxidiser is liquid and kept in a tank, the other solid with central hole	Quite simple, solid fuel is essentially inert without oxidiser, safer; cracks do not escalate, throttleable and easy to switch off.	Some oxidisers are monopropellants, can explode in own right; mechanical failure of solid propellant can block nozzle (very rare with rubberised propellant), central hole widens over burn and negatively affects mixture ratio.

Monopropellant rocket	Propellant such as Hydrazine, Hydrogen Peroxide or Nitrous Oxide, flows over catalyst and exothermically decomposes and hot gases are emitted through nozzle	Simple in concept, throttleable, low temperatures in combustion chamber	catalysts can be easily contaminated, monopropellants can detonate if contaminated or provoked, I_{sp} is perhaps 1/3 of best liquids
Liquid Bipropellant rocket	Two fluid (typically liquid) propellants are introduced through injectors into combustion chamber and burnt	Up to ~99% efficient combustion with excellent mixture control, throttleable, can be used with turbopumps which permits incredibly lightweight tanks, can be safe with extreme care	Pumps needed for high performance are expensive to design, huge thermal fluxes across combustion chamber wall can impact reuse, failure modes include major explosions, a lot of plumbing is needed.
Dual mode propulsion rocket	Rocket takes off as a bipropellant rocket, then turns to using just one propellant as a monopropellant	Simplicity and ease of control	Lower performance than bipropellants
Tripopellant rocket	Three different propellants (usually hydrogen, hydrocarbon and liquid oxygen) are introduced into a combustion chamber in variable mixture ratios, or multiple engines are used with fixed propellant mixture ratios and throttled or shut down	Reduces take-off weight, since hydrogen is lighter; combines good thrust to weight with high average I_{sp} , improves payload for launching from Earth by a sizeable percentage	Similar issues to bipropellant, but with more plumbing, more R&D
Air-augmented rocket	Essentially a ramjet where intake air is compressed and burnt with the exhaust from a rocket	Mach 0 to Mach 4.5+ (can also run exoatmospheric), good efficiency at Mach 2 to 4	Similar efficiency to rockets at low speed or exoatmospheric, inlet difficulties, a relatively undeveloped and unexplored type, cooling difficulties, very noisy,

			thrust/weight ratio is similar to ramjets.
Turborocket	A combined cycle turbojet/rocket where an additional oxidizer such as oxygen is added to the airstream to increase maximum altitude	Very close to existing designs, operates in very high altitude, wide range of altitude and airspeed	Atmospheric airspeed limited to same range as turbojet engine, carrying oxidizer like LOX can be dangerous. Much heavier than simple rockets.
Precooled jet engine / LACE (combined cycle with rocket)	Intake air is chilled to very low temperatures at inlet before passing through a ramjet or turbojet engine. Can be combined with a rocket engine for orbital insertion.	Easily tested on ground. High thrust/weight ratios are possible (~14) together with good fuel efficiency over a wide range of airspeeds, mach 0-5.5+; this combination of efficiencies may permit launching to orbit, single stage, or very rapid intercontinental travel.	Exists only at the lab prototyping stage. Examples include RB545, SABRE, ATREX

Electrically powered

Type	Description	Advantages	Disadvantages
Resistojet rocket (electric heating)	A monopropellant is electrically heated by a filament for extra performance	Higher I_{sp} than monopropellant alone, about 40% higher.	Uses a lot of power and hence gives typically low thrust
Arcjet rocket (chemical burning aided by electrical discharge)	Similar to resistojet in concept but with inert propellant, except an arc is used which allows higher temperatures	1600 seconds I_{sp}	Very low thrust and high power, performance is similar to Ion drive.
Pulsed plasma thruster (electric arc heating; emits plasma)	Plasma is used to erode a solid propellant	High I_{sp} , can be pulsed on and off for attitude control	Low energetic efficiency
Variable specific impulse magnetoplasma	Microwave heated plasma with magnetic	Variable I_{sp} from 1000 seconds to 10,000 seconds	similar thrust/weight ratio with ion drives (worse), thermal issues, as with ion

rocket throat/nozzle

drives very high power requirements for significant thrust, really needs advanced nuclear reactors, never flown, requires low temperatures for superconductors to work

Solar powered

The Solar thermal rocket would make use of solar power to directly heat reaction mass, and therefore does not require an electrical generator as most other forms of solar-powered propulsion do. A solar thermal rocket only has to carry the means of capturing solar energy, such as concentrators and mirrors. The heated propellant is fed through a conventional rocket nozzle to produce thrust. The engine thrust is directly related to the surface area of the solar collector and to the local intensity of the solar radiation and inversely proportional to the I_{sp} .

Type	Description	Advantages	Disadvantages
Solar thermal rocket	Propellant is heated by solar collector	Simple design. Using hydrogen propellant, 900 seconds of I_{sp} is comparable to Nuclear Thermal rocket, without the problems and complexity of controlling a fission reaction. Ability to productively utilize waste gaseous hydrogen—an inevitable byproduct of long-term liquid hydrogen storage in the radiative heat environment of space—for both orbital stationkeeping and attitude control.	Only useful once in space, as thrust is fairly low, but hydrogen has not been traditionally thought to be easily stored in space, otherwise moderate/low I_{sp} if higher-molecular-mass propellants are used. Using higher-molecular-weight propellants, for example water, lowers performance.

Beam powered

Type	Description	Advantages	Disadvantages
light beam powered rocket	Propellant is heated by light beam (often laser) aimed at vehicle from a distance, either directly or	simple in principle, in principle very high exhaust speeds can be achieved	~1 MW of power per kg of payload is needed to achieve orbit, relatively high accelerations, lasers are blocked by clouds, fog, reflected laser light may be dangerous, pretty much needs hydrogen

	indirectly via heat exchanger		monopropellant for good performance which needs heavy tankage, some designs are limited to ~600 seconds due to reemission of light since propellant/heat exchanger gets white hot
microwave beam powered rocket	Propellant is heated by microwave beam aimed at vehicle from a distance	microwaves avoid reemission of energy, so ~900 seconds exhaust speeds might be achievable	~1 MW of power per kg of payload is needed to achieve orbit, relatively high accelerations, microwaves are absorbed to a degree by rain, reflected microwaves may be dangerous, pretty much needs hydrogen monopropellant for good performance which needs heavy tankage, transmitter diameter is measured in kilometres to achieve a fine enough beam to hit a vehicle at up to 100 km.

Nuclear powered

Nuclear propulsion includes a wide variety of propulsion methods that use some form of nuclear reaction as their primary power source. Various types of nuclear propulsion have been proposed, and some of them tested, for spacecraft applications:

Type	Description	Advantages	Disadvantages
Radioisotope rocket/"Poodle thruster" (radioactive decay energy)	Heat from radioactive decay is used to heat hydrogen	about 700–800 seconds, almost no moving parts	low thrust/weight ratio.
Nuclear thermal rocket (nuclear fission energy)	propellant (typ. hydrogen) is passed through a nuclear reactor to heat to high temperature	I_{sp} can be high, perhaps 900 seconds or more, above unity thrust/weight ratio with some designs	Maximum temperature is limited by materials technology, some radioactive particles can be present in exhaust in some designs, nuclear reactor shielding is heavy, unlikely to be permitted from surface of the Earth, thrust/weight ratio is not high.
Gas core reactor rocket (nuclear fission energy)	Nuclear reaction using a gaseous state fission reactor in	Very hot propellant, not limited by keeping reactor solid, I_{sp}	Difficulties in heating propellant without losing fissionables in exhaust, massive thermal issues

	intimate contact with propellant	between 1500 and 3000 seconds but with very high thrust	particularly for nozzle/throat region, exhaust almost inherently highly radioactive. Nuclear lightbulb variants can contain fissionables, but cut I_{sp} in half.
Fission-fragment rocket (nuclear fission energy)	Fission products are directly exhausted to give thrust		Theoretical only at this point.
Fission sail (nuclear fission energy)	A sail material is coated with fissionable material on one side	No moving parts, works in deep space	Theoretical only at this point.
Nuclear salt-water rocket (nuclear fission energy)	Nuclear salts are held in solution, caused to react at nozzle	Very high I_{sp} , very high thrust	Thermal issues in nozzle, propellant could be unstable, highly radioactive exhaust. Theoretical only at this point.
Nuclear pulse propulsion (exploding fission/fusion bombs)	Shaped nuclear bombs are detonated behind vehicle and blast is caught by a 'pusher plate'	Very high I_{sp} , very high thrust/weight ratio, no show stoppers are known for this technology	Never been tested, pusher plate may throw off fragments due to shock, minimum size for nuclear bombs is still pretty big, expensive at small scales, nuclear treaty issues, fallout when used below Earth's magnetosphere.
Antimatter catalyzed nuclear pulse propulsion (fission and/or fusion energy)	Nuclear pulse propulsion with antimatter assist for smaller bombs	Smaller sized vehicle might be possible	Containment of antimatter, production of antimatter in macroscopic quantities isn't currently feasible. Theoretical only at this point.
Fusion rocket (nuclear fusion energy)	Fusion is used to heat propellant	Very high exhaust velocity	Largely beyond current state of the art.
Antimatter rocket (annihilation energy)	Antimatter annihilation heats propellant	Extremely energetic, very high theoretical exhaust velocity	Problems with antimatter production and handling; energy losses in neutrinos, gamma rays, muons; thermal issues. Theoretical only at this point

History of rocket engines

According to the writings of the Roman Aulus Gellius, in c. 400 BC, a Greek Pythagorean named Archytas, propelled a wooden bird along wires using steam. However, it would not appear to have been powerful enough to take off under its own thrust.

The *aeolipile* described in the first century BC (often known as *Hero's engine*) essentially consists of a steam rocket on a bearing. It was created almost two millennia before the Industrial Revolution but the principles behind it were not well understood, and its full potential was not realized for a millennium.

The availability of black powder to propel projectiles was a precursor to the development of the first solid rocket. Ninth Century Chinese Taoist alchemists discovered black powder in a search for the Elixir of life; this accidental discovery led to fire arrows which were the first rocket engines to leave the ground.

Rocket engines were also brought in use by Tippu Sultan, The king of Mysore. These rockets could be of various sizes, but usually consisted of a tube of soft hammered iron about 8" long and 1½ - 3" diameter, closed at one end and strapped to a shaft of bamboo about 4 ft. long. The iron tube acted as a combustion chamber and contained well packed black powder propellant. A rocket carrying about one pound of powder could travel almost 1,000 yards. These 'rockets', fitted with swords used to travel long distance, several meters above in air before coming down with swords edges facing the enemy. These rockets were used against British empire very effectively.

Slow development of this technology continued up to the later 20th Century, when the writings of Konstantin Tsiolkovsky first talked about liquid fuelled rocket engines.

These independently became a reality thanks to Robert Goddard. Goddard also used a De Laval nozzle for the first time on a rocket, doubling the thrust and multiplying up the efficiency by several times.

During the late 1930s, German scientists, such as Wernher von Braun and Hellmuth Walter, investigated installing liquid-fuelled rockets in military aircraft (Heinkel He 112, He 111, He 176 and Messerschmitt Me 163). The turbopump was first employed by German scientists in WWII. Until then cooling the nozzle was problematic, and the A4 ballistic missile used dilute alcohol for the fuel, which reduced the combustion temperature sufficiently.

Staged combustion (*Замкнутая схема*) was first proposed by Alexey Isaev in 1949. The first staged combustion engine was the S1.5400 used in the Soviet planetary rocket, designed by Melnikov, a former assistant to Isaev. About the same time (1959), Nikolai Kuznetsov began work on the closed cycle engine NK-9 for Korolev's orbital ICBM, GR-1. Kuznetsov later evolved that design into the NK-15 and NK-33 engines for the unsuccessful Lunar N1 rocket.

In the West, the first laboratory staged-combustion test engine was built in Germany in 1963, by Ludwig Boelkow.

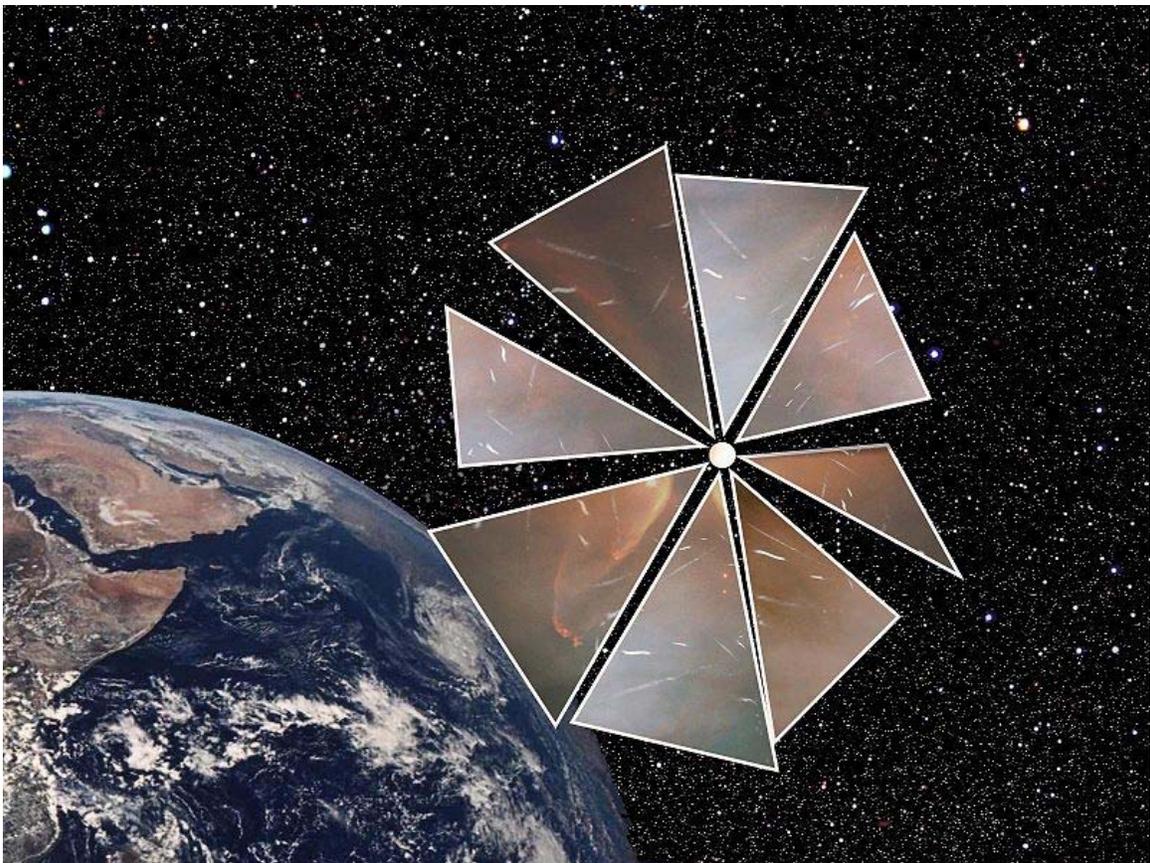
Hydrogen peroxide / kerosene fuelled engines such as the British Gamma of the 1950s used a closed-cycle process (arguably not *staged combustion*, but that's mostly a question of semantics) by catalytically decomposing the peroxide to drive turbines *before* combustion with the kerosene in the combustion chamber proper. This gave the efficiency advantages of staged combustion, whilst avoiding the major engineering problems.

Liquid hydrogen engines were first successfully developed in America, the RL-10 engine first flew in 1962. Hydrogen engines were used as part of the Project Apollo; the liquid hydrogen fuel giving a rather lower stage mass and thus reducing the overall size and cost of the vehicle.

The Space Shuttle's SSME is the highest ground-launched specific impulse rocket engine to fly.

Chapter-9

Solar Sail



An artist's depiction of a Cosmos 1 type spaceship in orbit

Solar sails (also called **light sails** or **photon sails**) are a form of spacecraft propulsion using the radiation pressure of light from a star or laser to push enormous ultra-thin mirrors to high speeds.

