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JTEC Panel Report on Space and Transatmospheric Propulsion Technology

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JTEC Panel Report on

Space And Transatmospheric Propulsion Technology

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August 1990

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JAPANESE TECHNOLOGY EVALUATION CENTER

SPONSOR The Japanese Technology Evaluation Center (JTEC) is operated for the Federal Government by Loyola College to provide assessments of Japanese research and development (R&D) in selected technologies. The National Science Foundation (NSF) is the lead support agency. Other sponsors include the Defense Advanced Research Project Agency (DARPA), the National Aeronautics and Space Administration (NASA) and the Department of Energy (DOE).

PURPOSE The JTEC assessments contribute to more balanced technology transfer between Japan and the U.S. The Japanese excel at acquisition and perfection of foreign technologies, but the U.S. has relatively little experience with this process. As the Japanese become leaders in research in targeted technologies, it is essential that the U.S. have access to the results. JTEC provides the essential first step in this process by alerting U.S. researchers to Japanese accomplishments. The JTEC findings can also be helpful in formulating Governmental research and trade policies.

APPROACH The assessments are performed by panels of about six U.S technical experts in each area. Panel members are leading authorities in the field, technically active, and knowledgeable of Japanese and U.S. research programs. Each panelist spends about one month of effort reviewing literature, making assessments, and writing reports on a part-time basis over a six-month period. Most panels conduct extensive tours of Japanese laboratories. To balance perspectives, panelists are selected from industry, academia, and government.

ASSESSMENTS The focus of the assessments is on the status and long-term direction of Japanese R&D efforts relative to those in the U.S. Other important aspects include the evolution of the technology, key Japanese researchers and R&D organizations, and funding sources. The time frame of the R&D forecasts is up to ten years, corresponding to future industrial applications in 5 to 20 years.

LITERATURE Loyola College provides Japanese literature and translation services to the panelists. Special efforts are made to provide panelists with timely source material, such as informal proceedings from seminars and conferences in the Japanese research community, results from recent technical committee meetings on Japanese national R&D projects, and from contacts at R&D centers in Japanese high technology industries.

REPORTS The panel findings are presented to small workshops where invited participants critique the preliminary results. The panel final reports are distributed by the National Technical Information Service (NTIS), 5285 Port Royal Road, Springfield, Virginia 22161. The panelists also present the technical findings in papers and books. All results are unclassified and public.

STAFF The Loyola College JTEC staff members help select topics to be assessed, recruit experts as panelists, organize and coordinate panel activities, provide literature support, organize tours of Japanese labs, assist in the preparation of workshop presentations and reports, and provide general administrative support. Dr. Alan Engel and Ms. Kaori Niida of International Science and Technology Associates provided literature support and advance work for the panel.

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JTEC Panel on

SPACE AND TRANSATMOSPHERIC PROPULSION TECHNOLOGY

<u>FINAL REPORT</u>

August, 1990

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ERRATA SHEET FOR JTEC
SPACE AND TRANSATMOSPHERIC PROPULSION REPORT

1. Page 11, third paragraph, second sentence: by this statement, the author means that there are seven large national research universities of international renown in Japan. As of May 1986, there were actually a total of ninety-five national universities (supported by the Ministry of Education), thirty-six public universities, and three hundred thirty-four private universities in Japan.

Foreword

This is the latest in a series of Japanese technology assessments that we have been conducting under the JTEC program since 1984. In 1983 George Gamota convinced me and Bill Finan, my counterpart at the Department of Commerce, that the Nation must do more to monitor Japanese research. The methodology chosen was to use an expert panel to take a snapshot of the status of Japanese research in a critical technology by an intensive study--and to communicate the implications of the Japanese efforts to policy makers in government and industry. To provide the logistics support we contracted with the Science Applications International Corporation (SAIC).

I will review the JTEC studies that are available, but will not attempt to summarize each of their findings in a sentence or two--the full reports are available from NTIS. I will make a few overall comments on our findings at the end.

Our first effort was a series of four panels in 1984 and 1985. We asked David Brandin, then at SRI International, to chair a panel which would take a look at the broad range of computer science. Dale Oxender at Michigan chaired a panel on biotechnology. Jim Nevin at the Draper Lab led a group that looked at mechatronics, which the Japanese define as the union of electrical and mechanical engineering--things like robotics and flexible manufacturing systems. Finally we had a group, jointly chaired by Bill Spicer of Stanford and Harry Weider of UCSD, research Japanese progress on non-silicon microelectronics, such as gallium arsenide devices and optical electronics devices.

In 1986 the National Science Foundation took over the role as lead agency, and with additional funding from DARPA, we organized six panels during the next three years. Our telecommunications panel was chaired by George Turin, then Dean of Engineering at UCLA. The group on advanced materials (primarily polymers) was chaired by Jim Economy of IBM. Under Marvin Denicoff of Thinking Machines, Inc., our second computer panel took a focused look at parallel architectures, particularly the Fifth Generation Project. This group was the first to include a tour of Japanese laboratories as a formal part of its procedures, which proved to be so illuminating that all subsequent panels have taken a similar trip there. In 1988 George Gamota and Wendy Frieman compiled the results of the first six panels, with some cross-cutting observations, into the book, *Gaining Ground: Japan's Strides in Science and Technology*, published by Ballinger.

The Japanese ERATO research initiative consists of more than twenty projects intended to foster creativity and cooperation in more basic research, particularly in electronic materials and biotechnology. We appointed joint chairmen, Bill Brinkman of Bell Labs and Dale Oxender of Michigan, to cover these areas. Computer-aided design and computer-integrated manufacture of semiconductors in Japan was studied by a panel under Bill Holton of the Semiconductor Research Cooperative. Finally research in advanced sensors was assessed by a panel under Laurie Miller of Bell Laboratories.

By 1989 the project had proven to be successful enough to warrant establishment of the Japanese Technology Evaluation Center at Loyola College under a grant from NSF with additional funding from DARPA, NASA, and the Department of Energy. The current phase includes nine panels. High definition products and systems have been assessed by a group under Dick Elkus. Millie Dresselhaus of MIT has compiled a report on superconductivity applications. The present report on spacecraft and transatmospheric propulsion technology is authored by a panel chaired by Charles Merkle at Penn State. Our third computer panel is supporting the implementation of the science and technology treaty signed by President Reagan and Prime Minister Takeshita in Toronto in 1988 by identifying opportunities for joint research in advanced scientific computing. It is chaired by Mike Harrison of Berkeley. Nuclear power generation in Japan is being researched by a panel under Kent Hansen at MIT, and high temperature composite materials are being studied by a group under Judd Diefendorf at Clemson. We have recently initiated panels on construction technology and on space robotics, chaired by Richard Tucker of the University of Texas and Red Whittaker of Carnegie-Mellon University, respectively. Finally, as of this writing, we are just about to start a study of nuclear power instrumentation and control technology in Europe, our first experiment at applying the JTEC methodology to the study of European technology. Throughout this phase of JTEC, literature support and advance work in Japan are being carried out by Dr. Alan Engel and Ms. Kaori Niida of ISTA, Inc. and by Mr. Cecil Uyehara of Uyehara International Associates.

We have seen Japanese research make great progress over the course of the JTEC studies. In 1984 the conventional wisdom held that the Japanese excelled at acquiring foreign technologies, performing competent applied research to perfect them, and then developing manufacturing techniques to make high- quality products. Our early panels frequently confirmed that model, but began to report centers of excellence in more basic research as well, particularly in areas targeted by the Japanese for long term commitments. Now we are seeing more technologies where the Japanese are using the revenue stream from their favorable balance of trade to strengthen their basic and applied research capabilities. As these investments produce innovations, we in the United States must learn how to better transfer Japanese technologies to the U.S. JTEC can be a useful first step in that process, by identifying areas where the Japanese have the world's best technologies.

Frank L. Huband, Director
Division of Electrical and
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EXECUTIVE SUMMARY

An assessment of Japan's current capabilities in the areas of space and transatmospheric propulsion is presented. The report focuses primarily upon Japan's programs in liquid rocket propulsion and in propulsion for spaceplane and related transatmospheric areas. It also includes brief reference to Japan's solid rocket programs, as well as to supersonic air-breathing propulsion efforts that are just getting underway. The results are based upon the findings of a panel of U.S. engineers made up of individuals from academia, government, and industry, and are derived from a review of a broad array of the open literature, combined with visits to the primary propulsion laboratories and development agencies in Japan. The opportunity to meet with many of the Japanese scientists, engineers, and leaders was the key that made the study possible. Not only did their courtesy and cooperation aid us while we were in Japan, but also proved to be crucial in helping us establish an identity with research work reported in the literature.

Japan's long-term plans for space activity, as well as its generic paths for achieving these plans, are outlined in the *Fundamental Policy of Japan's Space Development.* This document was originally written in 1978, and has since been revised twice to include a rapidly broadening space vision, first in 1984, and most recently in 1989. Future periodic updates are expected to keep the policy consistent with anticipated advances in technology, and changing socio-economic factors. Even a cursory review of Japan's space program shows that it is a very aggressive and forward-looking program. Its ongoing activities as well as its planned programs are broad, bold, and far-reaching in perspective. Japan's future goals in space include virtually all aspects of space activity.

The current emphasis in Japan's space policy is on developing appropriate internal resources for a variety of space activities. Japan is particularly cognizant of a need to develop a space infrastructure that will enable it to encourage well-coordinated but diverse domestic space development activities, while keeping pace with, and contributing to, international space development. Its motivation arises from a desire to advance basic science and technology, expand participation in international space ventures, and satisfy a growing interest in a broad range of domestic space activities. Domestic space interests encompass activities that exploit the unique environmental conditions of space, prepare for civil space development, and promote manned space activities. Japan's plans for international collaborations include cooperation in programs established by other countries, initiation of collaborative programs of its own, including regional cooperative projects and assisting developing countries with space activities.