In 2010, IKAROS was the world's first spacecraft designed to use solar sailing propulsion to be successfully launched.

Physics

There are two sources of solar forces. The first is radiation pressure, and the second is due to solar wind. The radiation pressure is much stronger than the wind pressure. In 1924, the Russian space engineer Friedrich Zander proposed that, since light provides a small amount of thrust, this effect could be used as a form of space propulsion requiring no fuel. Einstein proposed – and experiments confirm – that photons have a momentum $p=E/c$, hence each light photon absorbed by or reflecting from a surface exerts a small amount of radiation pressure. This results in forces of about $4.57 \times 10^{-6} \text{ N/m}^2$ for absorbing surfaces perpendicular to the radiation in earth orbit, and twice as much, if the radiation is reflected. This was proven experimentally by Russian physicist Peter Lebedev in 1900, and independently by Nichols and Hull at Dartmouth in 1901 using a Nichols radiometer.

Charged particles from the solar wind are able to knock out power grids on Earth, and point the tails of comets away from the sun. The solar wind averages 6.7 billion tons per hour at 520 km/s with "slow" low energy coronal ejections reaching 400 km/s and "fast" higher energy ejections averaging 750 km/s. At the distance of the earth, this results in average solar wind pressure of $3.4 \times 10^{-9} \text{ N/m}^2$, three orders of magnitude less than the photonic radiation pressure. Still the solar wind dominates many phenomena because its interaction cross section with gases and charged particles is about 10^9 times larger than that of the photons.

Both of these forces are small and decrease with the inverse square distance from the sun. Even large sails produce minute acceleration, but over time, sails can build up considerable speeds. Because the force on the sails and the force of gravity from the sun both vary as inverse square functions, solar sail vessels can be rated by the ratio of the sail's force divided by the gravitational force. Solar sail vessels with the same rating are able to follow the same trajectories.

Changing course trajectories can be accomplished in two ways. First, tilting the sail with respect to the light source changes the direction of acceleration because the force on a sail from reflected radiation and wind acts in a direction perpendicular to its surface. Smaller auxiliary vanes can be used to gently pull the main sail into its new position. Second, gravity from a nearby mass, such as a star or planet, will alter the direction of a spaceship. When orbiting a star or planet, sails can be used to slow down and spiral inward, or to increase the velocity and spiral outward. If the planet has moons or the star has planets, these techniques can be used to achieve slingshots around these bodies.

Fly modes

Escaping planetary orbit

Sails orbit, and therefore do not need to hover or move directly toward or away from the sun. Almost all missions would use the sail to change orbit, rather than thrusting directly away from a planet or the sun. The sail is rotated slowly as the sail orbits around a planet

so the thrust is in the direction of the orbital movement to move to a higher orbit or against it to move to a lower orbit. When an orbit is far enough away from a planet, the sail then begins similar maneuvers in orbit around the sun.

Beam propelled

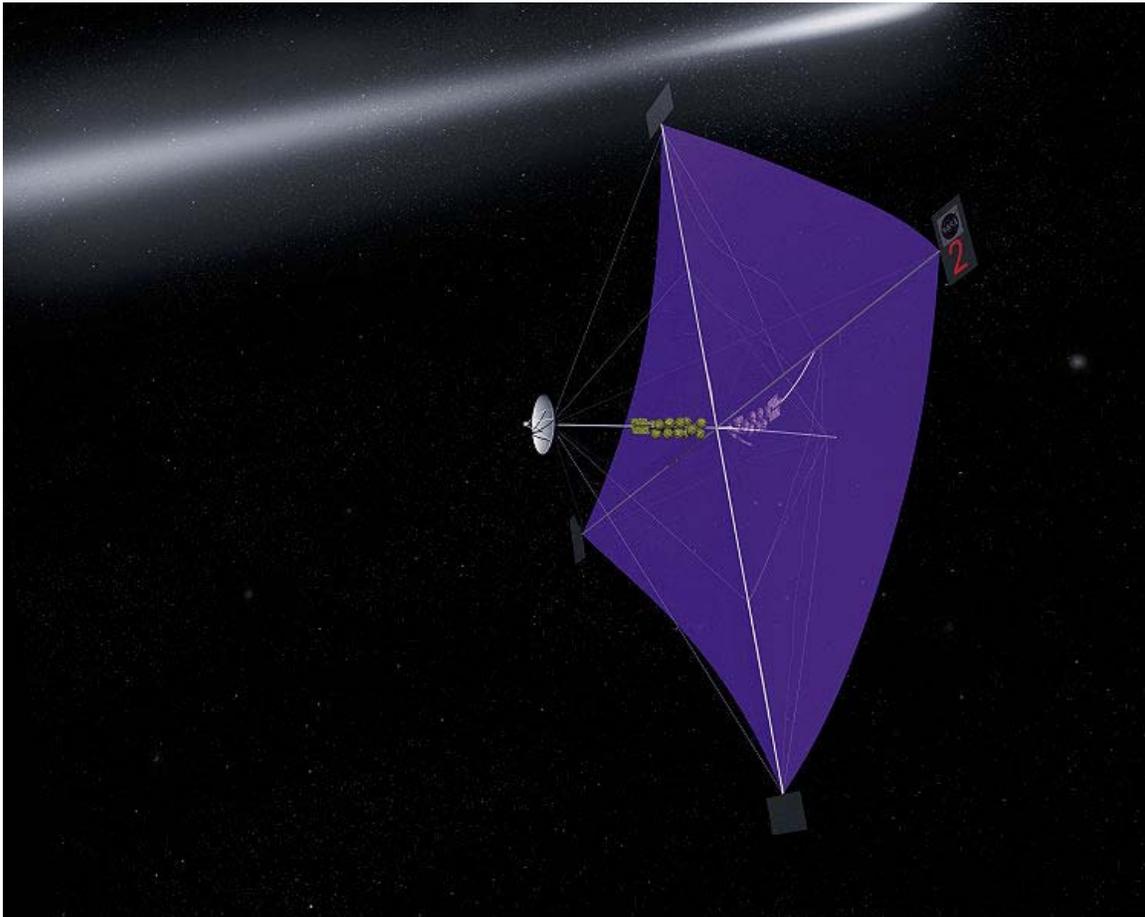
Most theoretical studies of interstellar missions with a solar sail plan to push the sail with a very large laser beam-powered propulsion direct impulse beam. The thrust vector (spatial vector) would therefore be away from the Sun and toward the target.

In theory a lightsail driven by a laser or other beam from Earth can be used to slow down a spacecraft approaching a distant star or planet, by detaching part of the sail and using it to focus the beam on the forward-facing surface of the rest of the sail. In practice, however, most of the slowing would happen while the two parts are at a great distance from each other, and that means that, to do that focusing, it would be necessary to give the detached part an accurate optical shape and orientation. This solution is also limited because the lasers used to accelerate or decelerate a sail ship could take years, decades, or centuries to reach the craft, depending on the distance.

Limitations of solar sails

Solar sails do not work well, if at all, in low Earth orbit below about 800 km altitude due to erosion or air drag. Above that altitude they give very small accelerations that take months to build up to useful speeds. Solar sails have to be physically large, and payload size is often small. Deploying solar sails is also highly challenging to date.

Investigated sail designs



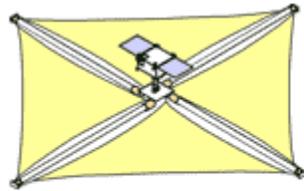
NASA study of a solar sail. The sail would be half a kilometre wide.

"Parachutes" would have very low mass, but theoretical studies show that they will collapse from the forces placed by shrouds. Radiation pressure does not behave like aerodynamic pressure.

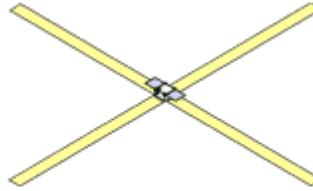
The highest thrust-to-mass designs known (2007) were theoretical designs developed by Eric Drexler. He designed a sail using reflective panels of thin aluminium film (30 to 100 nanometres thick) supported by a purely tensile structure. It rotated and would have to be continually under slight thrust. He made and handled samples of the film in the laboratory, but the material is too delicate to survive folding, launch, and deployment, hence the design relied on space-based production of the film panels, joining them to a deployable tension structure. Sails in this class would offer area per unit mass and hence accelerations up to "fifty times higher" than designs based on deployable plastic films.

The highest-thrust to mass designs for ground-assembled deployable structures are square sails with the masts and guy lines on the dark side of the sail. Usually there are four masts that spread the corners of the sail, and a mast in the center to hold guy-wires. One of the

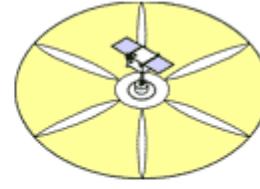
largest advantages is that there are no hot spots in the rigging from wrinkling or bagging, and the sail protects the structure from the sun. This form can therefore go quite close to the sun, where the maximum thrust is present. Control would probably use small sails on the ends of the spars.



Square Sail (not to scale)



Heliogyro (not to scale)



Spinning Disk Sail (not to scale)

In the 1970s JPL did extensive studies of rotating blade and rotating ring sails for a mission to rendezvous with Halley's Comet. The intention was that such structures would be stiffened by their angular momentum, eliminating the need for struts, and saving mass. In all cases, surprisingly large amounts of tensile strength were needed to cope with dynamic loads. Weaker sails would ripple or oscillate when the sail's attitude changed, and the oscillations would add and cause structural failure. So the difference in the thrust-to-mass ratio was almost nil, and the static designs were much easier to control.

JPL's reference design was called the "heliogyro" and had plastic-film blades deployed from rollers and held out by centrifugal forces as it rotated. The spacecraft's attitude and direction were to be completely controlled by changing the angle of the blades in various ways, similar to the cyclic and collective pitch of a helicopter. Although the design had no mass advantage over a square sail, it remained attractive because the method of deploying the sail was simpler than a strut-based design.

JPL also investigated "ring sails" (Spinning Disk Sail in the above diagram), panels attached to the edge of a rotating spacecraft. The panels would have slight gaps, about one to five percent of the total area. Lines would connect the edge of one sail to the other. Masses in the middles of these lines would pull the sails taut against the coning caused by the radiation pressure. JPL researchers said that this might be an attractive sail design for large manned structures. The inner ring, in particular, might be made to have artificial gravity roughly equal to the gravity on the surface of Mars.

A solar sail can serve a dual function as a high-gain antenna. Designs differ, but most modify the metallization pattern to create a holographic monochromatic lens or mirror in the radio frequencies of interest, including visible light.

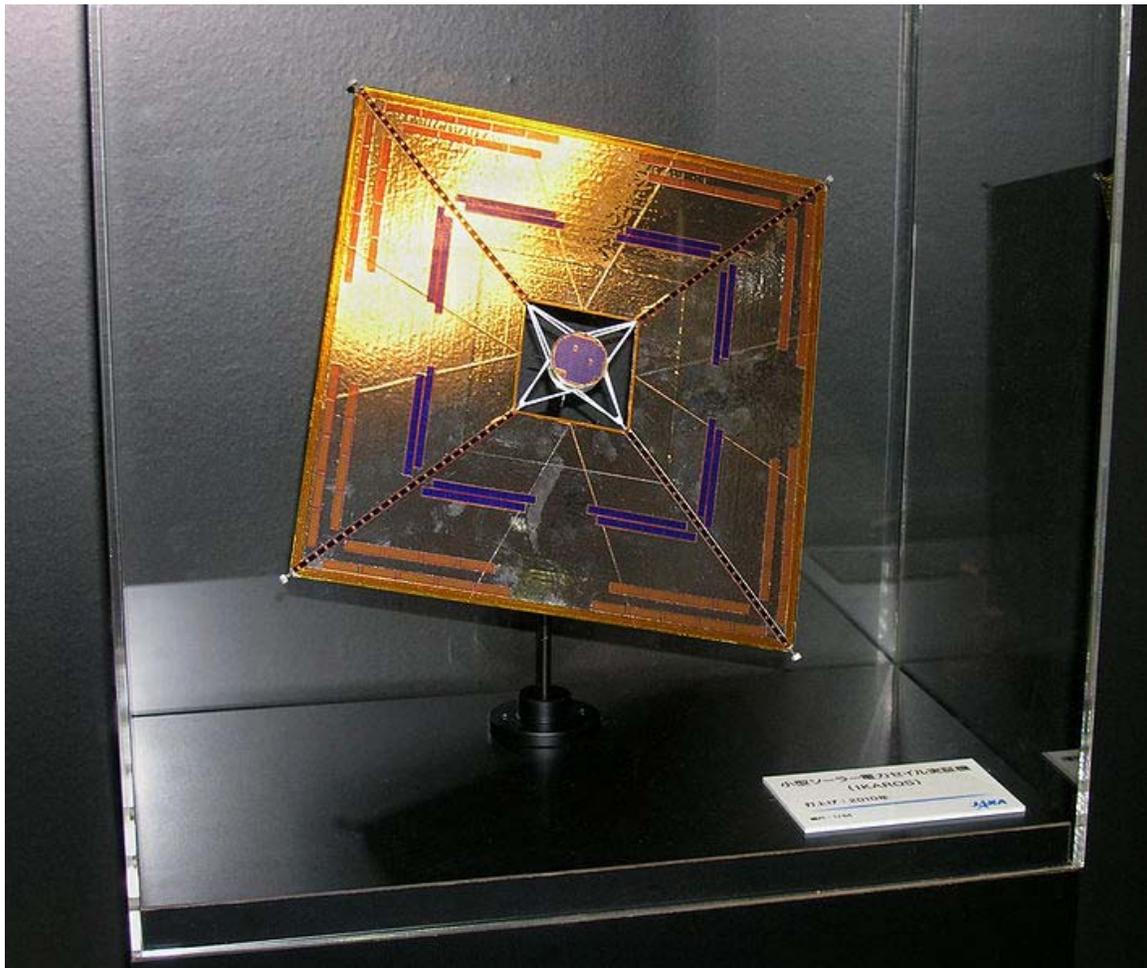
Pekka Janhunen from FMI has invented a type of solar sail called the electric solar wind sail. Mechanically it has little in common with the traditional solar sail design, because the sails are substituted with straightened conducting tethers (wires) which are placed radially around the host ship. The wires are electrically charged and thus an electric field is created around the wires. The electric field of the wires extends a few tens of metres

into the surrounding solar wind plasma. Because the solar wind electrons react on the electric field (similarly to the photons on a traditional solar sail), the functional radius of the wires is based on the electric field that is generated around the wire rather than the actual wire itself. This fact also makes it possible to maneuver a ship with an electric solar wind sail by regulating the electric charge of the wires. A full-sized operational electric solar wind sail would have 50-100 straightened wires with a length of about 20 km each.

A quite similar concept is the Magnetic sail, which would also employ the solar wind, but interact with the magnetic charge of the particles in the wind, rather than the electric. Typically it is also constructed with wires as "sails", but in contrast to a electric sail, it uses wire **loops**, and runs a static current through them instead of applying a static voltage.

Both designs have limited ability to direct the thrust, compared to a traditional solar sail, which can thrust sideways by angling the mirror relative to the light source.

Sail testing in space



The model of IKAROS at the 61st International Astronautical Congress in 2010

Until 2010, no solar sails had been successfully used in space as primary propulsion systems. On 21 May 2010, the Japan Aerospace Exploration Agency (JAXA) launched the “IKAROS” (Interplanetary Kite-craft Accelerated by Radiation Of the Sun) spacecraft, which deployed a 200 m² polyimide experimental solar sail on June 10. In July, the next phase for the demonstration of acceleration by radiation began. On 9 July, it was verified that IKAROS collected radiation from the sun and began photon acceleration by the orbit determination of IKAROS by range-and-range-rate (RARR) that is newly calculated in addition to the data of the relativization accelerating speed of IKAROS between IKAROS and the Earth that has been taken since before the Doppler effect was utilized. The data showed that IKAROS appears to have been solar-sailing since 3 June when it deployed the sail.

IKAROS has a diagonal spinning square sail 20 m (66 ft) made of a 7.5-micrometre (0.30-mil) thick sheet of polyimide. A thin-film solar array is embedded in the sail. Eight LCD panels are embedded in the sail, whose reflectance can be adjusted for attitude control. IKAROS will spend six months traveling to Venus, and then will begin a three-year journey to the far side of the Sun.

Solar pressure demonstrated for attitude control

Both the Mariner 10 mission, which flew by the planets Mercury and Venus, and the MESSENGER mission to Mercury demonstrated the use of solar pressure as a method of attitude control in order to conserve attitude-control propellant.

Hayabusa also used solar pressure as a method of attitude control to compensate for broken reaction wheels and chemical thruster.

Solar sail deployment tests



Full-scale (20mx20m) deployment test by DLR/ESA in 1999

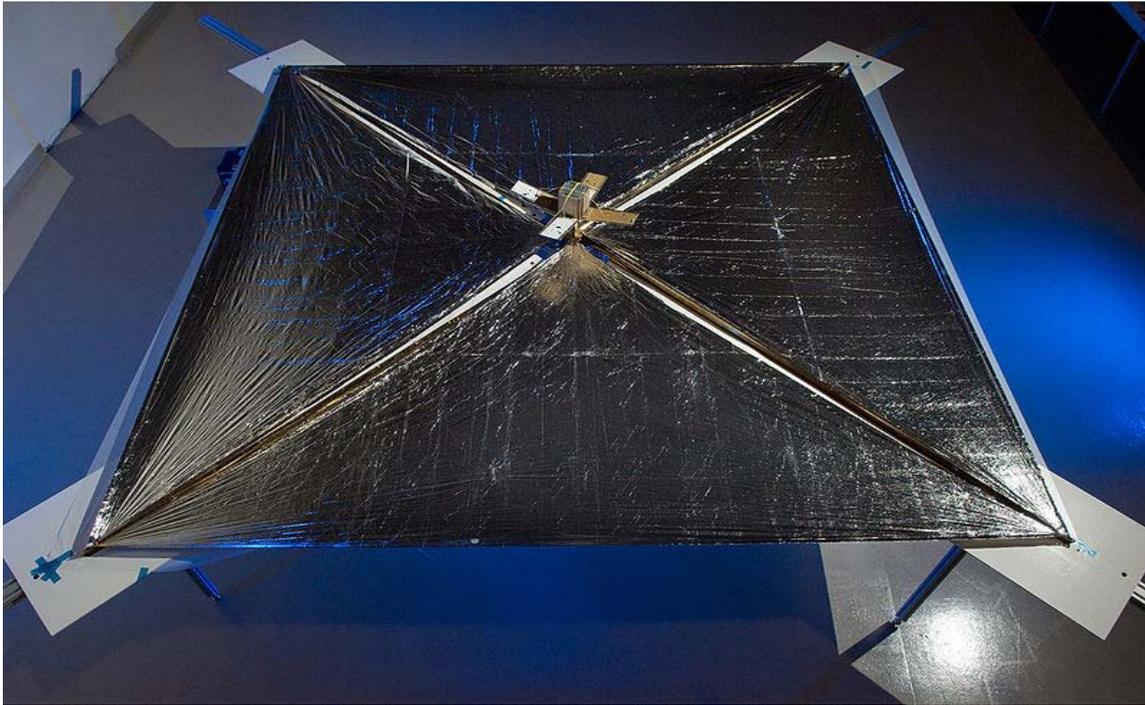
NASA has successfully tested deployment technologies on small scale sails in vacuum chambers.

On February 4, 1993, Znamya 2, a 20-meter wide aluminized-mylar reflector, was successfully tested from the Russian Mir space station. Although the deployment test was successful, the experiment only demonstrated the deployment, not propulsion. A second test, Znamaya 2.5, failed to deploy properly.

In 1999, a full-scale deployment test of a solar sail has been performed on ground at DLR/ESA in Cologne.

On August 9, 2004, the Japanese ISAS successfully deployed two prototype solar sails from a sounding rocket. A clover type sail was deployed at 122 km altitude and a fan type sail was deployed at 169 km altitude. Both sails used 7.5 micrometer thick film. The experiment was purely a test of the deployment mechanisms, not of propulsion.

Partially successful solar sail propulsion tests



NanoSail-D of LightSail-1 with sail deployed

A joint private project between Planetary Society, Cosmos Studios and Russian Academy of Science launched *Cosmos 1* on June 21, 2005, from a submarine in the Barents Sea, but the Volna rocket failed, and the spacecraft failed to reach orbit. A solar sail would have been used to gradually raise the spacecraft to a higher earth orbit. The mission would have lasted for one month. A suborbital prototype test by the group failed in 2001 as well, also because of rocket failure. The same group announced plans on Carl Sagan's 75th birthday (November 9, 2009) to make three further attempts, dubbed LightSail-1, -2, and -3. The new design will use a 32-square-meter Mylar sail, deployed in four triangular segments like NanoSail-D. The launch configuration is that of three adjacent CubeSats, and is scheduled to launch on a Minotaur IV rocket in Q4 2010.

A 15-meter-diameter solar sail (SSP, solar sail sub payload, *soraseiru sabupeiro-do*) was launched together with ASTRO-F on a M-V rocket on February 21, 2006, and made it to orbit. It deployed from the stage, but opened incompletely.

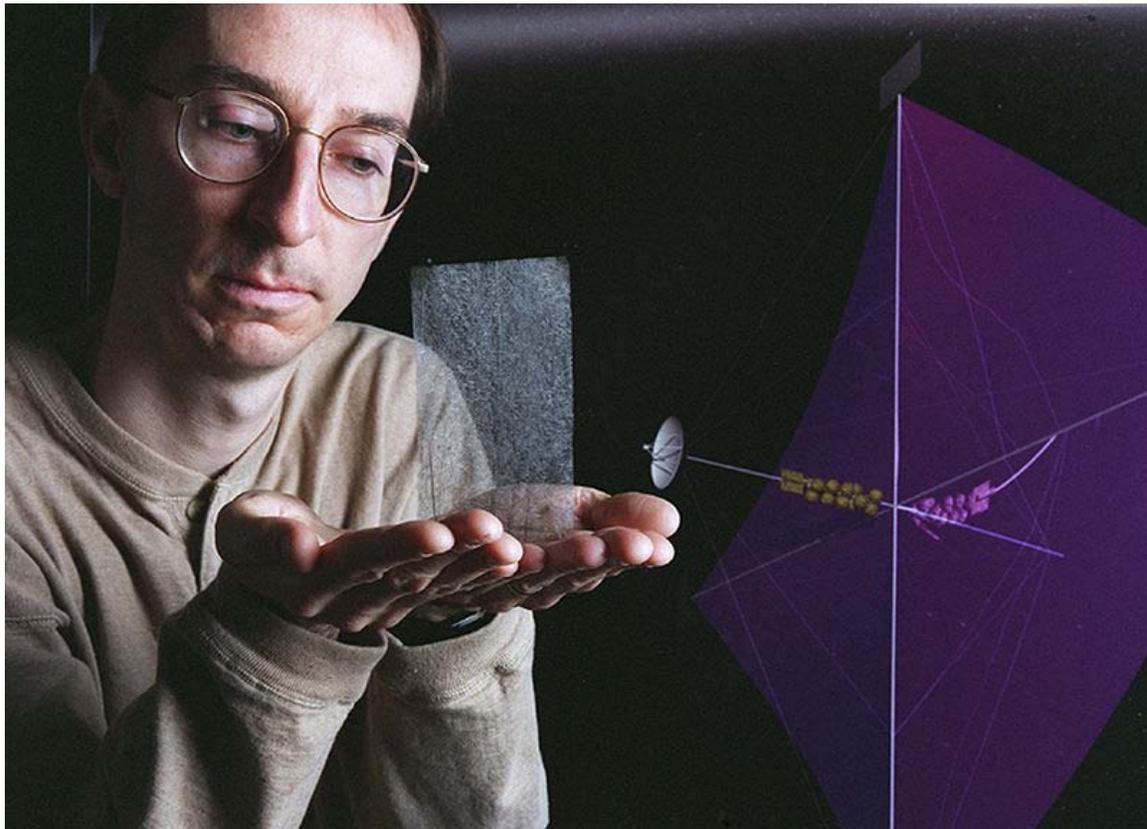
A team from the NASA Marshall Space Flight Center (Marshall), along with a team from the NASA Ames Research Center, developed a solar sail mission called **NanoSail-D** which was lost in a launch failure aboard a Falcon 1 rocket on 3 August 2008. The second backup version will be launched with FASTSAT on a Minotaur IV on November 19, 2010. The primary objective of the mission is to test sail deployment technologies. The spacecraft might not have returned useful data about solar sail propulsion, according to Edward E. Montgomery, technology manager of Solar Sail Propulsion at Marshall,

"The orbit available to us in this launch opportunity is so low, it may not allow us to stay in orbit long enough for solar pressure effects to accumulate to a measurable degree." The NanoSail-D structure was made of aluminium and plastic, with the spacecraft massing less than 10 pounds (4.5 kg). The sail has about 100 square feet (9.3 m²) of light-catching surface.

Future solar sail propulsion tests

A team from the Surrey Space Centre at the University of Surrey are developing a solar sail demonstration mission called the "CubeSail". This mission is due to launch in late 2011. The CubeSail is based on the CubeSat standard and when stowed it will occupy a 3U standard volume (3, 100mm x 100mm x 100mm). When in orbit, it will extend four 3.6m booms, deploying a sail of 25m². The mission's primary objective is to demonstrate deployment of a solar sail and the concept of solar sailing. Finally and at its end-of-life it will use its sail to change its ballistic coefficient and reenter the Earth's atmosphere. This final phase of the mission has attracted much media attention as it has the potential to be used on board larger spacecraft as a de-orbiting device and potentially to solve the Space debris problem.

Sail materials



NASA engineer Les Johnson views interstellar sail material

The material developed for the Drexler solar sail was a thin aluminum film with a baseline thickness of 0.1 micrometres, to be fabricated by vapor deposition in a space-based system. Drexler used a similar process to prepare films on the ground. As anticipated, these films demonstrated adequate strength and robustness for handling in the laboratory and for use in space, but not for folding, launch, and deployment.

The most common material in current designs is aluminized 2 μm Kapton film. It resists the heat of a pass close to the Sun and still remains reasonably strong. The aluminium reflecting film is on the Sun side. The sails of *Cosmos 1* were made of aluminized PET film (Mylar).

Research by Dr. Geoffrey Landis in 1998-9, funded by the NASA Institute for Advanced Concepts, showed that various materials such as alumina for laser lightsails and carbon fiber for microwave pushed lightsails were superior sail materials to the previously standard aluminium or Kapton films.

In 2000, Energy Science Laboratories developed a new carbon fiber material which might be useful for solar sails. The material is over 200 times thicker than conventional solar sail designs, but it is so porous that it has the same mass. The rigidity and durability of this material could make solar sails that are significantly sturdier than plastic films. The material could self-deploy and should withstand higher temperatures.

There has been some theoretical speculation about using molecular manufacturing techniques to create advanced, strong, hyper-light sail material, based on nanotube mesh weaves, where the weave "spaces" are less than half the wavelength of light impinging on the sail. While such materials have so far only been produced in laboratory conditions, and the means for manufacturing such material on an industrial scale are not yet available, such materials could mass less than 0.1 g/m^2 , making them lighter than any current sail material by a factor of at least 30. For comparison, 5 micrometre thick Mylar sail material mass 7 g/m^2 , aluminized Kapton films have a mass as much as 12 g/m^2 , and Energy Science Laboratories' new carbon fiber material masses 3 g/m^2 .

Applications

Satellites

Robert L. Forward pointed out that a solar sail could be used to modify the orbit of a satellite around the Earth. In the limit, a sail could be used to "hover" a satellite above one pole of the Earth. Spacecraft fitted with solar sails could also be placed in close orbits about the Sun that are stationary with respect to either the Sun or the Earth, a type of satellite named by Forward a statite. This is possible because the propulsion provided by the sail offsets the gravitational potential of the Sun. Such an orbit could be useful for studying the properties of the Sun over long durations.

Such a spacecraft could conceivably be placed directly over a pole of the Sun, and remain at that station for lengthy durations. Likewise a solar sail-equipped spacecraft could also

remain on station nearly above the polar terminator of a planet such as the Earth by tilting the sail at the appropriate angle needed to just counteract the planet's gravity.

In his book, *The Case for Mars*, Robert Zubrin points out that the reflected sunlight from a large statite placed near the polar terminator of the planet Mars could be focussed on one of the Martian polar ice caps to significantly warm the planet's atmosphere. Such a statite could be made from asteroid material.

Trajectory corrections

The MESSENGER probe en route to Mercury is using light pressure reacting against its solar panels to perform fine trajectory corrections. By changing the angle of the solar panels relative to the sun, the amount of solar radiation pressure can be varied to adjust the spacecraft trajectory more delicately than is possible with thrusters. Minor errors are greatly amplified by gravity assist maneuvers, so very small corrections before lead to large savings in propellant afterward.

Interstellar flight

In the 1980s, Robert Forward proposed two beam-powered propulsion schemes using either lasers or masers to push giant sails to a significant fraction of the speed of light.

In *The Flight of the Dragonfly*, Forward described a light sail propelled by superlasers. As the starship neared its destination, the outer portion of the sail would detach. The outer sail would then refocus and reflect the lasers back onto a smaller, inner sail. This would provide braking thrust to stop the ship in the destination star system.

Both methods pose monumental engineering challenges. The lasers would have to operate for years continuously at gigawatt strength. Second, they would demand more energy than the Earth currently consumes. Third, Forward's own solution to the electrical problem requires enormous solar panel arrays to be built at or near the planet Mercury. Fourth, a planet-sized mirror or fresnel lens would be needed several dozen astronomical units from the Sun to keep the lasers focused on the sail. Fifth, the giant braking sail would have to act as a precision mirror to focus the braking beam onto the inner "deceleration" sail.

A potentially easier approach would be to use a maser to drive a "solar sail" composed of a mesh of wires with the same spacing as the wavelength of the microwaves, since the manipulation of microwave radiation is somewhat easier than the manipulation of visible light. The hypothetical "Starwisp" interstellar probe design would use a maser to drive it. Masers spread out more rapidly than optical lasers owing to their longer wavelength, and so would not have as long an effective range.

Masers could also be used to power a painted solar sail, a conventional sail coated with a layer of chemicals designed to evaporate when struck by microwave radiation. The

momentum generated by this evaporation could significantly increase the thrust generated by solar sails, as a form of lightweight ablative laser propulsion.

To further focus the energy on a distant solar sail, designs have considered the use of a large zone plate. This would be placed at a location between the laser or maser and the spacecraft. The plate could then be propelled outward using the same energy source, thus maintaining its position so as to focus the energy on the solar sail.

Additionally, it has been theorized by da Vinci Project contributor T. Pesando that solar sail-utilizing spacecraft successful in interstellar travel could be used to carry their own zone plates or perhaps even masers to be deployed during flybys at nearby stars. Such an endeavor could allow future solar-sailed craft to effectively utilize focused energy from other stars rather than from the Earth or Sun, thus propelling them more swiftly through space and perhaps even to more distant stars. However, the potential of such a theory remains uncertain if not dubious due to the high-speed precision involved and possible payloads required.

Another more physically realistic approach would be to use the light from the home star to accelerate. The ship would first orbit continuously away around the home star until the appropriate starting velocity is reached, then the ship would begin its trip away from the system using the light from the star to keep accelerating. Beyond some distance, the ship would no longer receive enough light to accelerate it significantly, but would maintain its course due to inertia. When nearing the target star, the ship could turn its sails toward it and begin to orbit inward to decelerate. Additional forward and reverse thrust could be achieved with more conventional means of propulsion such as rockets.