Japan's space program is founded upon two basic tenets that underscore all of its activities: the first is that it wishes to develop "assured access" to space; the second is that its space activities are for "solely peaceful" purposes. Although it encourages international cooperation in space, concerns may sometimes arise in conjunction with potential collaborations with the United States, because synergisms between NASA and U.S. Air Force programs may conflict with its guideline for purely peaceful uses of space. Further, its "assured access" policy dictates that it develop autonomous capabilities in space, which in many instances will duplicate capabilities in other countries. Japan's space aspirations, however, leave ample room for cooperative Japan-U.S. space endeavors, and it was clear from our visit that Japan is committed to establishing new joint ventures with the United States as well as continuing existing ones. Future joint ventures between Japan and the United States would appear to be mutually beneficial.

Japan's goals for the present decade include plans for continuing its already strong thrust in scientific space research, bringing its satellite and launch technologies up to international standards, creating the infrastructure for space station activities, and developing the basic technologies required for its own manned space activities. These near-term goals include the promotion of advanced satellite technologies such as its Engineering Test Satellite (ETS) series, and communication, broadcasting, and navigation satellites. Importance is also given to the development of scientific satellites with supporting efforts in space sciences, facilities, and tracking and control systems. These scientific areas are seen as being particularly appropriate for international cooperation. Japan's near-term plans also reaffirm its significant participation in the U.S. Space Station through the Japan Experiment Module (JEM) module. Eventually, Japan's plans call for an independent manned spacecraft, built on current technology programs.

A topic that is of primary emphasis in Japan's near-term space plans, and one that is also of central focus in this report, concerns the establishment of its own space transportation system. Self-assured access to space is envisioned as being indispensable to Japan's long-range space development activities. Near-term goals in space transportation are the development of an expendable launch system for transportation of materials to geostationary orbit, the establishment of a technology for unmanned space-to-ground transportation, and the promotion of fundamental research and development for long-term manned space transportation capabilities. Current transportation plans for expendable launch vehicles are focused on developing and enhancing the H- and M-series of liquid and solid rocket systems. These near-term systems form the basis of the first half of the present report.

Japan's long-term vision for the first decade of the new century and beyond include the implementation of its own manned space capabilities, the launch and operation of a geostationary platform, the development of an orbital servicing vehicle and an orbital transfer vehicle, and the ultimate development of its own space station. Japan also places emphasis on the commercial uses of space, with plans for manufacturing experiments, materials development, and a space factory. The advanced transportation capabilities required for these activities are discussed in detail in the spaceplane and transatmospheric propulsion sections of the present report.

The space program in Japan is under the auspices of the Space Activities Commission (SAC), an advisory body to the Prime Minister that oversees the space activities of the entire country. The primary operative body under SAC is the Science and Technology Agency (STA) which oversees and coordinates the efforts of all space programs in Japan. There are three primary agencies devoted to space initiatives. These are the Institute of Space and Astronautical Science (ISAS), the National Aerospace Laboratory (NAL), and the National Space Development Agency of Japan (NASDA). Each of these agencies has major responsibility for certain areas of space initiatives.

The current governmental budget (FY 1989) for Japan's space program, including satellites, launch vehicles, and propulsion systems, is about 176 billion yen ($1.26 billion). Detailed plans of the various agencies and laboratories, as well as the expanded goals set forth in the 1989 update of the *Fundamental Policy of Japan's Space Development*, suggest that this amount will grow in the future. Guidelines for space budgets are targeted at a level commensurate with Japan's current share of the world economy. To strengthen and encourage private sector participation it is anticipated that the government will consider appropriate financing strategies, tax incentives, and other considerations, to encourage the private sector to participate in various space activities.

There are several distinct space transportation efforts in Japan, including three expendable rocket launch vehicle programs and three air-breathing hypersonic vehicle concepts. The rocket launch vehicles include both operational systems and ones under development, the N-series, the H-series, and the M-series, while all the air-breathing hypersonic vehicles are in the concept definition phase. The N-series of launch vehicles was based upon U.S. technology and developed under license, while the currently operational H-I vehicle includes technology that is partly based on Japanese design and development, and in part retains technology developed under license from the United States. The H-II vehicle, which is currently under development and scheduled for first use in 1993, is completely Japanese in design, and positions Japan as a full-fledged member of the world launch community. The M-series rockets are solid boosters that have long been based upon Japanese design. These solid propellant rockets, which are developed at ISAS, are highly advanced with well-demonstrated launch capabilities for scientific satellites.

Japan's launch facilities at Tanegashima are located at 30.4 degrees north latitude, nearly the same latitude as our launch facilities at Kennedy Space Center (KSC), which are at 28.5 degrees north latitude. The size of the launch site is much smaller than KSC, and the transportation facilities in the immediate area are somewhat limited, but they appear to be adequate for the H-II. An agreement with local residents currently limits launch windows to a few weeks per year.

The propulsion sources for Japan's various transportation efforts can be divided into eight programs which serve as a theme for most of the present report. These programs are in various stages ranging from concept development to operational. They include four cryogenic hydrogen-oxygen rocket engines, and four advanced air-breathing systems. In conjunction with current H-series expendable launch vehicle programs, the LE-5 cryogenic propulsion engine is presently operational, while propulsion development is ongoing for the LE-5a, and the LE-7 cryogenic engines. The HIPEX expander cycle engine represents an additional new liquid hydrogen-oxygen engine development that is currently underway. The LACE liquid air cycle engine, which is also in advanced development, is a generic propulsion system oriented towards advanced air-breathing systems such as strap-on boosters for up-rated versions of the H-II, or hypersonic propulsion applications. The sixth engine configuration is the ATREX engine, an air turboramjet system that is in a similar development stage. The remaining two propulsion systems are a scramjet engine concept development program at NAL for eventual hypersonic applications, and the newly announced Mach 5 turbojet/turboramjet engine development which is being supported by the Ministry of International Trade and Industry (MITI) for high-speed commercial transportation in the Pacific Rim area.

The systems and performance of Japan's cryogenic liquid rocket engines are comparable to those of engines developed in the United States. In their designs, the Japanese have made extensive use of U.S. data, procedures, and technology, and their engines have similar specific impulse and vacuum thrust-to-weight ratios. The new engines are, however, decidedly Japanese designs, and show a number of subtle but significant philosophical differences from U.S. systems. Japanese engine development programs are composed of carefully planned steps involving low-risk, well-characterized options. The general result is a conservative design that is heavily based upon experimental engine testing, and is built upon a phased project management concept not unlike that used by NASA and USAF. This allows necessary adjustment of engine designs as the experimental results dictate. This slightly more conservative design should facilitate reliability, and may be particularly beneficial if these engines and/or their derivatives are man-rated.

In most of Japan's space propulsion program, the emphasis is based on building a launch capability to fill a need. The design requirements are set by the end product's use. Whereas, for example, the need for man-rated reliability, reusability, and high performance has driven turbopump designs for the Space Shuttle Main Engine (SSME), the Japanese have placed emphasis on expendable launch vehicles with low cost and limited life. Manned activities emphasize longer life, improved diagnostic measurements, and in general, a well-perfected product; areas in which Japan has long demonstrated expertise. Certainly Japan's capabilities in fabrication and manufacturing as well as its broad-based expertise in high technology in general, place it in a position to make very rapid advances in space capabilities, and to contribute effectively to the world's space activities.

In the area of turbomachinery, the Japanese turbopumps and turbines again demonstrate performance levels that are similar to U.S. capabilities. The Japanese are behind the United States in some areas of turbomachinery, but they are ahead in others. Their basic approach to design is to first ascertain the technology level, then apply an adequate margin to increase the probability of success, conduct component testing to verify and anchor the design, and then to proceed with the flight version. For example, in one instance, they have chosen a two-stage over a three-stage pump to avoid a technology development program. Their overall effort is a cooperative one that minimizes duplication of effort and maximizes the rate of advancement.

In the transatmospheric and hypersonic propulsion area, the Japanese are beginning a study of spaceplane concepts that emphasizes diverse topics such as aerodynamics, structures, slush hydrogen fuel, Computational Fluid Dynamics (CFD), advanced propulsion, and system development scenarios. The propulsive cycles under study are similar to those being considered in the United States, and include the turbojet, the ramjet, the turboramjet, and/or the supersonic combustion ramjet (scramjet). The propulsion systems of primary interest appear to be those for the Mach 3 to 6 range for the low Mach number portion of hypersonic cruise or SSTO vehicles, strap-on booster augmentation engines for launch systems, or air-breathing engines for a civilian SST. Efforts in higher Mach number propulsion systems are directed more toward accumulating a data base.

In terms of engine development, there are two classes of engine that are presently in the prototype phase: the LACE engine at Mitsubishi Heavy Industries (MHI), and the ATREX air turboramjet at Ishikashima Heavy Industries (IHI). There was also reference to the development of a turboramjet engine at Kawasaki Heavy Industries (KHI), but even though this is probably the least complex and risky cycle, it does not appear that engine components are presently available for this engine. Demonstration engines are currently available (or nearly so) for the LACE and ATREX engines, but the development programs have been put on temporary hold because all LH_2 facilities are now dedicated to the LE-7 development effort.