Future approaches

Despite the losses of *Cosmos 1* and NanoSail-D (which were due to failure of their launchers), scientists and engineers around the world remain encouraged and continue to work on solar sails. While most direct applications created so far intend to use the sails as inexpensive modes of cargo transport, some scientists are investigating the possibility of using solar sails as a means of transporting humans. This goal is strongly related to the management of very large (i.e. well above 1 km²) surfaces in space and the sail making advancements. Thus, in the near/medium term, solar sail propulsion is aimed chiefly at accomplishing a very high number of non-crewed missions in any part of the solar system and beyond.

Solar sail launching projects in 2010 and 2011

On 21 May 2010, Japan Aerospace Exploration Agency (Jaxa) launched the world's first interplanetary solar sail spacecraft "IKAROS" (*Interplanetary Kite-craft Accelerated by Radiation Of the Sun*) to Venus. NASA launched the second NanoSail-D unit stowed inside the FASTSAT satellite on the Minotaur IV on November 19, 2010. The ejection date from the FASTSAT microsatellite was planned for December 6, 2010 but

deployment only occurred on January 20, 2011. The Planetary Society of the United States plans to launch an artificial satellite "LightSail-1" onto the Earth's orbit in 2011.

Mathematical survey

The Extended Heliocentric Reference Frame

- In the 1991-92 the classical equations of solar sail motion in the solar gravitational field were written using a different mathematical formalism, namely, *the lightness vector* fully characterizing the sailcraft dynamics. In addition, a solar-sail spacecraft has been supposed to be able to reverse its motion (in the solar system) provided that its sail is sufficiently light that *sailcraft sail loading* (σ) is not higher than 2.1 g/m². This value entails a very high-performance technology, but probably within the capabilities of emerging technologies.
- For describing the concept of fast sailing and some related items, we need to define two frames of reference. The first is an inertial Cartesian coordinate system centred on the Sun or a heliocentric inertial frame (HIF, for short). For instance, the plane of reference, or the XY plane, of HIF can be the mean ecliptic at some standard epoch such as J2000. The second Cartesian reference frame is the so-called heliocentric orbital frame (HOF, for short) with the origin in the sailcraft barycenter. The x-axis of HOF is the direction of the Sun-to-sailcraft vector, or position vector, the z-axis is along the sailcraft orbital angular momentum, whereas the y-axis completes the counterclockwise triad. Such a definition can be extended to sailcraft trajectories, including both counterclockwise and clockwise arcs of motion, in such a way that HOF is always a continuous positively-oriented triad. The sail orientation unit vector (defined in sailcraft), say, \mathbf{n} can be specified in HOF by a pair of angles, e.g. the azimuth α and the elevation δ . Elevation is the angle that \mathbf{n} forms with the xy-plane of HOF ($-90^\circ \leq \delta \leq 90^\circ$). Azimuth is the angle that the projection of \mathbf{n} onto the HOF xy-plane forms with the HOF x-axis ($0 \leq \alpha < 360^\circ$). In HOF, azimuth and elevation are equivalent to longitude and latitude, respectively.
- The sailcraft lightness vector $\mathbf{L} = [\lambda_r, \lambda_t, \lambda_n]$ depends on α and δ (non-linearly) and the thermo-optical parameters of the sail materials (linearly). Neglecting a small contribution coming from the aberration of light, one has the following particular cases (irrespective of the sail material):
 1. $\alpha = 0, \delta = 0 \Leftrightarrow [\lambda_r, 0, 0] \Leftrightarrow \lambda = |\mathbf{L}| = \lambda_r$
 2. $\alpha \neq 0, \delta = 0 \Leftrightarrow [\lambda_r, \lambda_t, 0]$
 3. $\alpha = 0, \delta \neq 0 \Leftrightarrow [\lambda_r, 0, \lambda_n]$

A flight example

Conventional strategy

- Suppose a sailcraft is built with an all-metal sail of aluminium and chromium such that $\sigma = 2 \text{ g/m}^2$. A launcher delivers the (packed) sailcraft at some million kilometers from the Earth. There, the whole sailcraft is deployed and begins its flight in the solar system (here, for the sake of simplicity, any gravitational perturbation from planets is neglected). A conventional spacecraft would move approximately in a circular orbit at about 1 AU from the Sun. In contrast, a sailcraft like this one is sufficiently light to be able to escape the solar system or to point to some distant object in the heliosphere. If the direction that sail's surface faces, represented by surface normal vector \mathbf{n} , is parallel to the local sun-light direction (i.e. the sail faces toward the sun), then $\lambda_r = \lambda = 0.725$ (i.e. $1/2 < \lambda < 1$); as a result, this sailcraft moves on a hyperbolic orbit. Its speed *at infinity* is equal to 20 km/s. Strictly speaking, this potential solar sail mission would be faster than the current record speed for missions beyond the planetary range, that of Voyager 1, which is 17 km/s or about 3.6 AU/yr (1 AU/yr = 4.7404 km/s). However, three kilometers per second are not meaningful in the context of very deep space missions.
- As a consequence, one has to resort to some \mathbf{L} having more than one component different from zero. The classical way to gain speed is to tilt the sail at some suitable *positive* α . If $\alpha = +21^\circ$, then the sailcraft begins by accelerating; after about two months, it achieves 32 km/s. However, this is a speed peak inasmuch as its subsequent motion is characterized by a monotonic speed decrease towards an asymptotic value, or the cruise speed, of 26 km/s. After 18 years, the sailcraft is 100 AU away from the Sun. This would mean a pretty fast mission. However, considering that a sailcraft with 2 g/m^2 is technologically advanced, is there any other way to increase its speed significantly? Yes, there is. Let us try to explain this effect of non-linear dynamics.

Optimal strategy

- The above figures show that spiralling out from a circular orbit is not a convenient mode for a sailcraft to be sent away from the Sun since it would not have a high enough excess speed. On the other hand, it is known from astrodynamics that a conventional Earth satellite has to perform a rocket maneuver at/around its perigee for maximizing its speed at "infinity". Similarly, one can think of delivering a sailcraft close to the Sun to get much more energy from the solar photon pressure (that scales as $1/R^2$). For instance, suppose one starts from a point at 1 AU on the ecliptic and achieves a perihelion distance of 0.2 AU in the same plane by a two-dimensional trajectory. In general, there are three ways to deliver a sailcraft, initially at R_0 from the Sun, to some distance $R < R_0$:
 - using an additional propulsion system to send the folded-sail sailcraft to the perihelion of an elliptical orbit; there, the sail is deployed with its axis

parallel to the sun-light for getting the maximum solar flux at the chosen distance;

- spiralling in by α slightly negative, namely, via a slow deceleration;
- strongly decelerating by a "sufficiently large" sail-axis angle negative in HOF.

The first way - although usable as a good reference mode - requires another high-performance propulsion system.

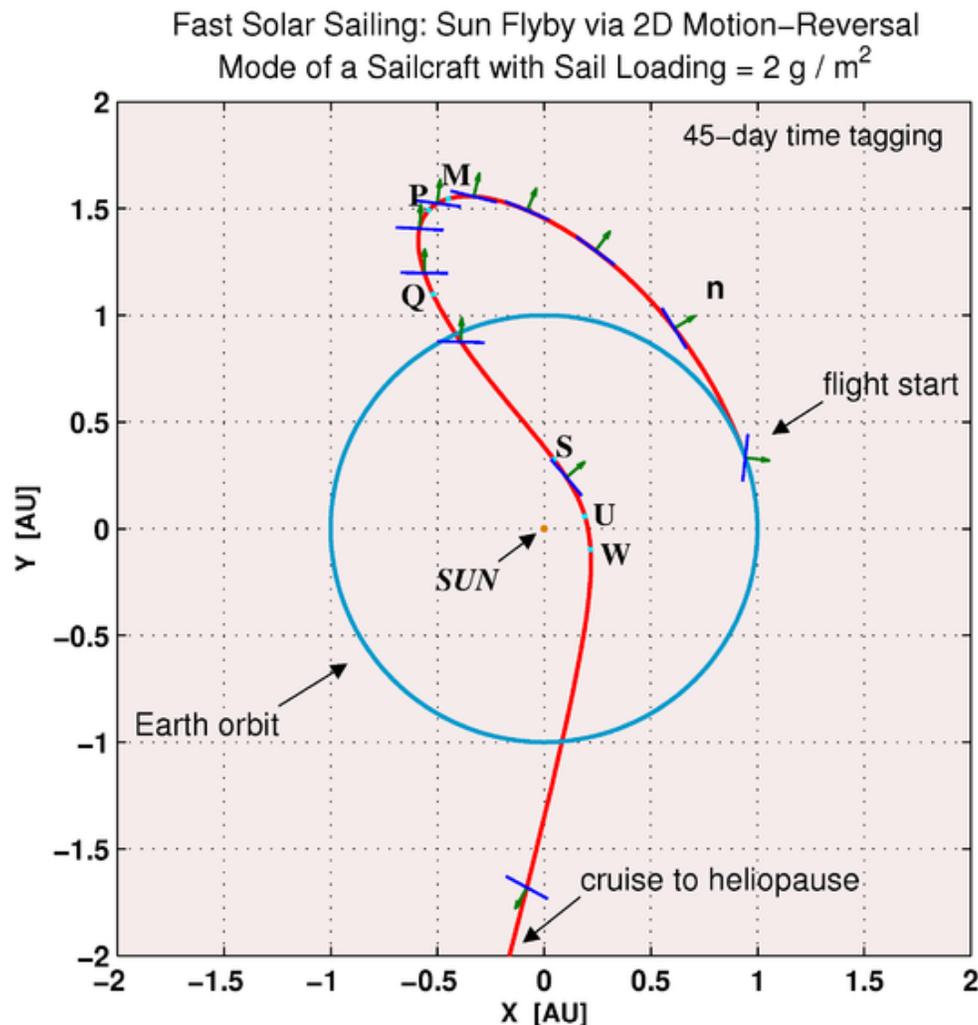
The second way is ruled out in the present case of $\sigma = 2 \text{ g/m}^2$; as a matter of fact, a small $\alpha < 0$ entails a λ_r too high and a negative λ_t too low in absolute value: the sailcraft would go far from the Sun with a decreasing speed (as discussed above). In the third way, there is a *critical negative* sail-axis angle in HOF, say, α_{cr} such that for sail orientation angles $\alpha < \alpha_{cr}$ the sailcraft trajectory is characterized as follows:

1. the distance (from the Sun) first increases, achieves a local maximum at some point M, then decreases. The orbital angular momentum (per unit mass), say, \mathbf{H} of the sailcraft decreases in magnitude. It is suitable to define the scalar $H = \mathbf{H} \cdot \mathbf{k}$, where \mathbf{k} is the unit vector of the HIF Z-axis;
2. after a short time (few weeks or less, in general), the sailcraft speed $V = |\mathbf{V}|$ achieves a local minimum at a point P. H continues to decrease;
3. past P, the sailcraft speed increases because the **total** vector acceleration, say, \mathbf{A} begins by forming an acute angle with the vector velocity \mathbf{V} ; in mathematical terms, $dV / dt = \mathbf{A} \cdot \mathbf{V} / V > 0$. *This is the first key-point to realize;*
4. eventually, the sailcraft achieves a point Q where $H = 0$; here, the sailcraft's total energy (per unit mass), say, E (including the contribution of the solar pressure on the sail) shows a (negative) local minimum. *This is the second key-point;*
5. past Q, the sailcraft - keeping the negative value of the sail orientation - regains angular momentum by **reversing** its motion (that is \mathbf{H} is oriented down and $H < 0$). R keeps on decreasing while dV/dt augments. *This is the third key-point;*
6. the sailcraft energy continues to increase and a point S is reached where $E=0$, namely, the escape condition is satisfied; the sailcraft keeps on accelerating. *S is located before the perihelion.* The (negative) H continues to decrease;
7. if the sail attitude α has been chosen appropriately (about -25.9 deg in this example), the sailcraft flies-by the Sun at the desired (0.2 AU) perihelion, say, U; however, differently from a Keplerian orbit (for which the perihelion is the point of maximum speed), past the perihelion, V increases further while the sailcraft recedes from the Sun.
8. past U, the sailcraft is very fast and pass through a point, say, W of local maximum for the speed, since $\lambda < 1$. Thus, speed decreases but, at a few AU from the Sun (about 2.7 AU in this example), both the (positive) E and the (negative) H begin a plateau or cruise phase; V becomes practically

constant and, the most important thing, takes on a cruise value considerably higher than the speed of the circular orbit of the departure planet (the Earth, in this case). This example shows a cruise speed of 14.75 AU/yr or 69.9 km/s. At 100 AU, the sailcraft speed is 69.6 km/s.

H-reversal sun flyby trajectory

The figure below shows the mentioned sailcraft trajectory. Only the initial arc around the Sun has been plotted. The remaining part is rectilinear, in practice, and represents the cruise phase of the spacecraft. The sail is represented by a short segment with a central arrow that indicates its orientation. Note that the complicate change of sail direction in HIF is very simply achieved by a **constant** attitude in HOF. That brings about a net non-Keplerian feature to the whole trajectory.



- As mentioned in point-3, the strong sailcraft speed increase is due to **both** the solar-light thrust and gravity acceleration vectors. In particular, dV/dt , or the

along-track component of the total acceleration, is positive and particularly high from the point-Q to the point-U. This suggests that if a quick sail attitude maneuver is performed just before H vanishes, $\alpha \rightarrow -\alpha$, the sailcraft motion continues to be a **direct** motion with a final cruise velocity equal in magnitude to the reversal one (because the above maneuver keeps the perihelion value unchanged). The basic principle both sailing modes *share* may be summarised as follows: *a sufficiently light sailcraft needs to lose most of its initial energy for subsequently achieving the absolute maximum of energy compliant with its given technology.*

- The above 2D class of new trajectories represents an ideal case. The realistic 3D fast sailcraft trajectories are considerably more complicated than the 2D cases. However, the general feature of producing a fast cruise speed can be further enhanced. Some of the enclosed references contain strict mathematical algorithms for dealing with this topic. Recently (July 2005), in an international symposium an evolution of the above concept of fast solar sailing has been discussed. A sailcraft with $\sigma = 1 \text{ g/m}^2$ could achieve over 30 AU/yr (0.000474 c) in cruise (by keeping the perihelion at 0.2 AU), namely, well beyond the cruise speed of any nuclear-electric spacecraft (at least as conceived today). Such paper has been published on the Journal of the British Interplanetary Society (JBIS) in 2006.

Chapter-10

Space Shuttle External Tank



The ET from STS-124 after separation from Space Shuttle Discovery.

A **Space Shuttle External Tank (ET)** is the component of the Space Shuttle launch vehicle that contains the liquid hydrogen fuel and liquid oxygen oxidizer. During lift-off and ascent it supplies the fuel and oxidizer under pressure to the three space shuttle main engines (SSME) in the orbiter. The ET is jettisoned just over 10 seconds after MECO (Main Engine Cut Off), where the SSMEs are shut down, and re-enters the Earth's atmosphere. Unlike the Solid Rocket Boosters, external tanks have not been re-used. They break up before impact in the Indian Ocean (or Pacific Ocean in the case of direct-insertion launch trajectories, which are currently utilized) away from known shipping lanes. The tanks are not recovered.

There have been plans to use external tanks for other purposes, such as incorporation into a space station

Overview



STS-1 at liftoff. The External Tank was painted white for the first two Space Shuttle launches. From STS-3 on, it was left unpainted.

The ET is the largest element of the space shuttle, and when loaded, it is also the heaviest. It consists of three major components:

- the forward liquid oxygen (LOX) tank
- an unpressurized intertank that contains most of the electrical components
- the aft liquid hydrogen (LH₂) tank; this is the largest part, but it is relatively light, due to hydrogen's very low density.

The ET is the "backbone" of the shuttle during launch, providing structural support for attachment with the solid rocket boosters (SRBs) and orbiter. The tank is connected to each SRB at one forward attachment point (using a crossbeam through the intertank) and

one aft bracket, and it is connected to the orbiter at one forward attachment bipod and two aft bipods. In the aft attachment area, there are also umbilicals that carry fluids, gases, electrical signals and electrical power between the tank and the orbiter. Electrical signals and controls between the orbiter and the two solid rocket boosters also are routed through those umbilicals.

Versions

Over the years, NASA has worked to reduce the weight of the ET to increase overall efficiency. For each pound of weight reduction, the cargo-carrying capability of the shuttle spacecraft is increased almost one pound.

Standard Weight Tank

The original ET is informally known as the Standard Weight Tank (SWT). The first two, used for STS-1 and STS-2, were painted white to protect the tanks from ultraviolet light during the extended time that the shuttle spends on the launch pad prior to launch. Because this did not turn out to be a problem and in order to reduce weight, Lockheed Martin ceased painting the external tanks beginning with STS-3, leaving the rust-colored spray-on insulation bare, saving approximately 272 kg (600 lb) of weight.

After STS-4, several hundred pounds were eliminated by deleting the anti-geyser line. This line paralleled the oxygen feed line, providing a circulation path for liquid oxygen. This reduces accumulation of gaseous oxygen in the feed line during prelaunch tanking (loading of the LOX). After propellant loading data from ground tests and the first few space shuttle missions was assessed, the anti-geyser line was removed for subsequent missions. The total length and diameter of the ET remain unchanged. The last SWT tank, flown on STS-7, weighed approximately 35,000 kg (77,000 lb) inert.

Lightweight Tank



A Space Shuttle External Tank (ET) on its way to the Vehicle Assembly Building.

Beginning with the STS-6 mission, a lightweight ET (LWT), was introduced. This tank was used for the majority of the Shuttle flights, and was last used on the ill-fated Space Shuttle *Columbia* disaster (STS-107). Although tanks vary slightly in weight, each weighed approximately 30,000 kg (66,000 lb) inert.

The weight reduction from the SWT was accomplished by eliminating portions of stringers (structural stiffeners running the length of the hydrogen tank), using fewer stiffener rings and by modifying major frames in the hydrogen tank. Also, significant portions of the tank were milled differently to reduce thickness, and the weight of the ET's aft solid rocket booster attachments were reduced by using a stronger, yet lighter and less expensive titanium alloy.

Super Lightweight Tank

The Super Lightweight Tank (SLWT) was first flown in 1998 on STS-91 and has been used since with only two exceptions (STS-99 and STS-107). The SLWT is basically the same design as the LWT except that it uses an aluminum/lithium alloy (Al 2195) for a large part of the tank structure. This alloy provides a significant reduction in tank weight (~3,175 kg/7,000 lb) over the LWT. Although all ETs currently produced are of the SLWT configuration, one LWT remains in inventory and can be used if requested. The

SLWT provides 50% of the performance increase required for the shuttle to reach the International Space Station.



The barge carrying ET-119 is towed to Port Canaveral.

Technical specifications

SLWT Specifications

- Length: 153.8 ft (46.9 m)
- Diameter: 27.6 ft (8.4 m)
- Empty Weight: 58,500 lb (26,500 kg)
- Gross Liftoff Weight: 1,680,000 lb (760,000 kg)

LOX tank

- Length: 54.6 ft (16.6 m)
- Diameter: 27.6 ft (8.4 m)
- Volume (at 22 psig): 19,541.66 cu ft (146,181.8 US gal; 553,358 l)
- LOX mass (at 22 psig): 1,387,457 lb (629,340 kg)
- Operation Pressure: 20–22 psi (140–150 kPa) (gauge)

Intertank

- Length: 22.6 ft (6.9 m)
- Diameter: 27.6 ft (8.4 m)

LH₂ tank

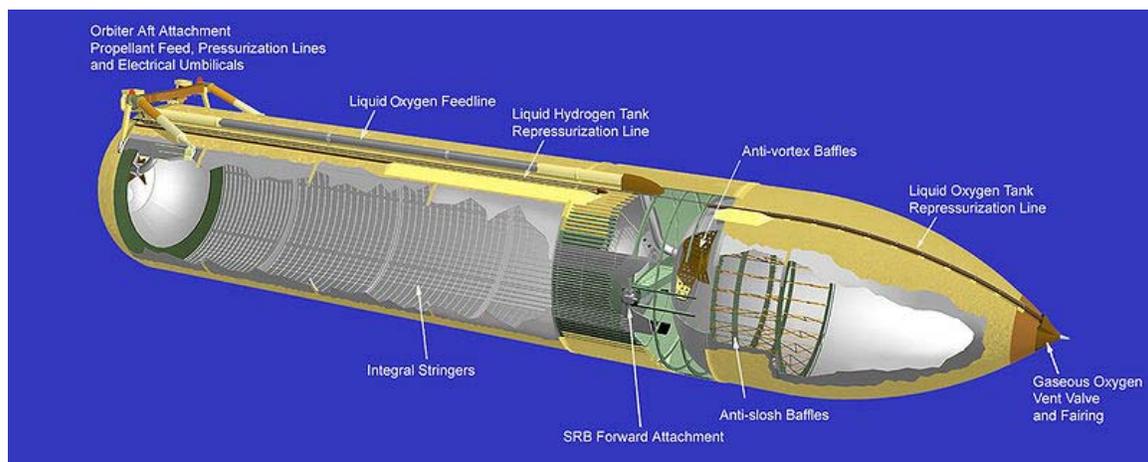
- Length: 97.0 ft (29.6 m)
- Diameter: 27.6 ft (8.4 m)
- Volume (at 29.3 psig): 52,881.61 cu ft (395,581.9 US gal; 1,497,440 l)
- LH₂ mass (at 29.3 psig): 234,265 lb (106,261 kg)
- Operation Pressure: 32–34 psi (220–230 kPa) (absolute)
- Operation Temperature: −423 °F (−252.8 °C)

Contractor

The contractor for the external tank is Lockheed Martin (previously Martin Marietta), New Orleans, Louisiana. The tank is manufactured at the Michoud Assembly Facility, New Orleans, and is transported to Kennedy Space Center by barge.

Components

The ET has three primary structures: an LOX tank, an intertank, and an LH₂ tank. Both tanks are constructed of aluminium alloy skins with support or stability frames as required. The intertank aluminium structure utilizes skin stringers with stabilizing frames. The primary aluminium materials used for all three structures are 2195 and 2090 alloys. AL 2195 is an Al-Li alloy designed by Lockheed Martin and Reynolds for storage of cryogenics. Al 2090 is a commercially available Al-Li alloy.



A cutaway diagram of External Tank.

Liquid oxygen tank

The LOX tank is located at the top of the ET and has an ogive shape to reduce aerodynamic drag and aerothermodynamic heating. The ogive nose section is capped by a flat removable cover plate and a nose cone. The nose cone consists of a removable conical assembly that serves as an aerodynamic fairing for the propulsion and electrical system components. The forward most element of the nose cone functions as a cast aluminium lightning rod. The LOX tank volume is 19,744 cu ft (559.1 m³) at 22 psig (250 kPa absolute) and -297 °F (90.4 K; -182.8 °C) (cryogenic).

The tank feeds into a 17 in (430 mm) diameter feed line that conveys the liquid oxygen through the intertank, then outside the ET to the aft right-hand ET/orbiter disconnect umbilical. The 17 in (430 mm) diameter feed line permits liquid oxygen to flow at approximately 2,787 lb/s (1264 kg/s) with the SSMEs operating at 104% or permits a maximum flow of 17,592 gal/min (1.1099 m³/s).

All loads except aerodynamic loads are transferred from the LOX tank at a bolted, flange-joint interface with the intertank.

The LOX tank also includes an internal slosh baffle and a vortex baffle to dampen fluid slosh. The vortex baffle is mounted over the LOX feed outlet to reduce fluid swirl resulting from slosh and to prevent entrapment of gases in the delivered LOX.

Intertank

The intertank is the ET structural connection which joins both the LOX and LH₂ tanks. Its primary functions are to receive and distribute all thrust loads from the SRBs and transfer loads between the tanks.

The two SRB forward attach fittings are located 180° apart on the intertank structure. A beam is extended across the intertank structure and is mechanically fastened to the attach fittings. When the SRBs are firing, the beam will flex due to high stress loads. These loads will be transferred to the fittings.

Adjoining the SRB attach fittings is a major ring frame. The loads are transferred from the fittings to the major ring frame which then distributes the tangential loads to the intertank skin. Two panels of the intertank skin, called the thrust panels, distribute the concentrated axial SRB thrust loads to the LOX and LH₂ tanks and to adjacent intertank skin panels. These adjacent panels are made up of six stringer-stiffened panels.

The intertank also functions as a protective compartment for housing the operational instrumentation.

Liquid hydrogen tank



The 70-foot (21 m)-long, 17-inch-diameter liquid oxygen feedline runs externally along the right side of the liquid hydrogen tank up and into the intertank. Two 5-inch (130 mm) diameter re-pressurization lines run beside it. One supplies hydrogen gas to the liquid hydrogen tank and the other supplies oxygen gas to the liquid oxygen tank. They are used to maintain the ullage pressure in each tank during the launch.

The LH₂ tank is the bottom portion of the ET. The tank is constructed of four cylindrical barrel sections, a forward dome, and an aft dome. The barrel sections are joined together by five major ring frames. These ring frames receive and distribute loads. The forward dome-to-barrel frame distributes the loads applied through the intertank structure and is also the flange for attaching the LH₂ tank to the intertank. The aft major ring receives orbiter-induced loads from the aft orbiter support struts and SRB-induced loads from the aft SRB support struts. The remaining three ring frames distribute orbiter thrust loads and LOX feedline support loads. Loads from the frames are then distributed through the barrel skin panels. The LH₂ tank has a volume of 53,488 cubic feet (1,514.6 m³) at 29.3 psig (3.02 bar absolute) and -423 °F (20.4 K; -252.8 °C) (cryogenic).

The forward and aft domes have the same modified ellipsoidal shape. For the forward dome, mounting provisions are incorporated for the LH₂ vent valve, the LH₂ pressurization line fitting, and the electrical feed-through fitting. The aft dome has a manhole fitting for access to the LH₂ feedline screen and a support fitting for the LH₂ feedline.

The LH₂ tank also has a vortex baffle to reduce swirl resulting from slosh and to prevent entrapment of gases in the delivered LH₂. The baffle is located at the siphon outlet just above the aft dome of the LH₂ tank. This outlet transmits the liquid hydrogen from the tank through a 17 inches (430 mm) line to the left aft umbilical. The liquid hydrogen feed line flow rate is 465 lb/s (211 kg/s) with the SSMEs at 104% or a maximum flow of 47,365 US gal/min (2.988 m³/s).

Thermal protection system



The Orbiter attachment hardware, liquid hydrogen umbilical connection (left), and liquid oxygen umbilical connection (right) are visible at the bottom of the tank.

The ET thermal protection system consists primarily of spray-on foam insulation (SOFI), plus preformed foam pieces and premolded ablator materials. The system also includes

the use of phenolic thermal insulators to preclude air liquefaction. Thermal isolators are required for liquid hydrogen tank attachments to preclude the liquefaction of air on exposed metal, and to reduce heat flow into the liquid hydrogen. While the warmer liquid oxygen results in fewer thermal requirements, the aluminum of the liquid oxygen tank forward areas require protection from aeroheating. Meanwhile insulation on the aft surfaces prevents liquified air from pooling in the intertank. The middle cylinder of the oxygen tank, and the propellant lines, could withstand the expected depths of frost accumulation condensed from humidity, but the orbiter could not take the damage from ice breaking free. The thermal protection system weighs 4,823 lb (2,188 kg).

Development of the ETs thermal protection system has been problematic. Anomalies in foam application were so frequent that they were treated as variances, not safety incidents. NASA has had difficulty preventing fragments of foam from detaching during flight for the entire history of the program:

- STS-1, 1981: Crew reports white material streaming past windows during orbiter-external-tank flight. Crew estimated sizes from 1/4-inch to fist-sized. Post-landing report describes probable foam loss of unknown location, and 300 tiles needing outright replacement due to various causes.
- STS-4, 1982: PAL ramp loss; 40 tiles require outright replacement.
- STS-5, 1982: Continued high rate of tile loss.
- STS-7, 1983: 50 by 30 cm (20 by 12 in) Bipod ramp loss photographed, dozens of spot losses.
- STS-27, 1988: One large loss of uncertain origin, causing one total tile loss. Hundreds of small losses.
- STS-32, 1990: Bipod ramp loss photographed; five spot losses up to 70 cm in diameter, plus tile damages.
- STS-50, 1992: Bipod ramp loss. 20×10×1 cm tile damage.
- STS-52, 1992: Portion of bipod ramp, jackpad lost. 290 total tile marks, 16 greater than an inch.
- STS-62, 1994: Portion of bipod ramp lost.

In 1995, chlorofluorocarbon-11 (CFC-11) began to be withdrawn from large-area, machine-sprayed foams in compliance with an Environmental Protection Agency ban on CFCs under section 610 of the Clean Air Act. In its place, a hydrochlorofluorocarbon known as HCFC-141b was certified for use and phased into the shuttle program. Remaining foams, particularly detail pieces sprayed by hand, continue to use CFC-11 to this day. These areas include the problematic bipod and PAL ramps, as well as some

fittings and interfaces. For the bipod ramp in particular, "the process of applying foam to that part of the tank had not changed since 1993." The "new" foam containing HCFC 141b was first used on the aft dome portion of ET-82 during the flight of STS-79 in 1996. Use of HCFC 141b was expanded to the ETs area, or larger portions of the tank, starting with ET-88, which flew on STS-86 in 1997.

During the lift-off of STS-107, a piece of foam insulation detached from one of the tank's bipod ramps and struck the leading edge of Space Shuttle *Columbia's* wing at a few hundred miles per hour. The impact is believed to have damaged one comparatively large reinforced carbon-carbon panel on the leading edge of the left wing, believed to be about the size of a basketball which then allowed super-heated gas to enter the wing superstructure several days later during re-entry. This resulted in the destruction of *Columbia* and the loss of its crew. The report determined that the external fuel tank, ET-93, "had been constructed with BX-250", a closeout foam whose blowing agent was CFC-11 and not the newer HCFC 141b.

In 2005, the problem of foam shed had not been fully cured; on STS-114, additional cameras mounted on the tank recorded a piece of foam separated from one of its Protuberance Air Load (PAL) ramps, which are designed to prevent unsteady air flow underneath the tank's cable trays and pressurization lines during ascent. The PAL ramps consist of manually sprayed layers of foam, and are more likely to become a source of debris. That piece of foam did not impact the orbiter.

Reports published concurrent with the STS-114 mission suggest that excessive handling of the ET during modification and upgrade may have contributed to the foam loss on *Discovery's* Return to Flight mission. However, three shuttle missions (STS-121, STS-115, and STS-116) have since been conducted, all with "acceptable" levels of foam loss. However on STS-118 a piece of foam (and/or ice) about 10 cm in diameter separated from a feedline attachment bracket on the tank, ricocheted off one of the aft struts and struck the underside of the wing, damaging two tiles. The damage was not considered dangerous.

Hardware



Space Shuttle *Discovery* before the scheduled launch of STS-116 in December 2006. Beneath *Discovery's* wings are the tail masts, which provide several umbilical connections to the orbiter, including a liquid-oxygen line through one and a liquid-hydrogen line through another. Seen above the golden external tank is the vent hood (known as the "beanie cap") at the end of the gaseous oxygen vent arm, extending from the fixed service structure. Vapor boils off the liquid oxygen in the external tank. The hood vents the oxygen vapor away from the Space Shuttle vehicle.

The external hardware, ET / orbiter attachment fittings, umbilical fittings, electrical and range safety system weigh 9,100 lb (4.1 t).

Vents and relief valves

Each propellant tank has a vent and relief valve at its forward end. This dual-function valve can be opened by ground support equipment for the vent function during prelaunch and can open during flight when the ullage (empty space) pressure of the liquid hydrogen tank reaches 38 psig (262 kPa) or the ullage pressure of the liquid oxygen tank reaches 25 psig (172 kPa).

The liquid oxygen tank contains a separate, pyrotechnically operated, propulsive tumble vent valve at its forward end. At separation, the liquid oxygen tumble vent valve is opened, providing impulse to assist in the separation maneuver and more positive control of the entry aerodynamics of the ET.