The LACE demonstrator engine uses the LH_2 pump and combustor from the LE-5 engine, along with new components for the air liquefier and the liquid air pump. This adaptation of components from existing rocket programs to new propulsion efforts is characteristic of Japanese space propulsion programs. They do a very effective job of using previously demonstrated components in advanced projects. In addition to the LACE engine, the HIPEX and the ATREX engines also contain heavy commonality with the liquid rocket engines. The ATREX engine relies upon IHI's existing turbojet-turbofan production and design experience, as well as the expander cycle technology developed in the HIPEX engine. This interchangeable component technology appears to be providing very cost-effective progress in Japan's new programs, while simultaneously enhancing the reliability of its liquid engines.

Although a considerable amount of technology development is directed toward scramjet applications, the Japanese program in this area is only in the concept definition phase, and demonstration engine development is not imminent. Japan appears to have significant interest in the development of a hypersonic vehicle as a member of a consortium, instead of all alone. The general feeling is that the technology is now available for the LACE and ATREX engines, but that technology for the scramjet engine is not yet accessible.

The scramjet technology programs include experimental studies of supersonic combustion, including ignition and diffusion flame studies, and shock tube studies of elementary reaction kinetics of hydrogen. In addition, high-speed inlet tests are currently underway on a scale model. This work takes place in the national laboratories and at several universities. Two new university efforts are also underway that involve some twenty faculty at several schools and are oriented towards hypersonic reacting flows and component technology for advanced propulsion systems. To complement these experimental studies, CFD studies of scramjet configurations are being conducted at NAL Chofu, where researchers are using this experimental data to validate and anchor CFD codes. In terms of facilities, there are scramjet facilities at NAL Chofu, NAL Kakuda, and the University of Tokyo, which all have capabilities for Mach 2. A new scramjet facility is being built at Kakuda.

Japan is also pursuing advanced fuels development and on plant construction for hydrogen production. Japan has developed two high-density hydrocarbon fuels for rocket applications, and is in the process of stepping up its hydrogen production capabilities to serve the H-II and advanced air-breathing propulsion systems. It is currently constructing a new hydrogen plant that makes hydrogen as the byproduct of ethylene production, and is building a pilot facility for the production of hydrogen from coal gasification.

In the area of advanced diagnostics, Japan is a user of the latest systems from the United States and Europe, but is a leader in the development and manufacture of many of the basic lasers, optics, and electro-optic components that go into these systems. Of particular interest to advanced diagnostics implementations are new tunable diode lasers that are being developed in Japan, and a new surface-emitting diode laser with reduced beam divergence that offers possibilities for improved spatial resolution.

The area of CFD, which is an important supporting area in all propulsion development, represents an area of strength in Japan. Japanese supercomputers are among the best in the world, and there are major supercomputer installations at NAL and at the privately owned Institute for Computational Fluid Dynamics. The national universities also have excellent supercomputing capabilities. This availability of supercomputer access has resulted in rapid progress in computational areas. The Japanese routinely include real gas effects and complex reaction kinetics in flow field analyses, and their codes are based on the latest algorithms. Their visualization and postprocessing capabilities are also at the leading edge. Clearly, they have appropriate CFD capabilities to enable them to move rapidly in this aspect of propulsion development.

Finally, we note that contractor selection in Japan is an area in which there are differences from that in the United States. Although competition exists, particularly at the concept development level, the award of new propulsion contracts is generally based on the technical capabilities which the contractors have demonstrated in previous projects. For example, MHI is generally the overall engine developer for liquid rocket engines, while IHI will generally emerge as the turbomachinery contractor. The project share generally appears to be set by historical factors, rather than by competitive procedures. The role of Japanese industry is coordinated and strengthened through the *Keidanren* and the Society of Japanese Aerospace Companies (SJAC).

In conclusion, the Japanese have a carefully thought out plan and a broadly based program for propulsion activity. Their schedule is ambitious, but achievable. At present, their overall propulsion program is behind that in the United States, but they are catching up fast across the board. They are at the forefront in such key areas as advanced materials, they enjoy a high level of project continuity and funding. Japan's space program has generally been evolutionary in nature, while the U.S. program has put a greater emphasis on revolutionary advances. Projects have typically been smaller than in the U.S., focusing on incremental advances in technology, with an excellent record of applying proven technology to new projects, such as their present HIPEX, LACE and ATREX programs. This evolutionary approach, coupled with an ability to take technology "off the shelf" from other countries, has resulted in relatively low development costs, rapid progress, and enhanced reliability. Their projected H-II growth rate for the late

1990s will enable the H-II to equal or surpass the payload capability of the largest U.S. expendable unmanned cargo carriers. Clearly Japan is positioned to be a world leader in the areas of space and transatmospheric propulsion technology by the year 2000.

CHAPTER 1

AN INTRODUCTION TO JAPAN'S SPACE PROGRAM

Charles L. Merkle

SCOPE OF THE STUDY

The present JTEC panel was originally conceived in discussions between NASA, NSF, and JTEC personnel as a panel on combustion. Over the course of a few months' deliberations, the topical area gradually evolved from this broad area of fundamental science to the more applications-oriented area of combustion as applied to propulsion, and finally settled upon propulsion itself. It was felt that this applied subject area would prove to be more appropriate for a JTEC assessment, and in addition, would represent an area that was of particular interest to NASA, the agency responsible for funding the panel. The immediate goals of this panel, which are consistent with those of previous JTEC panels, were to identify current and planned Japanese programs in the subject area (in this case propulsion) and determine their present technology status. This information is envisioned as being a first step in heightening awareness in the U.S. community of Japanese efforts and capabilities in propulsion with the end result of stimulating enhanced collaboration and technology transfer.

A kickoff meeting for members of the panel was held on May 9, 1989 in Washington, D.C. to familiarize the panel members with the goals of the JTEC program, to meet the supporting groups at NSF, Loyola College, and ISTA, and to initiate the actual panel study. Perhaps the most significant tasks of this May 9 meeting were to define those aspects of propulsion to be covered by the panel, and to assign an appropriate split of sub-areas among the individual panel members. The general guidelines were to stay within those propulsion areas of particular interest to NASA to select a panel coverage that was of proper breadth for the size of the study and the duration of the stay in Japan.

The primary propulsion areas of interest to NASA were air-breathing and non-air-breathing propulsion. These were subdivided into liquid and solid rockets, advanced space propulsion concepts, advanced hypersonic air-breathing propulsion, and supersonic and subsonic transport propulsion. Of these, the panel chose to focus on liquid rockets and advanced air-breathing concepts, centering on transatmospheric propulsion. The two additional areas most discussed were solid rockets and supersonic transport propulsion. Of these, the panel recognized Japan's key technological strength in solids in its Japan Defense Agency (JDA), but decided to omit this topic because of the possible difficulties in obtaining permission to visit military installations. Similarly, the area of supersonic transport propulsion was recognized as an area of importance in Japan, but was omitted because it brought in an entirely new group of Japanese corporations and agencies, and it was expected that these additional responsibilities would be more than the present panel could assume.

The final task of the May 9 meeting was to fix the date for our visit to Japan as the week of September 25, 1989 and to identify November 29, 1989 as the date for our oral presentation in Washington. These dates set the general calendar and pace of activities for the panel's mission.

NATIONAL ORGANIZATION OF THE SPACE PROGRAM

In Japan, as in the United States, both national space policy and the goals and aspirations of the space program are stipulated by the government. Accordingly, we begin this introductory chapter by reviewing the Japanese national space policy and the governmental agencies that are central players in its implementation. We then overview the entire space program with special focus on propulsion. Details of the propulsion programs are given in later chapters.

The Fundamental Policy of Japan's Space Development

The broad vision on policy statement of Japan's space program is given in the *Fundamental Policy of Japan's Space Development.*[1] This policy statement was first enunciated in March of 1978 and was revised and updated in February 1984 and again in June 1989. The primary reasons for revising the policy were to enlarge and broaden the focus of the space program. The current policy calls for a very aggressive and extensive mission in space but, it should be noted that this policy represents only Japan's vision for space; the specific future space activities included may not necessarily be matched by committed funding, and actual

[1] The author had access only to an unofficial translation of this Japanese Government document.

program budgets may develop in different directions. Prime tenets of the current space program are

To promote national space development in balance with national resources and social needs

To ensure autonomous space capability within Japan so it can freely conduct various diversified space activities

To work in harmony and cooperation with international space activities in other countries

The fundamental premise of Japan's space activity is that the program be carried out solely for peaceful purposes in keeping with the dictates of Japan's constitution.

The breadth of Japan's planned space vision is impressive. The plan includes autonomous launch capabilities by means of the H-II and successor launch vehicles, as well as a single-stage-to-orbit (SSTO) spaceplane. Japan's involvement in the U.S. Space Station is a key step in its development of manned space activities. Japan also envisions strong programs in both scientific and applications satellites, and support of these and other space programs requires eventual development of orbital servicing vehicles and orbital transfer vehicles. Finally, Japan envisions the deployment of a free flyer, a manned space platform, and a space factory.

In terms of staging, the Japanese policy calls for certain of its programs to be implemented within the coming decade, while the remainder are to be addressed after this decade. Specific projects for this decade are

To encourage the commercial development and manufacture of communication, broadcasting, and navigation satellites

To continue development and operation of scientific satellites for meteorological, oceanographic, and earth observation purposes

To promote the utilization of space for scientific and technological experiments such as materials research and life science experiments

To develop the necessary basic technologies for later autonomous manned space exploration through international cooperative programs such as the Japanese Experimental Module (JEM) of the U.S. Space Station

To continue to promote satellite technology development through programs such as the Engineering Test Satellite (ETS) series for satellite-to-satellite communications, and basic technologies development such as high-performance solar cells, high-precision attitude control systems, and active thermal control techniques

To maintain and expand Japanese capabilities for space transportation, including bringing the H-II series rocket to operational status and establishing technology for new manned and unmanned space transportation systems

This latter area (space transportation), which is of central importance to the present report, includes operational deployment of the H-II rocket as the primary service vehicle for the 1990s, along with continued enhancements of the M-series solid rockets. With these systems, the Japanese have the capability of providing flexible launch service for foreign as well as domestic customers. These developments are described in detail throughout the report with emphasis on the H-II system.