Each of the two aft external tank umbilical plates mate with a corresponding plate on the orbiter. The plates help maintain alignment among the umbilicals. Physical strength at the umbilical plates is provided by bolting corresponding umbilical plates together. When the orbiter GPCs command external tank separation, the bolts are severed by pyrotechnic devices.

The ET has five propellant umbilical valves that interface with orbiter umbilicals: two for the liquid oxygen tank and three for the liquid hydrogen tank. One of the liquid oxygen tank umbilical valves is for liquid oxygen, the other for gaseous oxygen. The liquid hydrogen tank umbilical has two valves for liquid and one for gas. The intermediate-diameter liquid hydrogen umbilical is a recirculation umbilical used only during the liquid hydrogen chill-down sequence during prelaunch.

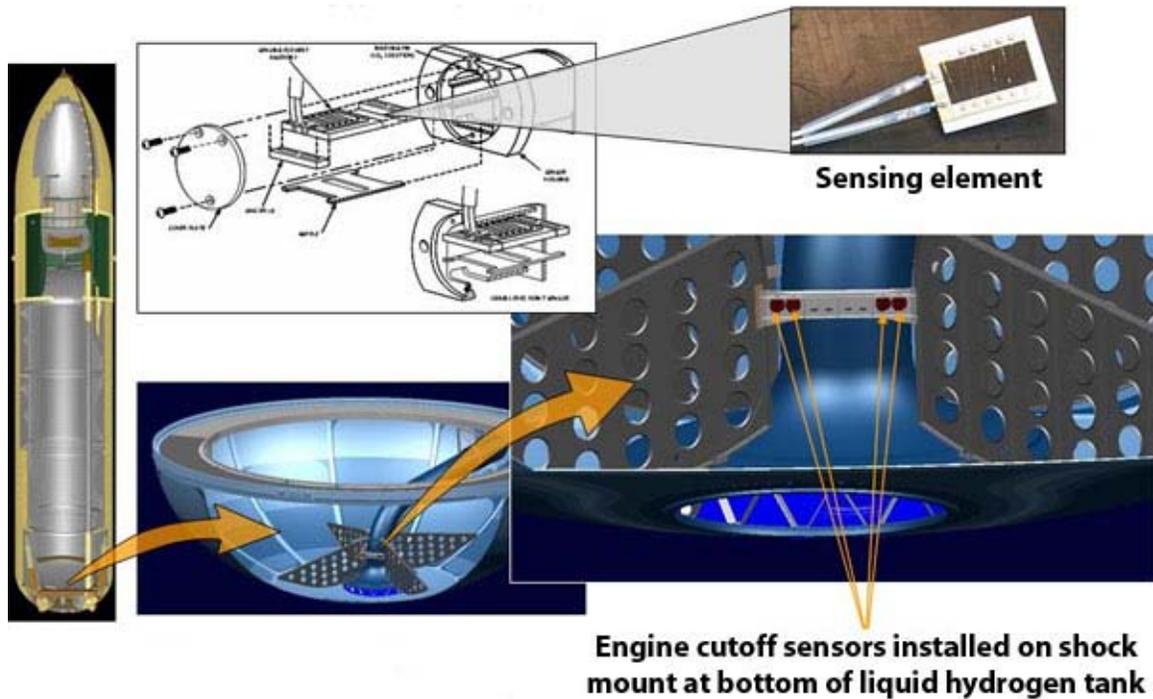


technicians inspect the GUCP following a scrub of STS-127 due to elevated Hydrogen levels at this connector

As the ET is filled, excess gaseous hydrogen is vented through umbilical connections over a large diameter pipe on an arm extended from the fixed service structure. The connection for this pipe between the ET and service structure is made at the ground umbilical carrier plate (GUCP). Sensors are also installed at the GUCP to measure Hydrogen levels. Countdowns of STS-80, STS-119, STS-127 and STS-133 have been halted and resulted in several week delays in the later cases due to hydrogen leaks at this connection. This requires complete draining of the tanks and removal of all hydrogen via helium gas purge, a 20 hour process, before technicians can inspect and repair problems.

A cap mounted to the swing-arm on the fixed service structure covers the oxygen tank vent on top of the ET during the countdown and is retracted about two minutes before lift-off. The cap siphons off oxygen vapor that threatens to form large ice on the ET, thus protecting the orbiter's thermal protection system during launch.

Sensors



The location of ECO sensors in the LH₂ tank.

There are eight propellant-depletion sensors, four each for fuel and oxidizer. The fuel-depletion sensors are located in the bottom of the fuel tank. The oxidizer sensors are mounted in the orbiter liquid oxygen feed line manifold downstream of the feed line disconnect. During SSME thrusting, the orbiter general-purpose computers constantly compute the instantaneous mass of the vehicle due to the usage of the propellants. Normally, main engine cutoff is based on a predetermined velocity; however, if any two of the fuel or oxidizer sensors sense a dry condition, the engines will be shut down.

The locations of the liquid oxygen sensors allow the maximum amount of oxidizer to be consumed in the engines, while allowing sufficient time to shut down the engines before the oxidizer pumps cavitate (run dry). In addition, 1,100 lb (500 kg) of liquid hydrogen are loaded over and above that required by the 6-1 oxidizer / fuel engine mixture ratio. This assures that cutoff from the depletion sensors is fuel-rich; oxidizer-rich engine shutdowns can cause burning and severe erosion of engine components, potentially leading to loss of the vehicle and crew.

Unexplained, erroneous readings from fuel depletion sensors have delayed several shuttle launch attempts, most notably STS-122. On 2007-12-18 a tanking test determined the cause of the errors to be a fault in a wiring connector, rather than a failure of the sensors themselves.

Four pressure transducers located at the top of the liquid oxygen and liquid hydrogen tanks monitor the ullage pressures.

The ET also has two electrical umbilicals that carry electrical power from the orbiter to the tank and the two SRBs and provide information from the SRBs and ET to the orbiter.

The ET has external cameras mounted in the brackets attached to the shuttle along with transmitters that can continue to send video data long after the shuttle and the ET have separated.

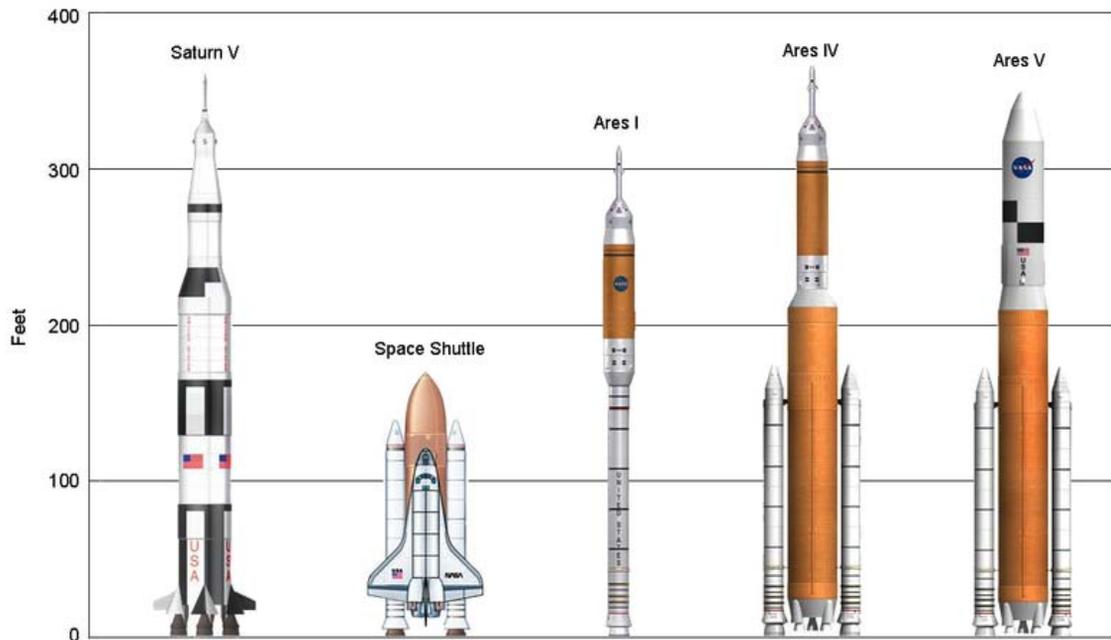
Range safety system

Earlier tanks incorporated a range safety system to disperse tank propellants if necessary. It included a battery power source, a receiver/decoder, antennas and ordnance. Starting with STS-79, this system was no longer used. The assembly was completely removed by the time STS-88 flew and has not been present on any tank since then. Consequently, it is no longer possible to destroy the vehicle during second stage ascent.

Variants

A cargo carrier addition was studied in 1984. This would consist of a cargo space mounted to the aft hydrogen dome, for cargoes greater than the payload bay diameter of 15 feet (4.6 m). It never flew.

Before the Challenger accident, proposed west coast launches by the military into polar orbits suffered a disadvantage in lifting capacity compared to low-inclination orbits. A booster module, derived from a Titan II first stage, was proposed. In addition to increasing capacity, the presence of tankage and rocket exhaust would have relieved heating on the aft hydrogen dome. This also failed to fly.



Comparison of the Saturn V, Space Shuttle, Ares I, Ares IV and Ares V.

The Ares I second stage, on the other hand, will only use the spray-on insulation foam currently used on the current ET. Originally configured like that of the Ares V and the Shuttle ET, NASA, upon completing its design review in 2006, decided, in order to save weight and costs, to reconfigure the internal structure of the second stage by using a combined LH₂/LOX tank with the propellants separated by a common bulkhead, a configuration successfully used on the S-II and S-IVB stages of the Saturn V rocket. Unlike the Ares V, which will use the same fill/drain/vent configuration used on the Shuttle, the Ares I system will utilize a traditional fill/drain/vent system used on the Saturn IB and Saturn V rockets, but with quick-retracting arms due to the "leap frog" speed the Ares I will expect upon SRB ignition.

As originally envisioned, both the Ares I and Ares V would have used a modified "throw away" version of the SSME, but in due course, because of the need to keep R&D costs down and to maintain a schedule set by NASA Administration Michael D. Griffin to launch the Ares and Orion by 2011, NASA decided to switch to the RS-68 engine for the Ares V and to an uprated J-2 engine for the Ares I. Because of the switch to the RS-68, the Ares V was widened from 28.6 to 33 feet (8.72 to 10.06 m) to accommodate the extra propellants, while the Ares I was reconfigured to incorporate a fifth solid-rocket segment as the J-2X, as the rocket engine is known, has less thrust than the SSME. Because of the trade-off, NASA would save an estimated USD \$35 million by using simplified, higher thrust RS-68 engines (reconfigured to fire and perform like the SSME), while at the same time, eliminate the costly tests needed for an air-startable SSME for the Ares I (as the J-2X and its predecessor were designed to be started in both mid-air and in a near vacuum).

The DIRECT project, a proposed alternative shuttle-derived vehicle, uses a modified, standard diameter, external tank with three SSMEs, with two standard SRBM, as a Crew Launch Vehicle. The same vehicle, with one extra SSME, and an EDS upper stage, serves as the Cargo Launch Vehicle. It is purported to save \$16 billion, eliminate NASA job losses, and reduce the post-shuttle, manned spaceflight gap from five plus years to two or less.

Unflown hardware

The last two scheduled Shuttle missions, the STS-134 and STS-135/335 will utilize ET-122 (refurbished after Katrina damage, fully certified and flight-worthy) and ET-138 (the last completely finished ET).

ET-94 (older version LWT), currently in storage at Michoud Assembly Facility, will be used for development and tests of in-line Shuttle-Derived Launch Vehicle, the Space Launch System.

Three other external tanks were in preparation, when the manufacturing stopped. ET-139 is at advanced stage of manufacturing; ET-140 and ET-141 are in early stages of manufacturing.

Chapter-11

Spacecraft Magnetometer

Spacecraft magnetometers are magnetometers used aboard spacecraft and satellites, mostly for scientific investigations. Magnetometers are among the most widely used scientific instruments in exploratory and observation satellites. These instruments were instrumental in the discovery of the Van Allen radiation belts around Earth by Explorer 1, and have detailed the magnetic fields of the Earth, Moon, Sun, Mars, Venus and other planets. There are ongoing missions using magnetometers, including attempts to define the shape and activity of Saturn's core.

The first spacecraft-borne magnetometer was placed on the Sputnik 3 spacecraft in 1958 and the most detailed magnetic observations of the Earth have been performed by the Magsat and Ørsted satellites. Magnetometers were taken to the Moon during the later Apollo missions. Many instruments have been used to measure the strength and direction of magnetic field lines around Earth and the solar system.

Spacecraft magnetometers basically fall into three categories: fluxgate, search-coil and ionized gas magnetometers. The most accurate magnetometer complexes on spacecraft contain two separate instruments, with a helium ionized gas magnetometer used to calibrate the fluxgate instrument for more accurate readings. Many later magnetometers contain small ring-coils oriented at 90° in two dimensions relative to each other forming a triaxial framework for indicating direction of magnetic field.

Magnetometer types

Magnetometers for non-space use evolved from the 19th to mid-20th centuries, and were first employed in spaceflight by Sputnik 3 in 1958. A main constraint on magnetometers used in space is the availability of energy and weight. Magnetometers have fallen into 3 major categories: the fluxgate type, search coil and the ionized vapor magnetometers. The newest type is the Obeurhauser type based on nuclear magnetic resonance technology.

Fluxgate magnetometers



magnetometers are mounted at both ends of the solar panel assemblies to isolate them from the spacecraft's magnetic fields

Fluxgate magnetometers are used for their electronic simplicity and low weight. There have been several types of fluxgate used in spacecraft, which vary in two regards. Primarily better readings are obtained with three magnetometers, each pointing in a different direction. Some spacecraft have instead achieved this by rotating the craft and taking readings at 120° intervals, but this creates other issues. The other difference is in the configuration, which is simple and circular.

Magnetometers of this type were equipped on the "Pioneer 0"/Able 1, "Pioneer 1"/Able 2, Ye1.1, Ye1.2, and Ye1.3 missions that failed in 1958 due to launch problems. The

Pioneer 1 however did collect data on the Van Allen belts. In 1959 the Soviet "Luna 1"/Ye1.4 carried a three-component magnetometer that passed the moon en-route to a heliocentric orbit at a distance of 6,400 miles (10,300 km), but the magnetic field could not be accurately assessed. Eventually the USSR managed a lunar impact with "Luna 2", a three component magnetometer, finding no significant magnetic field in close approach to the surface. Explorer 10 had an abbreviated 52 hr mission with two fluxgate magnetometers on board. During 1958 and 1959 failure tended to characterize missions carrying magnetometers: 2 instruments were lost on Able IVB alone. In early 1966 the USSR finally placed Luna 10 in orbit around the moon carrying a magnetometer and was able to confirm the weak nature of the moon's magnetic field. Venera 4, 5, and 6 also carried magnetometers on their trips to Venus, although they were not placed on the landing craft.



Lunar prospector satellite, the magnetometer is mounted on the boom-end facing toward the viewer

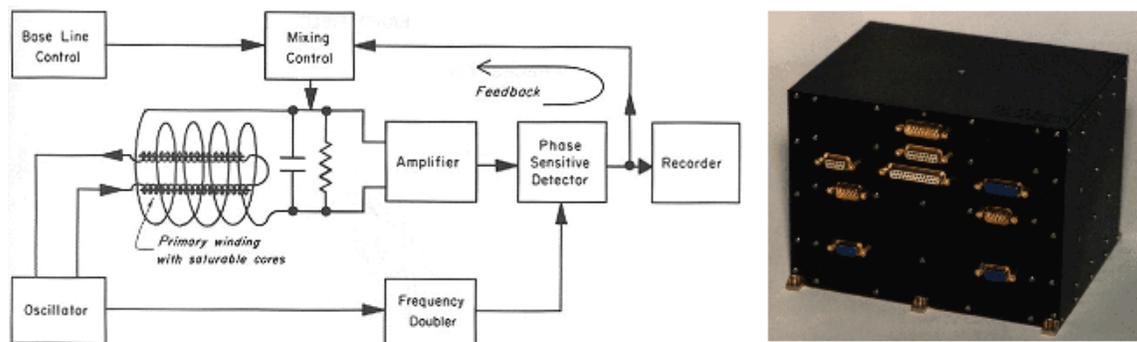
Vector sensors

The majority of early fluxgate magnetometers on spacecraft were made as vector sensors. However, the magnetometer electronics created harmonics which interfered with readings. Properly designed sensors had feedback electronics to the detector that effectively neutralized the harmonics. Mariner 1 and Mariner 2 carried fluxgate-vector sensor devices. Only Mariner 2 survived launch and as it passed Venus on December 14, 1962 it failed to detect a magnetic field around the planet. This was in part due to the

distance of the spacecraft from the planet, noise within the magnetometer, and a very weak Venusian magnetic field. Pioneer 6, launched in 1965, is one of 4 Pioneer satellites circling the sun and relaying information to Earth about solar winds. This spacecraft was equipped with a single vector-fluxgate magnetometer.

Ring core and spherical

Ring core sensor fluxgate magnetometers began replacing vector sensor magnetometers with the Apollo 16 mission in 1972, where a three axis magnetometer was placed on the moon. These sensors were used on a number of satellites including Magsat, Voyager, Ulysses, Giotto, AMPTE. The Lunar Prospector-1 uses ring-coil made of these alloys extended away from each other and its spacecraft to look for remnant magnetism in the moon's 'non-magnetic' surface.



Wiring diagram and picture of the Magnetometer used on Mars Global Surveyor

Properly configured the magnetometers are capable of measuring magnetic field difference of 1 nT. These devices, with cores about 1 cm in size, were of lower weight than vector sensor. However, these devices were found to have non-linear output with magnetic fields greater than >5000 nT. Later it was discovered that creating a spherical structure with feedback loops wire transverse to the ring in the sphere could negate this effect. These later magnetometers were called spherical fluxgate or compact spherical core (CSC) magnetometers used in the Ørsted satellite. The metal alloys that form the core of these magnetometers has also improved since Apollo-16 mission with latest using advanced molybdenum-pemalloy alloys, producing lower noise with more stable output.



Photograph of the search coil magnetometers used on the THEMIS and Cluster/Staff mission.

Search-coil magnetometer

Search-coil magnetometers are wound coils around a core of high magnetic permeability. Search coils concentrate magnetic field lines inside the core along with fluctuations. The benefit of these magnetometers is that they operate on alternating current and so can resolve changes in magnetic fields quickly, many times per second. The Pioneer 5 mission finally managed to get a working magnetometer of this type in orbit around the sun showing that magnetic fields existed between Earth and Venus orbits. A single magnetometer was oriented along the plane perpendicular to the spin axis of the space craft. Search coil magnetometers have become increasingly more common in Earth observation satellites. A commonly used instrument is the triaxial search-coil magnetometer. Orbiting Geophysical Observatory (OGO missions - OGO-1 to OGO-6) The Vela (satellite) mission used this type as part of a package to determine if nuclear weapons evaluation was being conducted outside earth's atmosphere. In September 1979 a Vela satellite collected evidence of a potential nuclear burst over the South Western Indian Ocean. In 1997 the US created the FAST that was designed to investigate aurora phenomena over the poles. And currently it is investigating magnetic fields at 10 to 30 Earth radii with the THEMIS satellites THEMIS, which stands for *Time History of Events and Macroscale Interactions during Substorms* is an array of five satellites which hope to gather more precise history of how magnetic storms arise and dissipate.

Ionized gas magnetometers

Heavy metal — scalar

Certain spacecraft, like Magsat are equipped with **scalar magnetometer**. The output of these device, often in out frequency, is proportional to the magnetic field. The Magsat and Grm-A1 had **cesium-vapor** (cesium-133) sensor heads of dual-cell design, this design left two small dead zones. Explorer 10 (P14) was equipped with a rubidium vapor magnetometer, presumably a scalar magnetometer since the spacecraft also had a fluxgate. The magnetometer was fouled accidentally which caused it to overheat, it worked for a period of time but 52 h into the mission transmission went dead and was not regained. Ranger 1 and 2 carried a rubidium vapor magnetometer, failed to reach lunar orbit.

Helium

This type of magnetometer depends on the variation in helium absorptivity, when excited, polarized infrared light with an applied magnetic field. A low field vector-helium magnetometer was equipped on the Mariner 4 spacecraft to Mars like the Venus probe a year earlier, no magnetic field was detected. Mariner 5 used a similar device For this experiment a low-field helium magnetometer was used to obtain triaxial measurements of interplanetary and Venusian magnetic fields. Similar in accuracy to the triaxial flux-gated magnetometers this device produced more reliable data.

Other types

Overhauser magnetometer provides extremely accurate measurements of the strength of the magnetic field. The Orsted (satellite) uses this type of magnetometer to map the magnetic fields over the surface of the earth.

On the Vanguard 3 mission (1959) a **proton processional magnetometer** was used to measure geomagnetic fields. The proton source was hexane.

Configurations of magnetometers

Unlike ground based magnetometers that can be oriented by the user to determine the direction of magnetic field, in space the user is linked by telecommunications to a satellite traveling at 25,000 km per hour. The magnetometers used need to give an accurate reading quickly to be able to deduce magnetic fields. Several strategies can be employed, it is easier to rotate a space craft about its axis than to carry the weight of an additional magnetometer. Other strategy is to increase the size of the rocket, or make the magnetometer lighter and more effective. One of the problems, for example in studying planets with low magnetic fields like Venus, does require more sensitive equipment. The equipment has necessarily needed to evolve for today's modern task. Ironically satellites launched more the 20 years ago still have working magnetometers in places where it

would take decades to reach today, at the same time the latest equipment is being used to analyze changes in the Earth here at home.

Uniaxial

These simple fluxgate magnetometers were used on many missions. On Pioneer 6 and Injun 1 the magnetometers were mounted to a bracket external to the space craft and readings were taken as the spacecraft rotated every 120°. Pioneer 7 and Pioneer 8 are configured similarly. The fluxgate on Explorer 6 was mounted along the spin axis to verify spacecraft tracking magnetic field lines. Search coil magnetometers were used on Pioneer 1, Explorer 6, Pioneer 5, and Deep Space 1.

Di axial

A two axis magnetometer was mounted to the ATS-1 (Applications Technology Satellite). One sensor was on a 15 cm boom and the other on the spacecrafts spin axis (Spin stabilized satellite). The sun was used to sense the position of the boom mounted device, and triaxial vector measurements could be calculated. Compared to other boom mounted magnetometers, this configuration had considerable interference. Interestingly with this spacecraft, the sun induce magnetic oscillations and this allowed the continued use of the magnetometer after the sun sensor failed. Explorer 10 had two fluxgate magnetometers but is technically classified as a dual technique since it also had a rubidium vapor magnetometer.

Triaxial

The Sputnik-3 had a **vector fluxgate** magnetometer, however because the orientation of the spacecraft could not be determined the direction vector for the magnetic field could not be determined. Three axis magnetometers were used on Luna 1, Luna 2, Pioneer Venus, Mariner 2, Venera 1, Explorer 12, Explorer 14, and Explorer 15. Explorer 33 was 'to be' the first US spacecraft to enter stable orbit around the moon was equipped with the most advanced magnetometer, a boom-mounted triaxial fluxgate (GFSC) magnetometer of the early-vector type. It had a small range but was accurate to a resolution of 0.25 nT. However after a rocket failure it was left in a highly elliptical orbit around Earth that orbited through the electro/magnetic tail.

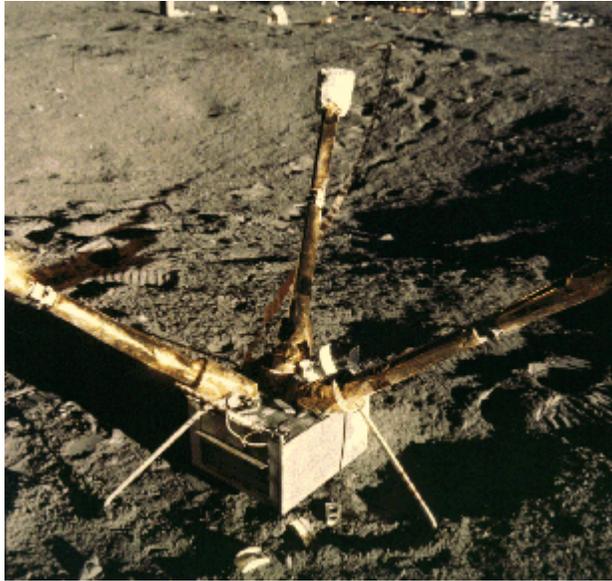
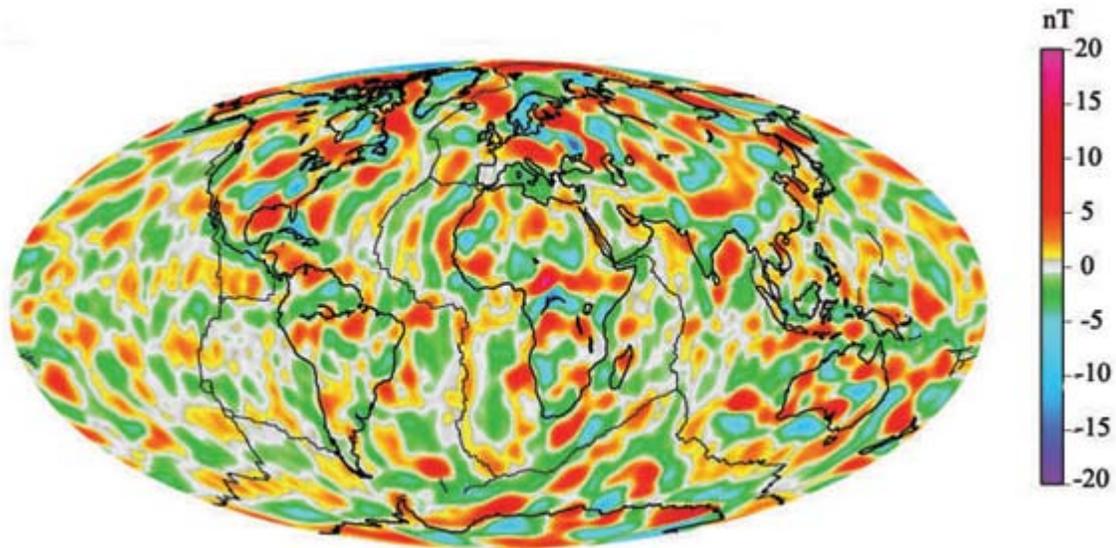


Image of the lunar stationed magnetometer as part of the ALSEP package

The Pioneer 9 and Explorer 34 used a configuration similar to Explorer 33 to survey the magnetic field within Earth's solar orbit. Explorer 35 was the first of its type to enter stable orbit around the moon, this proved important because with the sensitive triaxial magnetometer on board, it was found the moon effectively had no magnetic field, no radiation belt, and solar winds directly impacted the moon. Lunar Prospector surveyed for surface magnetism around the moon (1998–99), using the triaxial (extended) magnetometers. With Apollo 12 improved magnetometers were placed on the moon as part of the Lunar Module/Apollo Lunar Surface Experiments Package

(ALSEP). The magnetometer continued to work several months after that return module departed. As part of the Apollo 14 ALSEP, there was a portable magnetometer.

The first use of the three axis ring-coil magnetometer was on the Apollo 16 moon mission. Subsequently it was used on the Magsat. The MESSENGER mission has triaxial ring-coil magnetometer with a range of ± 1000 mT and a sensitivity of 0.02 mT, still in progress, the mission is designed to get detailed information about Mercurian magnetosphere. The first use of spherical magnetometer in three axis configuration was on the Orsted (satellite).



Modeled Earth magnetic fields, data created by satellites with sensitive magnetometers

Dual technique

Each type of magnetometer has its own built in 'weakness'. This can result from the design of the magnetometer to the way the magnetometer interacts with the spacecraft, radiation from the sun, resonances, etc. Using completely different design is a way to measure which readings are the result of natural magnetic fields and the sum of magnetic fields altered by spacecraft systems. In addition each type has its strengths. The fluxgate type is relatively good at providing data that finds magnetic sources. One of the first Dual technique systems was the abbreviated Explorer 10 mission which used a rubidium vapor and biaxial fluxgate magnetometers. Vector helium is better at tracking magnetic field lines and as a scalar magnetometer. Cassini spacecraft used a **Dual Technique Magnetometer**. One of these devices is the ring-coil vector fluxgate magnetometer (RCFGM). The other device is a vector/scalar helium magnetometer. The RCFGM is mounted 5.5 m out on an 11 m boom with the helium device at the end.

Explorer 6 (1959) used a search coil magnetometer to measure the gross magnetic field of the Earth and vector fluxgate., however because of induced magnetism in the space craft the fluxgate sensor became saturated and did not send data. Future missions would attempt to place magnetometers further away from the space craft.

Magsat Earth geological satellite was also Dual Technique. This satellite and Grm-A1 carried a scalar cesium vapor magnetometer and vector fluxgate magnetometers. The Grm-A1 satellite carried the magnetometer on 4 meter boom. This particular spacecraft was designed to hold in a precise equi-gravitational orbit, while taking measurements. For purposes similar to Magsat, the Ørsted satellite, also used a dual technique system. The Overhauser magnetometer is situated at the end of an 8 meter long boom, in order to minimize disturbances from the satellite's electrical systems. The CSC fluxgate magnetometer is located inside the body and associated with a star tracking device. One

of the greater accomplishments of the two missions, the Magsat and Orsted missions happen to capture a period of great magnetic field change, with the potential of a loss of dipole, or pole reversal.

Chapter-12

Rocket Propellant

Rocket propellant is mass that is stored in some form of propellant tank, prior to being used as the propulsive mass that is ejected from a rocket engine in the form of a fluid jet to produce thrust. A fuel propellant is often burned with an oxidizer propellant to produce large volumes of very hot gas. These gases expand and push on a nozzle, which accelerates them until they rush out of the back of the rocket at extremely high speed, making thrust. Sometimes the propellant is not burned, but can be externally heated for more performance. For smaller attitude control thrusters, a compressed gas escapes the spacecraft through a propelling nozzle.

Chemical rocket propellants are most commonly used, which undergo exothermic chemical reactions which produce hot gas which is used by a rocket for propulsive purposes.

In ion propulsion, the propellant is made of electrically charged atoms (ions), which are electromagnetically pushed out of the back of the spacecraft. Magnetically accelerated ion drives are not usually considered to be rockets however, but a similar class of thrusters use electrical heating and magnetic nozzles.

propellant, 10% structure, and hence a mass ratio of 10:1 . The impulse delivered by the motor to the rocket vehicle per weight of fuel consumed is often reported as the rocket propellant's *specific impulse*. A propellant with a higher specific impulse is said to be more efficient because more thrust is produced while consuming a given amount of propellant.

Lower stages will usually use high-density (low volume) propellants because of their lighter tankage to propellant weight ratios and because higher performance propellants require higher expansion ratios for maximum performance than can be attained in atmosphere. Thus, the Apollo-Saturn V first stage used kerosene-liquid oxygen rather than the liquid hydrogen-liquid oxygen used on its upper stages Similarly, the Space Shuttle uses high-thrust, high-density solid rocket boosters for its lift-off with the liquid hydrogen-liquid oxygen SSMEs used partly for lift-off but primarily for orbital insertion.

Chemical propellants

There are three main types of propellants: solid, liquid, and hybrid.

Solid propellants

History

The earliest rockets were created hundreds of years ago by the Chinese, and were used primarily for fireworks displays and as weapons. They were fueled with black powder, a type of gunpowder consisting of a mixture of charcoal, sulfur and potassium nitrate (saltpeter). Rocket propellant technology did not advance until the end of the 19th century, by which time smokeless powder had been developed, originally for use in firearms and artillery pieces. Smokeless powders and related compounds have seen use as double-base propellants.