During the present decade, Japan will have to rely on other countries for space-to-ground missions, but its near-term plans call for research and development on several fronts to enable it to provide an ever-increasing range of space transportation. Specific projects envisioned include an H-II-launched, winged recovery vehicle (the HOPE), as well as spaceplane-type vehicles for unmanned and manned flight. Space transportation efforts will also include maintenance and upgrade of launch site and tracking-control facilities.

Key Governmental Agencies

Japan's space program is overseen and promoted by the Space Activities Commission (SAC), which is an advisory body to the Prime Minister's Office. This commission is composed of the Minister of State for Science and Technology (who is a member of the cabinet) plus four other members nominated by the Prime Minister. Its functions are to plan, deliberate, and decide on matters of space policy, budget, cultivation and training of space researchers and engineers, and other matters of importance in space activity. In addition, SAC coordinates space-related activities among the various government agencies, and seeks to unify and promote their functions. In particular, as part of its policy-making role, SAC is responsible for formulating the *Fundamental Policy of Japan's Space Development.*

While SAC oversees and coordinates Japanese space policy, the implementation of these policies and plans is carried out by the Science and Technology Agency (STA), a government agency that reports to the Prime Minister's Office. STA plans and programs the basic space-related policy, and ensures the coordination of space-related activities among the various government agencies. Because of their key role in implementing the major Japanese space policies and programs, STA personnel are in a privileged position to provide policy inputs. STA is one of several cabinet-level agencies that have space-related appropriations. Others include the Ministry of Education (Monbusho), the Ministry of International Trade and Industry (MITI), the Ministry of Transport, and the Ministry of Posts and Telecommunications.

In addition to its oversight functions, STA also conducts research and development in space-related activities through two organizations attached to it: the National Aerospace Laboratory (NAL), and the National Space Development Agency of Japan (NASDA). NAL is the government laboratory that serves as STA's basic research arm for both space and aeronautics, while the NASDA is a quasi-government agency that functions as the development organization for space activities. Because of its quasi-governmental status, NASDA is some what less dependent on STA than is NAL.

The National Aerospace Laboratory was originally chartered in 1955 as the National Aeronautical Laboratory, with the mission of developing aeronautical technology in Japan. NAL's mission was expanded in 1963 to include research in space technology while continuing its efforts in aeronautics. It was at this time that NAL's title was changed to its current form. Today NAL remains the lead laboratory for space and aeronautics technology. Its headquarters are at Chofu in Tokyo, with additional installations at the nearby Chofu Airfield Branch in Mitaka. NAL also has a major experimental facility, the Kakuda Research Center, near Sendai to the north of Tokyo.

The primary functions of the second agency, the National Space Development Agency, are to develop applications satellites and appropriate launch vehicles. In addition, NASDA conducts the actual launch of the vehicles, and the operation and tracking of the satellites. NASDA's launch activities started with the N-I and N-II vehicles that were based on technology licensed from the United States. NASDA developed the LE-5 second stage hydrogen-oxygen engine for the H-I, and is now developing the H-II vehicle, a launch vehicle completely indigenous to Japan. NASDA's primary facilities include its headquarters in the World Trade Center Building in Tokyo; rocket launch facilities at the Tanegashima Space Center in Kagoshima; its tracking and control center, as well as major satellite research and development facilities, at the Tsukuba Space Center; the Earth Observation Center for remote sensing at Hatoyamimachi, and the Kakuda Propulsion Center. This last center is situated adjacent to NAL's Kakuda Research Center, and has major facilities for propulsion system testing, as is detailed later.

The Institute for Space and Astronautical Science (ISAS) under the Ministry of Education is the third primary agency for propulsion research and development in Japan. Along with NAL and NASDA, ISAS completes the triad of key governmental (and quasi-governmental) agencies for propulsion. ISAS's primary responsibility is in the area of space science. In addition, ISAS serves as the coordinator of all space science research in Japan. Even though ISAS is an inter-university organization, it has a very broad charter in space science. This charter includes responsibility for research, development, and operation of both the scientific satellites that are used for space exploration and the balloons, suborbital sounding rockets and M (Mu) family of orbital solid rockets that are used to launch the satellites. Historically, ISAS and NASDA have maintained quite separate programs, each charged independently with maintaining appropriate launch vehicles.

Until 1981, ISAS was a part of the University of Tokyo, but is now an independent inter-university research institute. In 1989, the Institute moved to a very impressive new campus in Sagimihara on the outskirts of Tokyo, but still maintains its old Komada Campus in central Tokyo. In addition to these two campuses, ISAS maintains the Kagoshima Space Center for launching scientific and sounding rockets; the remote Noshiro Test Center for LOX/LH_2 engine testing; the Usuda Deep Space Center for deep space tracking, telemetry, and control; and the Sanriku Balloon Center for balloon launch, telemetry, and control.

The Ministry of International Trade and Industry (MITI) supports space activities through Japan's Earth Resources Satellite (ERS) program and the Space Free Flyer Unit (SFU). In these areas, MITI is responsible for the development of remote sensing technologies and research on advanced industrial technology for the SFU. MITI also has a special focus on in-space technology for controls and robotics through its Agency of Industrial Science and Technology. In areas more closely related to propulsion, MITI is involved in studies on spaceplanes, and is also just beginning the development of a new air-breathing propulsion system for supersonic transports. MITI's interest in propulsion has grown more from its involvement in aeronautics than from its involvement in space activities which has traditionally been relatively small.

Other space-related activities in Japan include research and development on ship and aircraft navigation systems, and geodetic and meteorological surveys by the Ministry of Transport. The Ministry of Posts and Telecommunications also conducts research on broadcasting and communications, including current plans for launching and operating a high-definition television (HDTV) satellite. Finally, additional space-related budget allocations for communication are made by the National Police Agency, the Fire Defense Agency, and the Geographical Survey Institute.

Role of Private Industry in Space Activities

Industry is involved in the space program as contractors to the major government agencies and through investment of private capital, but in addition to this, Japan has a federation of economic organizations called *Keidanren* that functions as a planning agency. *Keidanren* is a private, non-profit economic organization that addresses economic problems and contributes to economic development. It has 933 corporate and 120 associated members and is headed by Japanese business leaders. *Keidanren's* space activities are conducted under the Space Activities Promotion Council, which has been active since 1968 and currently represents 69 companies. The council cooperates closely with the government in the formulation of space programs, presents the views of industry to SAC, endeavors to improve technological development capability and domestic production, and strives to further international cooperation.

BUDGET FOR SPACE ACTIVITIES[2]

In the above section, we have identified the primary government agencies as including NAL, NASDA, and ISAS, with additional efforts from MITI and other cabinet ministries. Here, we look at the overall federal budget for space activities, the allocations to each agency, and its personnel levels. There is some uncertainty in presenting any such set of figures because of the grey areas where activities could be construed as either space or non-space-related, but the numbers are generally indicative of the overall level of effort. The budgetary estimates presented here do not include private corporate-supported efforts, which would appear to be proportionally larger than similar investments in the U. S. In general, the Japanese space budget grew rapidly during the seventies, and then slowed to a modest growth of about 5% per year between 1976 and 1986. The impact of Japan's broader role in space is seen in the space budget from 1986 to 1989, when it grew by about 10% per year. Discussions with various space program leaders while we were in Japan, as well as the projected increase in activities that the recent revisions to the *Fundamental Policy of Japan's Space Development* indicate, suggest this more robust growth rate will continue (or accelerate) in the near future.

For fiscal year 1989 (April 1, 1989 to March 31, 1990) the budget is set at 155 billion yen. Additional support (identified below) raises this total to about 176 billion yen. At 140 yen per U.S. dollar, this corresponds to an annual budget of $1.26 billion.

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[2] Information and statistics presented in this section are based on a compendium of brochures given to the Panel by our Japanese hosts at the various sites visited.

Major ongoing projects that are scheduled for completion by 1995 include the H-II rocket (¥400 billion) and the JEM module for the Space Station (¥310 billion). The ETS-VI satellite program (¥730 billion), space science (¥400 billion), and research and facility construction for tracking and control (¥880 billion) represent other ongoing projects which are planned through the year 2000. New starts of major programs include the spaceplane (¥2,300 billion) and an orbital transfer vehicle (¥900 billion). Space infrastructure research and launch and operations costs are expected to run to about ¥560 billion. Tentative plans are to start efforts on Japan's own space station will start in the year 2000.

OVERVIEW OF INDIVIDUAL AGENCIES

National Aerospace Laboratory (NAL)

The National Aerospace Laboratory is a research arm of the Science and Technology Agency. Its charter is to conduct research in aeronautics and space technology. NAL has approximately 450 employees, including some 325 research staff. The number of employees has been approximately constant since the mid-sixties. NAL's annual budget of approximately ¥10 billion ($72 million) is split into roughly 25% for personnel, 50% for research, and 25% for facilities. Its budget has been flat since about 1982; however, there has been a major increase in facility expenditures at the expense of the research budget for the past four years. It is presently expecting substantial near-term budget growth.

Major research areas at NAL include research on Short Take-off and Landing (STOL) aircraft, space transportation systems, the utilization of space environment and satellite systems, and numerical simulation technologies. Recently NAL has started a new major emphasis on innovative aerospace transportation technologies to aid in developing high-speed flight and SSTO-type launch capabilities. NAL is also involved in technology transfer applications from space technology and aeronautical sciences to other fields.

NAL's emphasis on STOL has been focused on the Asuka experimental aircraft. Research on this aircraft began in 1977 with preliminary design, followed by detailed design and manufacturing. The aircraft entered flight experiments in 1985 with the final year of testing scheduled for 1988. The purpose of this project was to verify new technologies pertinent to STOL performance for a quiet domestic fanjet for commercial transport.

The research on the utilization of space environment includes a variety of space-based technologies, including flexible structures, microgravity technologies, solar power generation, journal bearing research, and the development of a xenon ion thruster. The ion thruster development program, which is not discussed further in the present report, is aimed at station-keeping of a geostationary satellite. A life span of ten years is targeted, with first implementation in the 1990s. Emphasis is on improved endurance and high performance. Both a conventional Kaufman thruster concept and a ring cusp thruster concept are being evaluated. The program includes both an experimental program and numerical simulations.

The major portion of Japan's space transportation research has focused on the development and improvement of LOX/LH_2 rocket engines. At present, NAL is developing the high-pressure LOX pump for the LE-7 first stage of the H-II rocket. This work, along with other rocket engine development work at NAL, is described in detail in Chapters 2 through 4.

The innovative Aerospace Technologies Project was initiated in 1987 with a view to developing necessary technology for a spaceplane and subsonic/supersonic/hypersonic airplanes. The spaceplane is viewed as essential to meeting Japan's plans for autonomous space activities, while the hypersonic work is for advanced civil transportation of the future. This work is discussed in detail in Chapters 5 through 7.

With respect to technology transfer to other fields, NAL is studying low-emission catalytic converters for stationary gas turbines, and conducting research in ceramic gas turbines for MITI.

NAL also possesses major facilities such as wind tunnels, gas turbine test facilities, flight simulators, materials testing machines, and rocket test facilities. In addition, NAL has major supercomputer facilities, including a FACOM VP 400 (1.1 GFlops, 1Gb) and a FACOM VP 200 (570 MFlops, 128 Mb). These computer and rocket test facilities are described further in succeeding chapters.

The National Space Development Agency (NASDA)

The National Space Development Agency's charter is to develop and operate applications satellites and launch vehicles. NASDA was created as a quasi-government corporation in 1969. During its early years in the first half of the 1970s, its budget grew at a rate of about 50% per year. From 1975 onwards, its budget growth slowed to a few percent per year, but began to increase again in 1988. After a 2% increase in 1987, NASDA received a 4% increase in 1988, and a nearly 11% increase in 1989. This trend again reflects the stepped-up pace of space activities in Japan. Continued budget growth is expected. In terms of manpower, NASDA is a relatively small organization. Its employment is currently at about 950 people with only slow growth since 1980.

NASDA's current budget is about ¥124 billion, including ¥107 billion from STA. The remainder represents revenues from other sources such as satellite launches which have constituted a sizeable portion of its budget since it started N-1 launches in 1975. NASDA's budget goes to fund both in-house activities and to support industrial contractors.

Current major projects at NASDA include development of the H-II launch vehicle which is described in the following chapters, launch and operations for the H-1 vehicle which is currently in use, and the development, launch, and operation of applications satellites for broadcasting, meteorology, and telecommunications. NASDA is also in charge of the development of in-space experiments such as the First Material Processing Test (FMPT) scheduled for 1991 on the Space Shuttle/Spacelab, and the Japanese Experiment Module (JEM) for the Space Station. NASDA also participates in the Space Free Flyer Unit (SFU) with ISAS and MITI. At present, NASDA is just beginning efforts on the development of air-breathing propulsion systems for future unmanned and manned space flight.

The Institute for Space and Astronautical Sciences (ISAS)

The Institute for Space and Astronautical Sciences (ISAS) is an inter-university research institute under the direction of the Ministry of Education. ISAS has a broad charter for research in space science, including research, development, and launch of both rockets and satellites. Besides its own M-series of solid rockets, ISAS is also heavily involved in other propulsion development programs including its previous involvement in the initial development of Japan's first LOX/LH_2 engine, and its current involvement in the development of a LOX/LH_2 expander cycle engine. Through its broad-ranging research topics and informal connections with industry, ISAS commands a very important position in Japan's space program.

ISAS has a total of about 300 employees including 80 faculty (30 professors, 27 associate professors and 23 visiting professors and associate professors). There are just over 100 students, all of whom are pursuing graduate degrees. Over the years, ISAS has seen steady budget growth of several percent per year, except for a brief interlude in the mid-eighties. Its current budget estimate from SAC for FY 1989 is ¥20.8 billion, with an additional base budget of ¥3.5 billion. Its balloon observation program also has a separate budget of ¥0.1 billion to bring the total for FY89 to ¥24.4 billion, or approximately $175 million.

ISAS has been active in space research since 1955 and during this time has developed the M-series of solid rockets. The newest version is the M-3SII, a three-stage solid booster system with strap-ons. The M-3SII can launch a 770 kg payload into low earth orbit, and a 170 kg payload into solar orbit.

The Ministry of International Trade and Industry (MITI)

The Ministry of International Trade and Industry has a number of space-related activities with the primary objective of promoting the industrial utilization of space. MITI is involved in developing the Earth Resources Satellite-1 (ERS-1) and is a partner in the development of the Space Free Flyer Unit (SFU). As noted earlier, MITI also has a role in the development of space-based robots and remote control system technology. MITI also influences aerospace activities through the roles of tax and investment incentives and financing strategies.

MITI's direct role in propulsion is small, but it is important. At present, it is actively encouraging the development of high-speed civil aviation. It has just recently announced the start of a $200 million program to develop a turbojet/ramjet engine for a Mach 5 Supersonic Transport (SST0). This program will undoubtedly have important implications for Japan's spaceplane and hypersonic propulsion program, as well as for civil air transport.

The Role of Universities in Japan's Space Program

In addition to the above government and government-related agencies, we also note the importance of Japan's universities in the space program. Japan has a total of seven national universities plus an abundance of public and private universities. The national universities are the University of Hokkaido, the University of Kyoto, the University of Kyushu, the University of Nagoya, the University of Osaka, the University of Tohoku, and the University of Tokyo. The national universities all have broad-based research programs that are supported at various levels by the Ministry of Education. A particular point of interest is that all seven national universities have supercomputers on their campuses.

In terms of supporting the space program, the universities perform the standard research and education functions with which we are familiar in the United States. The professors, however, probably play a more significant role than do their American counterparts in influencing national programs through their participation in advising functions on both formal and informal levels.

The direct research efforts of the national universities are funded through an annual base budget for research activities. These funds are split up among faculty at each institution and are essentially discretionary with regard to research topics. In addition, the Ministry of Education provides separate grants-in-aid to individual professors or groups of professors on the basis of competitive proposals. Frequently, these grants-in-aid are used to encourage channeling of the base research funds into selected areas. Faculty at the national universities may also receive grants-in-aid from public and private organizations, but are generally limited to the Ministry of Education as their sole source of government grants-in-aid.

In general, the direct financial research support provided to individual professors is quite small, but the differences in the way finances are reckoned in Japan and the United States make direct comparisons misleading. For example, U.S. faculty members use research funds to pay for their own salaries as well as those of students and staff members in addition to paying overhead rates. Faculty and student salaries in Japan are generally covered separate from the research grants. As an example of the direct monetary support received by faculty, the Aeronautics Department at the University of Tokyo receives approximately $20 thousand per faculty member per year. Recently, a group of 20 professors at four universities including the University of Tokyo received a $130 thousand grant for FY1989-91. This program, entitled "Hypersonic Reactive Flows in Scramjets," is thus used to encourage these faculty to funnel their base research funds into the hypersonic area. Despite the apparently small size of this grant, faculty indicated it was a major help in building a strong research effort in this area.

Finally, a new cooperative agreement between the University of Tokyo and NAL has just been reached which forms a joint institute that will enable University of Tokyo faculty to use research facilities at NAL. This is also a promising new avenue for university research.

Major Space Propulsion Contractors

The major space propulsion contractors include Mitsubishi Heavy Industries (MHI), Ishikawajima-Harima Heavy Industries (IHI), Nissan Motors, and Kawasaki Heavy Industries (KHI). MHI is the systems contractor for Japan's liquid booster systems; IHI has responsibility for the turbomachinery in the liquid engines; Nissan is the primary solid rocket company in Japan; and KHI is involved in advanced air-breathing propulsion development and related studies. The specific efforts of MHI and IHI on rocket and spaceplane propulsion are detailed in succeeding chapters.

OVERVIEW OF JAPAN'S ROCKET LAUNCH VEHICLES

Japan's launch vehicles include the N-I and N-II rockets which were produced in Japan under license to U.S. technology, and the H-I and H-II vehicles which represent Japanese designs and technology. The N-I is a three-stage vehicle whose first and third stages were based on Thor-Delta technology. The second-stage (storable propellants) engine, the LE-3, was developed by NASDA. The N-II vehicle likewise relied upon U.S. technology with most production taking place in Japan. Between 1975 and 1982 the N-I was used to lift seven satellites, while the N-II was used to orbit eight satellites between 1981 and 1987. Additional details are given in Chapter 2.

The H-I vehicle, which is currently operational, was Japan's next vehicle used to increase launch capability. The H-I includes the same U.S.-based first stage and strap-ons as the N-II. The second-stage engine, however, is developed by NASDA. This engine, the LE-5 is a high-performance LOX/LH_2 engine, and represents Japan's first LOX/LH_2 engine.

In the H-II system currently under development, the entire vehicle, including all propulsion systems, is designed and developed by NASDA. The H-II uses the LE-7 engine in the first stage and the LE-5A, a redesigned LE-5, in the second stage. This system is scheduled to become operational in 1992 or 1993. Chapters 2 through 4 give details of these Japanese-developed propulsion systems.