Description

Solid propellants (and almost all rocket propellants) consist of an oxidizer and a fuel. In the case of gunpowder, the fuel is charcoal, the oxidizer is potassium nitrate, and sulfur serves as a catalyst. (Note: sulfur is not a true catalyst in gunpowder as it is consumed to a great extent into a variety of reaction products such as K_2S . The sulfur acts mainly as a sensitizer lowering threshold of ignition.) During the 1950s and 60s researchers in the United States developed what is now the standard high-energy solid rocket fuel, Ammonium Perchlorate Composite Propellant (APCP). This mixture is primarily ammonium perchlorate powder (an oxidizer), combined with fine aluminium powder (a fuel), held together in a base of PBAN or HTPB (rubber-like fuels). The mixture is formed as a liquid, and then cast into the correct shape and cured into a rubbery solid.

Advantages

Solid-fueled rockets are much easier to store and handle than liquid-fueled rockets, which makes them ideal for military applications. In the 1970s and 1980s the U.S. switched entirely to solid-fueled ICBMs: the LGM-30 Minuteman and LG-118A Peacekeeper (MX). In the 1980s and 1990s, the USSR/Russia also deployed solid-fueled ICBMs (RT-23, RT-2PM, and RT-2UTTH), but retains two liquid-fueled ICBMs (R-36 and UR-100N). All solid-fueled ICBMs on both sides have three initial solid stages and a precision maneuverable liquid-fueled bus used to fine tune the trajectory of the reentry vehicle.

Their simplicity also makes solid rockets a good choice whenever large amounts of thrust are needed and cost is an issue. The Space Shuttle and many other orbital launch vehicles use solid-fueled rockets in their first stages (solid rocket boosters) for this reason.

Disadvantages

Relative to liquid fuel rockets, solid rockets have a number of disadvantages. Solid rockets have a lower specific impulse than liquid-fueled rockets. It is also difficult to build a large mass ratio solid rocket because almost the entire rocket is the combustion chamber, and must be built to withstand the high combustion pressures. If a solid rocket is used to go all the way to orbit, the payload fraction is very small. (For example, the Orbital Sciences Pegasus rocket is an air-launched three-stage solid rocket orbital booster. Launch mass is 23,130 kg, low earth orbit payload is 443 kg, for a payload fraction of 1.9%. Compare to a Delta IV Medium, 249,500 kg, payload 8600 kg, payload fraction 3.4% without air-launch assistance.)

A drawback to solid rockets is that they cannot be throttled in real time, although a predesigned thrust schedule can be created by altering the interior propellant geometry.

Solid rockets can often be shut down before they run out of fuel. Essentially, the rocket is vented or an extinguishant injected so as to terminate the combustion process. In some cases termination destroys the rocket, and then this is typically only done by a Range Safety Officer if the rocket goes awry. The third stages of the Minuteman and MX rockets have precision shutdown ports which, when opened, reduce the chamber pressure so abruptly that the interior flame is blown out. This allows a more precise trajectory which improves targeting accuracy.

Finally, casting very large single-grain rocket motors has proved to be a very tricky business. Defects in the grain can cause explosions during the burn, and these explosions can increase the burning propellant surface enough to cause a runaway pressure increase, until the case fails.

Liquid propellants

History

Though early rocket theorists, such as Konstantin Tsiolkovsky, proposed liquid hydrogen and liquid oxygen as propellants, the first liquid-fueled rocket, launched by Robert Goddard on March 16, 1926, used gasoline and liquid oxygen. Liquid hydrogen was first used by the engines designed by Pratt and Whitney for the Lockheed CL-400 Suntan reconnaissance aircraft in the mid-1950s. In the mid-1960s, the Centaur and Saturn upper stages were both using liquid hydrogen and liquid oxygen.

The highest specific impulse chemistry ever test-fired in a rocket engine was lithium and fluorine, with hydrogen added to improve the exhaust thermodynamics (making this a tripropellant). The combination delivered 542 seconds (5.32 kN·s/kg, 5320 m/s) specific impulse in a vacuum. The impracticality of this chemistry highlights why exotic propellants are not actually used: to make all three components liquids, the hydrogen must be kept below $-252\text{ }^{\circ}\text{C}$ (just 21 K) and the lithium must be kept above $180\text{ }^{\circ}\text{C}$ (453 K). Lithium and fluorine are both extremely corrosive, liquid lithium ignites on contact with air, fluorine ignites on contact with most fuels, and hydrogen, while not hypergolic, is an explosive hazard. Fluorine and the hydrogen fluoride (HF) in the exhaust are very toxic, which damages the environment, makes work around the launch pad difficult, and makes getting a launch license that much more difficult. The rocket exhaust is also ionized, which would interfere with radio communication with the rocket.

Current Types

The most common liquid propellants in use today:

- LOX and kerosene (RP-1). Used for the lower stages of most Russian and Chinese boosters, the first stages of the Saturn V and Atlas V, and all stages of the developmental Falcon 1 and Falcon 9. Very similar to Robert Goddard's first rocket. This combination is widely regarded as the most practical for boosters that lift off at ground level and therefore must operate at full atmospheric pressure.
- LOX and liquid hydrogen, used in the Space Shuttle orbiter, the Centaur upper stage of the Atlas V, Saturn V upper stages, the newer Delta IV rocket, the H-IIA rocket, and most stages of the European Ariane rockets.
- Nitrogen tetroxide (N_2O_4) and hydrazine (N_2H_4), MMH, or UDMH. Used in military, orbital and deep space rockets, because both liquids are storable for long periods at reasonable temperatures and pressures. N_2O_4 /UDMH is the main fuel for the Proton rocket. This combination is hypergolic, making for attractively simple ignition sequences. The major inconvenience is that these propellants are highly toxic, hence they require careful handling.
- Monopropellants such as hydrogen peroxide, hydrazine and nitrous oxide are primarily used for attitude control and spacecraft station-keeping where their

long-term storability, simplicity of use and ability to provide the tiny impulses needed, outweighs their lower specific impulse as compared to bipropellants. Hydrogen peroxide is also used to drive the turbopumps on the first stage of the Soyuz launch vehicle.

Historical propellants

These include propellants such as syntin, which is an expensive high energy hydrocarbon fuel which was used on Soyuz U2 until 1995.

Advantages

Liquid fueled rockets have higher specific impulse than solid rockets and are capable of being throttled, shut down, and restarted. Only the combustion chamber of a liquid fueled rocket needs to withstand combustion pressures and temperatures and they can be regeneratively cooled by the liquid propellant. On vehicles employing turbopumps, the propellant tanks are at very much less pressure than the combustion chamber, and thus can be built far more lightly than a solid propellant rocket case, permitting a higher mass ratio. For these reasons, most orbital launch vehicles use liquid propellants.

The primary performance advantage of liquid propellants is due to the oxidizer. Several practical liquid oxidizers (liquid oxygen, nitrogen tetroxide, and hydrogen peroxide) are available which have much better specific impulse than the ammonium perchlorate used in most solid rockets, when paired with comparable fuels. These facts have led to the use of hybrid propellants: a storable oxidizer used with a solid fuel, which retain most virtues of both liquids (high ISP) and solids (simplicity).

While liquid propellants are cheaper than solid propellants, for orbital launchers, the cost savings do not, and historically have not mattered; the cost of propellant is a very small portion of the overall cost of the rocket.

Disadvantages

The main difficulties with liquid propellants are also with the oxidizers. These are generally at least moderately difficult to store and handle due to their high reactivity with common materials, may have extreme toxicity (nitric acids), moderately cryogenic (liquid oxygen), or both (liquid fluorine, FLOX- a fluorine/LOX mix). Several exotic oxidizers have been proposed: liquid ozone (O₃), ClF₃, and ClF₅, all of which are unstable, energetic, and toxic.

Liquid fueled rockets also require potentially troublesome valves and seals and thermally stressed combustion chambers, which increase the cost of the rocket. Many employ specially designed turbopumps which raise the cost enormously due to difficult fluid flow patterns that exist within the casings.

Gas propellants

A gas propellant usually involves some sort of compressed gas. However, due to the low density and high weight of the pressure vessel, gases see little current use, but are sometimes used for vernier engines, particularly with inert propellants.

GOX was used as one of the propellant for the Buran program for the orbital manoeuvring system.

Hybrid propellants

A hybrid rocket usually has a solid fuel and a liquid or gas oxidizer. The fluid oxidizer can make it possible to throttle and restart the motor just like a liquid fueled rocket. Hybrid rockets are also cleaner than solid rockets because practical high-performance solid-phase oxidizers all contain chlorine, versus the more benign liquid oxygen or nitrous oxide used in hybrids. Because just one propellant is a fluid, hybrids are simpler than liquid rockets.

Hybrid motors suffer two major drawbacks. The first, shared with solid rocket motors, is that the casing around the fuel grain must be built to withstand full combustion pressure and often extreme temperatures as well. However, modern composite structures handle this problem well, and when used with nitrous oxide and a solid rubber propellant (HTPB), relatively small percentage of fuel is needed anyway, so the combustion chamber is not especially large.

The primary remaining difficulty with hybrids is with mixing the propellants during the combustion process. In solid propellants, the oxidizer and fuel are mixed in a factory in carefully controlled conditions. Liquid propellants are generally mixed by the injector at the top of the combustion chamber, which directs many small swift-moving streams of fuel and oxidizer into one another. Liquid fueled rocket injector design has been studied at great length and still resists reliable performance prediction. In a hybrid motor, the mixing happens at the melting or evaporating surface of the fuel. The mixing is not a well-controlled process and generally quite a lot of propellant is left unburned, which limits the efficiency and thus the exhaust velocity of the motor. Additionally, as the burn continues, the hole down the center of the grain (the 'port') widens and the mixture ratio tends to become more oxidiser rich.

There has been much less development of hybrid motors than solid and liquid motors. For military use, ease of handling and maintenance have driven the use of solid rockets. For orbital work, liquid fuels are more efficient than hybrids and most development has concentrated there. There has recently been an increase in hybrid motor development for nonmilitary suborbital work:

- The Reaction Research Society, although known primarily for their work with liquid rocket propulsion, has a long history of research and development with hybrid rocket propulsion.

- Several universities have recently experimented with hybrid rockets. Brigham Young University, the University of Utah and Utah State University launched a student-designed rocket called Unity IV in 1995 which burned the solid fuel hydroxy-terminated polybutadiene (HTPB) with an oxidizer of gaseous oxygen, and in 2003 launched a larger version which burned HTPB with nitrous oxide. Stanford University researches nitrous-oxide/paraffin hybrid motors.
- The Rochester Institute of Technology was building a HTPB hybrid rocket to launch small payloads into space and to several near Earth objects. Its first launch was scheduled for Summer 2007.
- Scaled Composites SpaceShipOne, the first private manned spacecraft, is powered by a hybrid rocket burning HTPB with nitrous oxide. The hybrid rocket engine was manufactured by SpaceDev. SpaceDev partially based its motors on experimental data collected from the testing of AMROC's (American Rocket Company) motors at NASA's Stennis Space Center's E1 test stand. Motors ranging from as small as 1000 lbf (4.4 kN) to as large as 250,000 lbf (1.1 MN) thrust were successfully tested. SpaceDev purchased AMROCs assets after the company was shut down for lack of funding.

Inert propellants

Some rocket designs have their propellants obtain their energy from non chemical or even external sources. For example water rockets use the compressed gas, typically air, to force the water out of the rocket.

Solar thermal rockets and Nuclear thermal rockets typically propose to use liquid hydrogen for an I_{sp} (Specific Impulse) of around 600–900 seconds, or in some cases water that is exhausted as steam for an I_{sp} of about 190 seconds.

Additionally for low performance requirements such as attitude jets, inert gases such as nitrogen have been employed.

Mixture ratio

The theoretical exhaust velocity of a given propellant chemistry is a function of the energy released per unit of propellant mass (specific energy). Unburned fuel or oxidizer drags down the specific energy. However, most rockets run fuel-rich.

The usual explanation for fuel-rich mixtures is that fuel-rich mixtures have lower

molecular weight exhaust, which by reducing M increases the ratio $\frac{\sqrt{T_c}}{M}$ which is approximately equal to the theoretical exhaust velocity. This explanation, though found in some textbooks, is wrong. Fuel-rich mixtures actually have lower theoretical exhaust velocities, because $\sqrt{T_c}$ decreases as fast or faster than M .

The nozzle of the rocket converts the thermal energy of the propellants into directed kinetic energy. This conversion happens in a short time, on the order of one millisecond. During the conversion, energy must transfer very quickly from the rotational and vibrational states of the exhaust molecules into translation. Molecules with fewer atoms (like CO and H₂) store less energy in vibration and rotation than molecules with more atoms (like CO₂ and H₂O). These smaller molecules transfer more of their rotational and vibrational energy to translation energy than larger molecules, and the resulting improvement in nozzle efficiency is large enough that real rocket engines improve their actual exhaust velocity by running rich mixtures with somewhat lower theoretical exhaust velocities.

The effect of exhaust molecular weight on nozzle efficiency is most important for nozzles operating near sea level. High expansion rockets operating in a vacuum see a much smaller effect, and so are run less rich. The Saturn-II stage (a LOX/LH₂ rocket) varied its mixture ratio during flight to optimize performance.

LOX/hydrocarbon rockets are run only somewhat rich (O/F mass ratio of 3 rather than stoichiometric of 3.4 to 4), because the energy release per unit mass drops off quickly as the mixture ratio deviates from stoichiometric. LOX/LH₂ rockets are run very rich (O/F mass ratio of 4 rather than stoichiometric 8) because hydrogen is so light that the energy release per unit mass of propellant drops very slowly with extra hydrogen. In fact, LOX/LH₂ rockets are generally limited in how rich they run by the performance penalty of the mass of the extra hydrogen tankage, rather than the mass of the hydrogen itself.

Another reason for running rich is that off-stoichiometric mixtures burn cooler than stoichiometric mixtures, which makes engine cooling easier. And as most engines are made of metal or carbon, hot oxidizer-rich exhaust is extremely corrosive, where fuel-rich exhaust is less so. American engines have all been fuel-rich. Some Soviet engines have been oxidizer-rich.

Additionally, there is a difference between mixture ratios for optimum I_{sp} and optimum thrust. During launch, shortly after takeoff, high thrust is at a premium. This can be achieved at some temporary reduction of I_{sp} by increasing the oxidiser ratio initially, and then transitioning to more fuel-rich mixtures. Since engine size is typically scaled for takeoff thrust this permits reduction of the weight of rocket engine, pipes and pumps and the extra propellant use can be more than compensated by increases of acceleration towards the end of the burn by having a reduced dry mass.

Propellant density

Although liquid hydrogen gives a high I_{sp} , its low density is a significant disadvantage: hydrogen occupies about 7x more volume per kilogram than dense fuels such as kerosene. This not only penalises the tankage, but also the pipes and fuel pumps leading from the tank, which need to be 7x bigger and heavier. (The oxidiser side of the engine and tankage is of course unaffected.) This makes the vehicle's dry mass much higher, so the use of liquid hydrogen is not such a big win as might be expected. Indeed, some

dense hydrocarbon/LOX propellant combinations have higher performance when the dry mass penalties are included.

Due to lower I_{sp} , dense propellant launch vehicles have a higher takeoff mass, but this does not mean a proportionately high cost; on the contrary, the vehicle may well end up cheaper. Liquid hydrogen is quite an expensive fuel to produce and store, and causes many practical difficulties with design and manufacture of the vehicle.

Because of the higher overall weight, a dense-fueled launch vehicle necessarily requires higher takeoff thrust, but it carries this thrust capability all the way to orbit. This, in combination with the better thrust/weight ratios, means that dense-fueled vehicles reach orbit earlier, thereby minimizing losses due to gravity drag. Thus, the effective delta-v requirement for these vehicles are reduced.

However, liquid hydrogen does give clear advantages when the overall mass needs to be minimised; for example the Saturn V vehicle used it on the upper stages; this reduced weight meant that the dense-fueled first stage could be made significantly smaller, saving quite a lot of money.