In terms of solid rockets, ISAS has developed the M-series of launchers. Payloads in this series range from 25 to approximately 2000 kg. With this family of launch vehicles, ISAS launched Japan's first satellite in February 1970. From that time through February 1989 ISAS successfully launched 17 more satellites of increasing complexity. Its current plans call for continued launches at the rate of about one per year for the foreseeable future.

JAPANESE SPACEPLANE CONCEPTS

Japan's interest in advanced air-breathing propulsion includes several different concepts. As these form the basis of Chapter 5, only an introduction to these concepts is given here. The three major agencies each has its own version of an advanced air-breathing vehicle, each with a separate purpose. At present all are in a study phase, although substantial progress in propulsion systems has been made, including fabrication and testing of subcomponents, with full-scale testing of some advanced engines ready to begin.

NASDA's spaceplane concept is the HOPE (H-II Orbiting Plane) vehicle. This vehicle is to be lifted by the H-II and to be used for return missions to earth. The HIMES (highly maneuverable experimental space vehicle) is being studied by ISAS. Its purpose is to develop technology for high-speed flight. The National Aerospace Laboratory is performing systems studies on a hypersonic experimental plane to be used for manned single-stage-to-orbit scenarios. Finally, MITI's new initiative on propulsive engines for Supersonic and High Speed Transport (SST/HST) applications should also be included in this list of advanced concept applications.

SUMMARY OF ON-GOING PROPULSION PROGRAMS

The remainder of this report will focus on eight propulsion programs. These are for use in the launch vehicles and spaceplanes noted above. These include the four hydrogen-oxygen rocket engines, the LE-5, LE-5A, Le-7, and HIPEX. The LE-5 is a gas generator cycle engine used for the second stage engine of the H-I. The LE-5A is an expander bleed cycle, a modification of the LE-5, and serves as the second stage for the H-II vehicle. The first stage of the H-II is powered by the LE-7, a pre-burner cycle engine. Finally, the expander cycle HIPEX engine is an experimental engine that is presently undergoing testing.

In terms of air-breathing propulsion, two experimental engines are currently in development. These are the ATREX engine, an expander cycle turboramjet using much of the technology from the HIPEX engine, and the LACE (liquid air cycle) engine. The third air-breathing engine is the turbojet/ramjet Mach 5 engine being developed by MITI. In addition, NAL is studying a scramjet engine which is presently in the concept phase.

ACKNOWLEDGMENTS

As part of the present review of Japanese space and transatmospheric propulsion technology, the JTEC panelists visited a number of key organizations in Japan during the last week of September 1989. The sites visited included the National Aerospace Laboratory's main branch in Chofu, and Kakuda Branch near Sendai; the National Space Development Agency's headquarters in Tokyo, Kakuda Propulsion Center near Sendai, and Tanegashima Space Center in Kagoshima; the Science and Technology Agency in Tokyo; the Sagimihara Campus of the Institute of Space and Astronautical Science; and the Campus of Nagoya University. Industry sites visited included Mitsubishi Heavy Industries in Nagoya, and the Mizuho and Tanashi Plants of Ishikawajima-Harima Heavy Industries. In addition to these visits, we had discussions with representatives from Tohoku University and the University of Tokyo. (See Appendices for complete listing of sites visited.)

In all instances, we were very warmly received by our Japanese hosts, and they were more than cooperative in providing us with necessary information and in answering our questions. The panel is deeply indebted to our Japanese colleagues for their major part in the success of the panel.

We would like to thank our sponsors at NASA (Greg Reck and Len Harris) and NSF (Frank Huband) for making this study possible. We would also like to thank the JTEC staff (Duane Shelton, Geoff Holdridge, Bob Williams, Pat Johnson, Aminah Batta, and the Loyola College student research assistants) for their work in coordinating the study and producing the final report. Finally, we are indebted to ISTA, Inc. (Alan Engel and Kaori Niida) for their vital role in setting up the meetings with our Japanese colleagues.

The entire panel was highly impressed by Japan's space transportation capabilities and we look forward to more cooperative ventures between our two countries as a result of our visit.

CHAPTER 2

SPACE PROPULSION SYSTEMS

John P. McCarty

INTRODUCTION

This chapter presents the results of a brief study of liquid propellant rocket engines and propulsion systems in Japan. The purpose of this study was to understand the status and direction of this key element of the space program conducted by Japan. Although the propulsion elements of this program were originally based on U.S. Delta rocket technology, the program has evolved past this stage to where future capability will be based largely upon domestic Japanese technology.

The focus will be on liquid hydrogen fueled rockets. This is due to the Japanese focus as revealed in their domestic development programs; to the availability of data in the literature on capabilities, activities, and plans; and to the fundamental importance of hydrogen as a fuel for space propulsion systems. In addition, the section will necessarily deal more with technical parameters and data than with cost and reliability, due to the availability of comparable information.

VEHICLE PROPULSION SYSTEMS

As stated in Chapter 1, two organizations, the Institute of Space and Astronautical Science (ISAS) and the National Space Development Agency (NASDA), are responsible for the development of launch vehicles in Japan. ISAS, which was established by reorganizing the Institute of Space and Aeronautical Science of the University of Tokyo, has developed solid rockets for launching its space science missions and participated in technology development for liquid propellant propulsion. NASDA, which was established in 1969 to accomplish development and operations in the application fields, has built and operated the N-series and the H-series of launch vehicles. The N-I and N-II were developed based on Delta rocket technology imported from the United States. The last launch was in February 1987 and delivered a MOS-1 satellite into low earth orbit. The new series of H vehicles was initiated with a successful launch of the H-I in August 1986. The H-II, now in the development stage, will raise the low earth orbit capability to over 20,000 lbs.

An overall look at the launch capability of the Japanese N and H vehicles relative to U.S. expendable launch vehicles is contained in Figure 2.1. It shows that the launch efficiency, in terms of low earth orbit (LEO) payload-to-vehicle-liftoff-weight ratio and size of the N-I vehicle, is very comparable to the Delta vehicle on which it was based. The H-I introduces a modest efficiency improvement through the development of a cryogenic second stage. The H-II, with an improved second stage, a new cryogenic first stage and solid rocket motors, however, represents a substantial increase in efficiency and increase launch weight.

H-I Launch Vehicle

The H-I is a three-stage launch vehicle capable of launching payloads of about 5400 lbs. to low earth orbit, or 1,200 lbs. to geosynchronous orbit. Development of the vehicle was begun in 1981. The main propulsion changes in evolving from the N-II to the H-I were the development of a hydrogen fueled second stage and engine, and a solid rocket motor third stage using domestic technology. The H-I is now the main space transportation system of NASDA.

The second stage, which is the major propulsion change from the N-II vehicle, is a liquid oxygen and liquid hydrogen stage utilizing a high-performance, gas generator cycle engine, the LE-5. The stage has a restart capability providing for the efficient delivery of payloads to a range of destination orbits. The basic characteristics of this stage are shown in Table 2.1 (Refs. 2.4 and 2.42).

The propellant tank system consists of an integrated LOX/LH_2 tank with a common bulkhead. The tanks are equipped with an external insulation of polyurethane foam to prevent ice and liquid air formation prestart, and to control heat input to the propellants from aerodynamic and orbital heating during flight. Level sensors are mounted in each of the tanks for propellant loading. Each tank also has three redundant point level sensors for detecting propellant depletion.

Helium for tank pressurization, purges, and valve actuation is stored in ambient temperature bottles at 4,400 psi and in cryogenic temperature bottles inside the fuel tank at 3,000 psi. Ambient temperature helium is used for prestart tank pressurization of both tanks. During main stage operation, GH_2 tapped off the

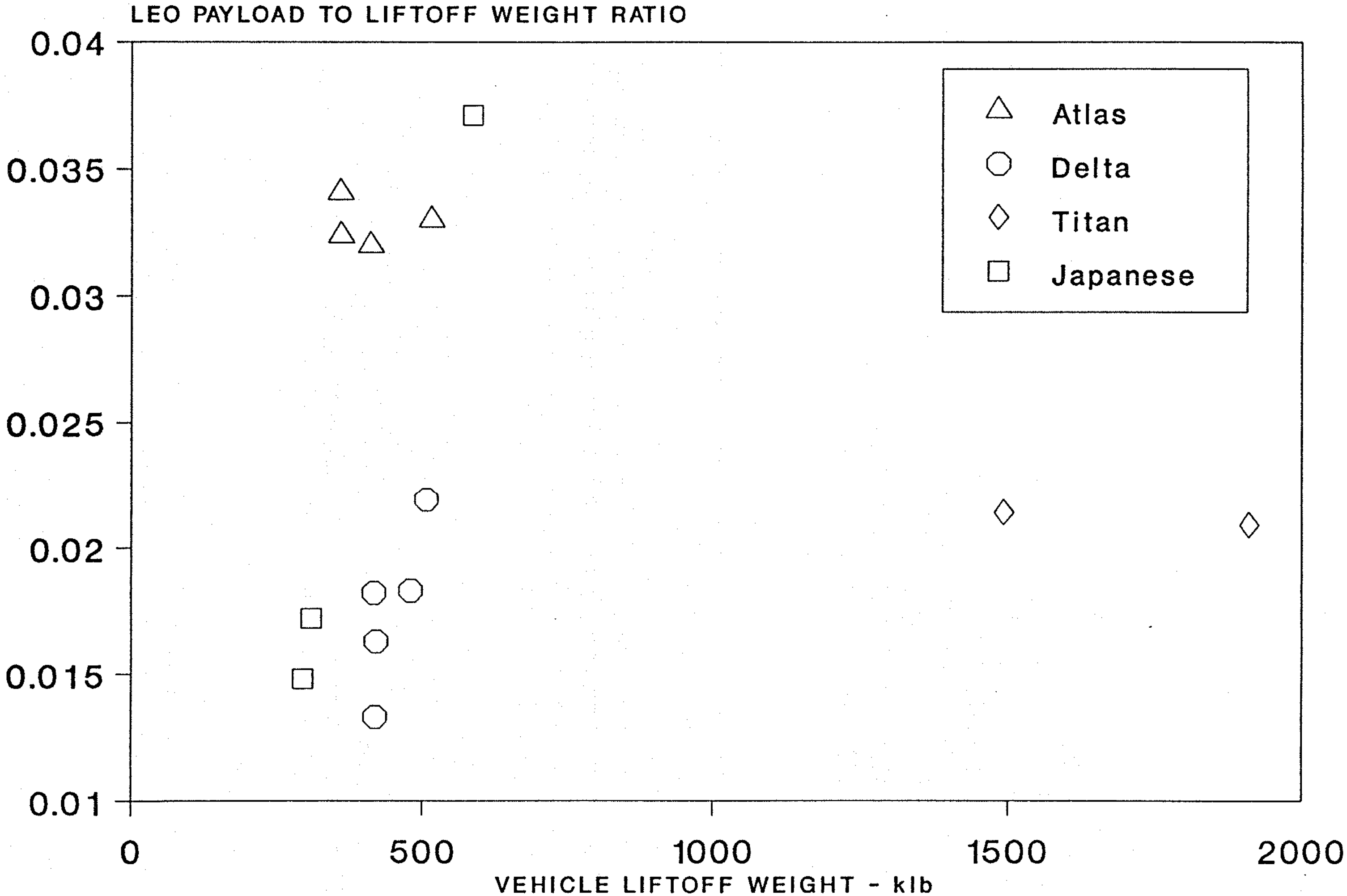

Figure 2.1 Vehicle Payload Efficiency

Table 2.1
H-I Second Stage Characteristics

LENGTH	33.8	FT
DIAMETER	8.2	FT
STAGE WEIGHT	23,400	LB
PROPELLANT WEIGHT	19,200	LB
BURN DURATION	370	SEC
PROPELLANT-TO-STAGE WEIGHT RATIO	0.82	
TANK MIXTURE RATIO	5.31	
ENGINE MODEL: LE-5		
ENGINE THRUST (VACUUM)	23,200	LB
ENGINE SPECIFIC IMPULSE	445	LB-SEC/LB
ENGINE MIXTURE RATIO, SINGLE BURN	5.5	
ENGINE MIXTURE RATIO, RESTART	5.6	

engine is used for fuel tank pressurization, and cryogenically stored helium warmed by an engine heat exchanger is used for oxidizer tank pressurization. After engine shutdown, tank pressure is reduced to keep the LH_2 temperature at an acceptable level to meet engine net positive suction pressure requirements for restart. The tank pressures during engine operation are controlled by the onboard computer to 46 psia in the LOX tank and 36 psia in the LH_2 tank by pressure transducers and pressurization valves.

For restart, the LE-5 engine is cooled by bleeding both propellants through the engine and out the chilldown ports. During stage static firing tests, it was determined that pump chilldown for the second burn condition was more difficult than initially expected. The tests provided data necessary to establish an acceptable pump chilldown procedure, including intermittent chilldown to suppress pump temperature rise during the coast period.

H-II Launch Vehicle

The H-II is planned to serve as NASDA's main space transportation system in the 1990s to meet the requirements for launching larger satellites at lower cost while maintaining a high degree of reliability. It consists of a two-stage core vehicle and two solid rocket boosters. Both stages of the core vehicle use liquid oxygen and liquid hydrogen propellants. The first stage is powered by the LE-7 engine now under development, and the second stage by the LE-5A, which is an improved version of the LE-5 engine used on the H-I vehicle. The solid rocket motor is a new design made up of four segments per booster. It provides thrust vector capability with a movable nozzle using a flexible joint.

The H-II first stage propulsion systems are shown in Figure 2.2 (Ref. 2.28). The basic characteristics of the stage are shown in Table 2.2.

Table 2.2
H-II First Stage Characteristics

LENGTH	95.1	FT
DIAMETER	13.1	FT
STAGE WEIGHT	214,000	LB
PROPELLANT WEIGHT	187,000	LB
BURN DURATION	316	SEC
PROPELLANT-TO-STAGE WEIGHT RATIO	0.88	
STAGE USABLE PROPELLANT RATIO	5.75	
ENGINE MODEL: LE-7		
ENGINE THRUST (VACUUM)	265,000	LB
ENGINE SPECIFIC IMPULSE	449	LB-SEC/LB
ENGINE MIXTURE RATIO	6.0	

The propellant tanks are of almost identical design except for sidewall length and wall thickness. They are covered with spray-on foam insulation for thermal control. Installed inside the tanks are anti-slosh baffles, continuous level sensors for propellant loading, point level sensors for depletion detection, temperature sensors, and pressure sensors. During mainstage, the LH_2 tank is pressurized to 50 psia with GH_2 tapped from the engine. Cryogenically stored helium, warmed by an engine heat exchanger, is used to pressurize the oxidizer tank to 63 psia. The pressure is regulated by redundant shutoff valves controlled by the onboard computer based upon data from three redundant pressure transducers.

Helium for tank pressurization, purges, and valve actuation is stored in ambient temperature bottles at 4400 psi and in cryogenic bottles inside the fuel tank at 3000 psi. Helium from the ambient bottles is supplied from a high-pressure regulator at 2200 psi for pneumatic operation of the main engine, auxiliary engines, and tank system valves, and from a low-pressure regulator at 500 psi for engine purges and pneumatic operation of other tank valves. A pogo suppression accumulator, using helium, is mounted on the engine immediately upstream of the oxidizer turbopump. The main engine and turbopumps are chilled for start by dump cooling LOX and LH_2 through chilldown bleed valves. The main engine is

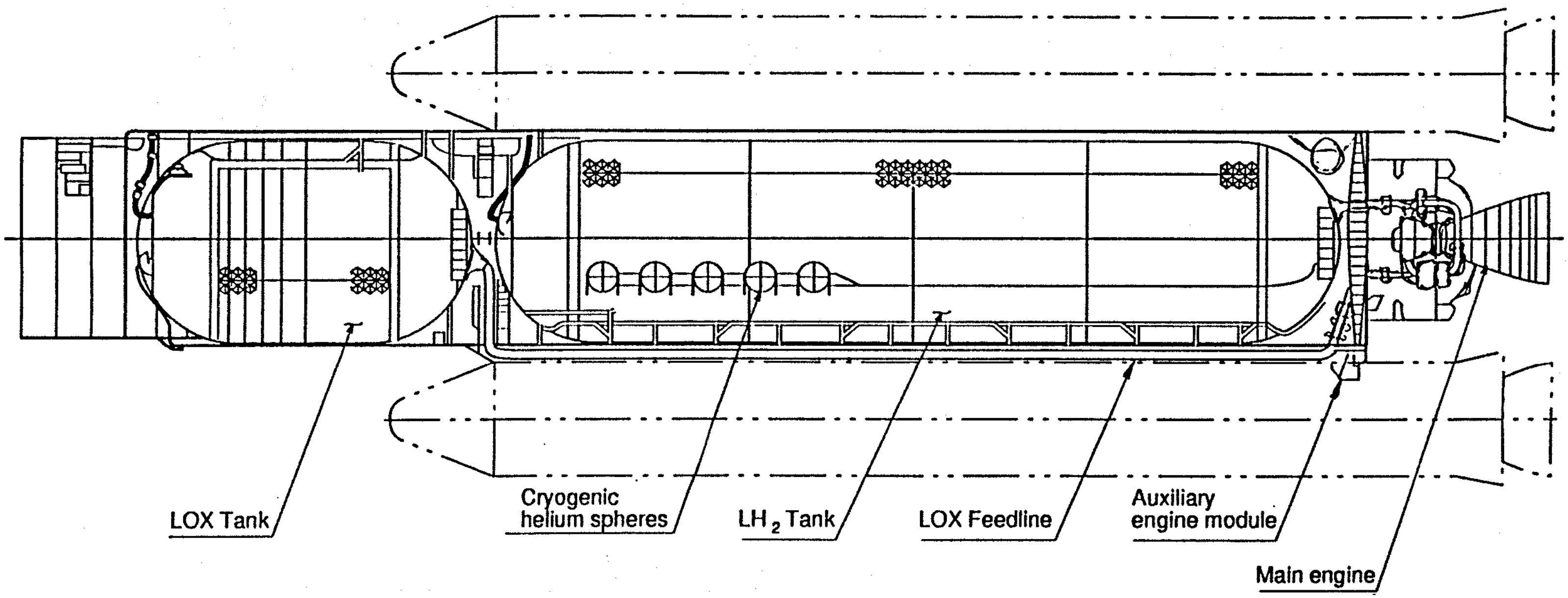

Figure 2.2 H-II Rocket First Stage Propulsion System

gimbaled $\pm$ 7 degrees by hydraulic actuators to provide pitch and yaw thrust vector control. Power for the hydraulic pump is provided by an auxiliary turbine powered by GH_2 used for fuel tank pressurization.

The H-II second stage propulsion systems, shown in Figure 2.3 (Ref. 2.27), is an upgraded version of those used on the second stage of the H-I vehicle. The basic design approach was to use the verified H-I design as much as possible, and to maximize usage of common components such as sensors, solenoid valves, regulators, and helium spheres on both the first and second stages. The basic characteristics of the stage are shown in Table 2.3.

Table 2.3
H-II Second Stage Characterization

LENGTH	31.8	FT
DIAMETER	13.1	FT
STAGE WEIGHT	37,000	LB
PROPELLANT WEIGHT	30,900	LB
BURN DURATION	525	SEC
PROPELLANT-TO-STAGE WEIGHT RATIO	0.83	
STAGE USABLE PROPELLANT RATIO	4.9	
ENGINE MODEL: LE-5A		
ENGINE THRUST (VACUUM)	26,500	LB
ENGINE SPECIFIC IMPULSE	451.9	LB-SEC/LB
ENGINE MIXTURE RATIO	5.0	

The oxidizer tank is basically the same design as the H-I second stage except that the tank side wall is elongated, keeping the same diameter, to achieve the 60% increase in oxidizer tank volume. The fuel tank is new, but the diameter and forward dome are common with the first stage tankage. As in the case of the H-I second stage, a common bulkhead is used. External spray-on foam insulation is used for thermal control. Capacitance type, continuous level sensors are installed in the upper section of each tank for propellant loading. Point level sensors are also installed to obtain inflight propellant consumption data and to sense propellant depletion.

During mainstage, the fuel tank is pressurized to 36 psia with GH_2 tapped from the engine while cryogenically stored helium, warmed by an engine heat exchanger, is used to pressurize the oxidizer tank to 43 psia. The pressure is regulated by redundant shutoff valves, controlled by the onboard computer using data from the redundant pressure transducers in each tank. Ambient temperature

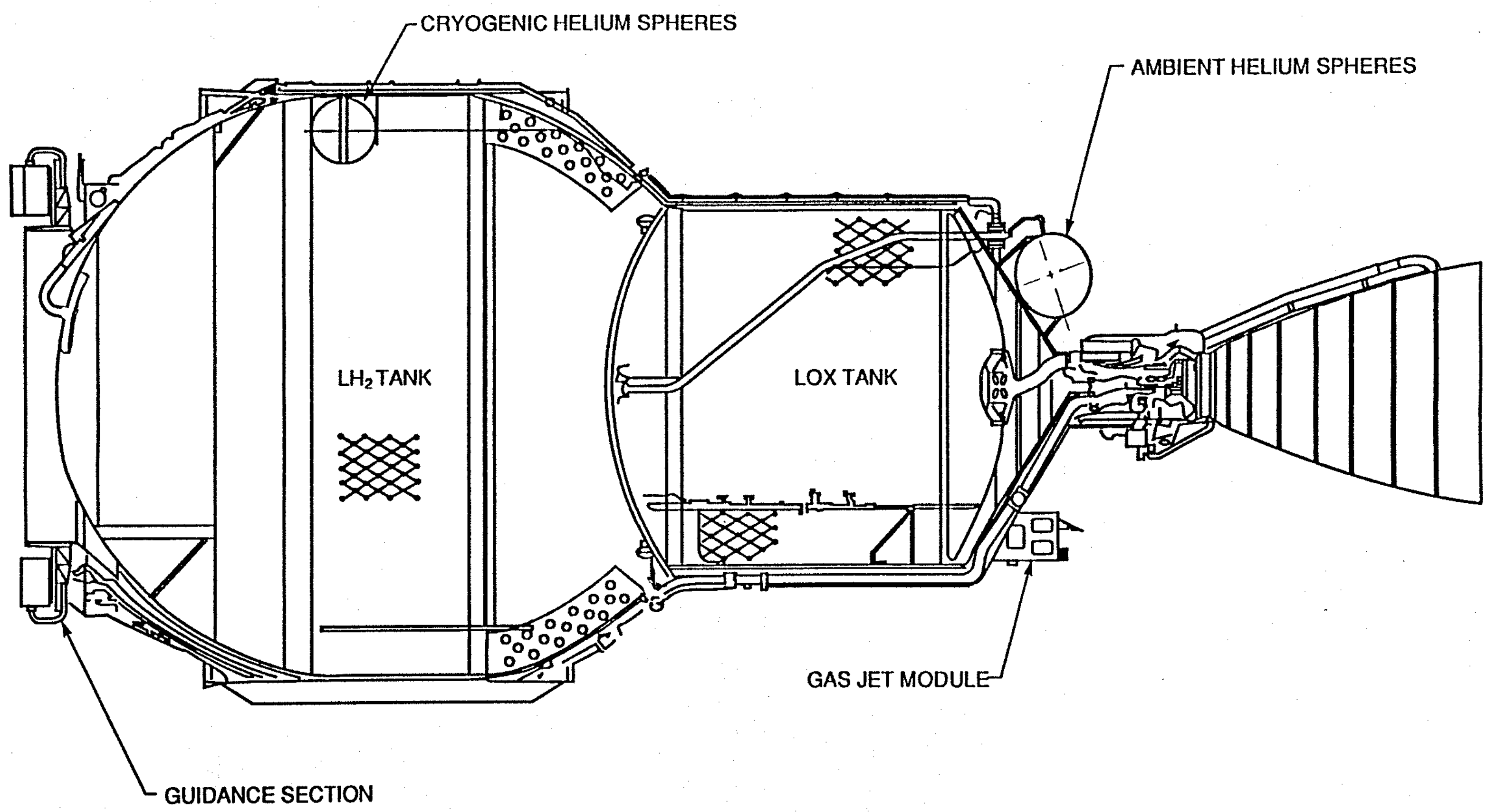

Figure 2.3 H-II Rocket Second Stage Propulsion System

helium is used for tank pressurization prior to engine start. Helium for tank pressurization, purges, and valve actuation is stored in ambient bottles at 4400 psi, and in cryogenic bottles inside the fuel tank at 3000 psi.

In the case of the H-II, a relatively short coast period is anticipated between the first and second burn (800 seconds), with a relatively large amount of propellant remaining in the tanks (43%). As a result, a nonventing coast between the first and second burn is being considered, to provide a weight saving and payload increase. Engine and turbopump cooling would still be provided by an intermittent propellant dump (every 200 seconds), or possibly a continuous low flowrate bleed and pump chilldown just prior to restart.

Aerospace America (Ref. 2.3) reports that, in addition to completing the development of the H-II, NASDA is focusing on reducing the launch cost of the H-II. This reduction is reported to be related to marketing the H-II and increasing production to achieve economies of scale. Other signs of flexibility to support marketing efforts are plans to offer an alternative payload fairing with a 15-foot interior diameter for compatibility with the U.S. Space Shuttle, as well as the current 12-foot interior diameter.

Vehicle Propulsion System Future Directions

The launch vehicle activity outlined above is in support of Japan's long-term policy for space as outlined by the Space Activities Commission. Japan's space development is to be based upon international-level, original, domestic technology (Refs. 2.11 and 2.14). The necessity for future space transportation development in Japan is outlined in information extracted from a presentation by Yamanaka of NAL (Ref. 2.41) as well as from several other sources. Space transportation systems are the fundamental elements of the space infrastructure needed for the promotion of space activity, and propulsion is essential to transportation systems. Continued development of improved systems is viewed as indispensable to Japan's autonomous space activities. A substantial Japanese launch capability also improves the robustness of the world's space transportation capability by providing an alternative means of achieving orbit. The development of both low-cost unmanned cargo and manned transportation is foreseen. The key issues for cargo transportation are payload fraction and cost effectiveness, while for manned transportation they are safety, reliability, comfort, and future economic aspects.

The range of space activities and space transportation elements anticipated in Japan is shown in Figure 2.4 (Ref. 2.14). The emphasis in space transportation from now through the first half of the 1990s includes operation of the H-I vehicle, development of the H-II vehicle to operational status, and the establishment of original, fundamental technology for large rockets which Japan expects to own and operate. From the last half of the 1990s to the beginning of the 21st century, the focus to be on operation of a space station and development of the space infrastructure around the station. Space infrastructure is outlined to include transportation measures like a spaceplane for personnel and cargo, which is expected to promote manned space activity. The plan anticipates the start of lunar and planetary exploration in this period. The vehicle activities projected for these periods, as shown in Figure 2.5 (Ref. 2.14) include completing the H-II development, operating the H-II, starting a spaceplane project and attempting an experimental flight, providing an orbital servicing or maneuvering vehicle, and starting the development of an orbital transfer vehicle to carry a geostationary platform from low earth orbit to geostationary orbit and recover it.

After the beginning of the 21st century, Japan's own space station, expanded from the earlier module in support of the international station, will be constructed and operated, thus forming an international community of space stations. Increased activity in space exploration, including solar and sample return missions is expected. Included in this projection is completion and placing in service of a low-cost transportation system.

Discussions with various people in Japan indicated that internal systems studies and cooperative reviews among the interested agencies were under way to clarify future directions. In addition, research activities and technology development projects to establish the necessary technical base are progressing. Included in these studies are ways to enhance the H-II. Various configurations are being investigated, including development and launch costs. The focus of this activity is indicated as doubling (or tripling) the H-II payload capability into geosynchronous orbit, and cargo transportation to and from the space station, co-orbiting platforms, and polar platforms.

Several derivatives of the basic H-II configuration, as shown in Figure 2.6 (Ref. 2.1), are being investigated to meet the varied future demands. As shown, these operating concepts include adding additional solid rocket boosters, using liquid rocket boosters, or transitioning to a two-stage rocket plane. Several different types of engines are being evaluated (to be discussed in the section below entitled "Rocket Engine Future Direction"), as are different propellant configurations, including hydrogen, methane, and kerosene (RP-1) fuels with liquid oxygen.

Another focus of these studies is spaceplane research. Research and development is in progress at the Institute of Space and Astronautical Science (ISAS), the National Aerospace Laboratory (NAL), and the National Space Development Agency (NASDA). This activity has a somewhat different focus at each sponsoring agency.

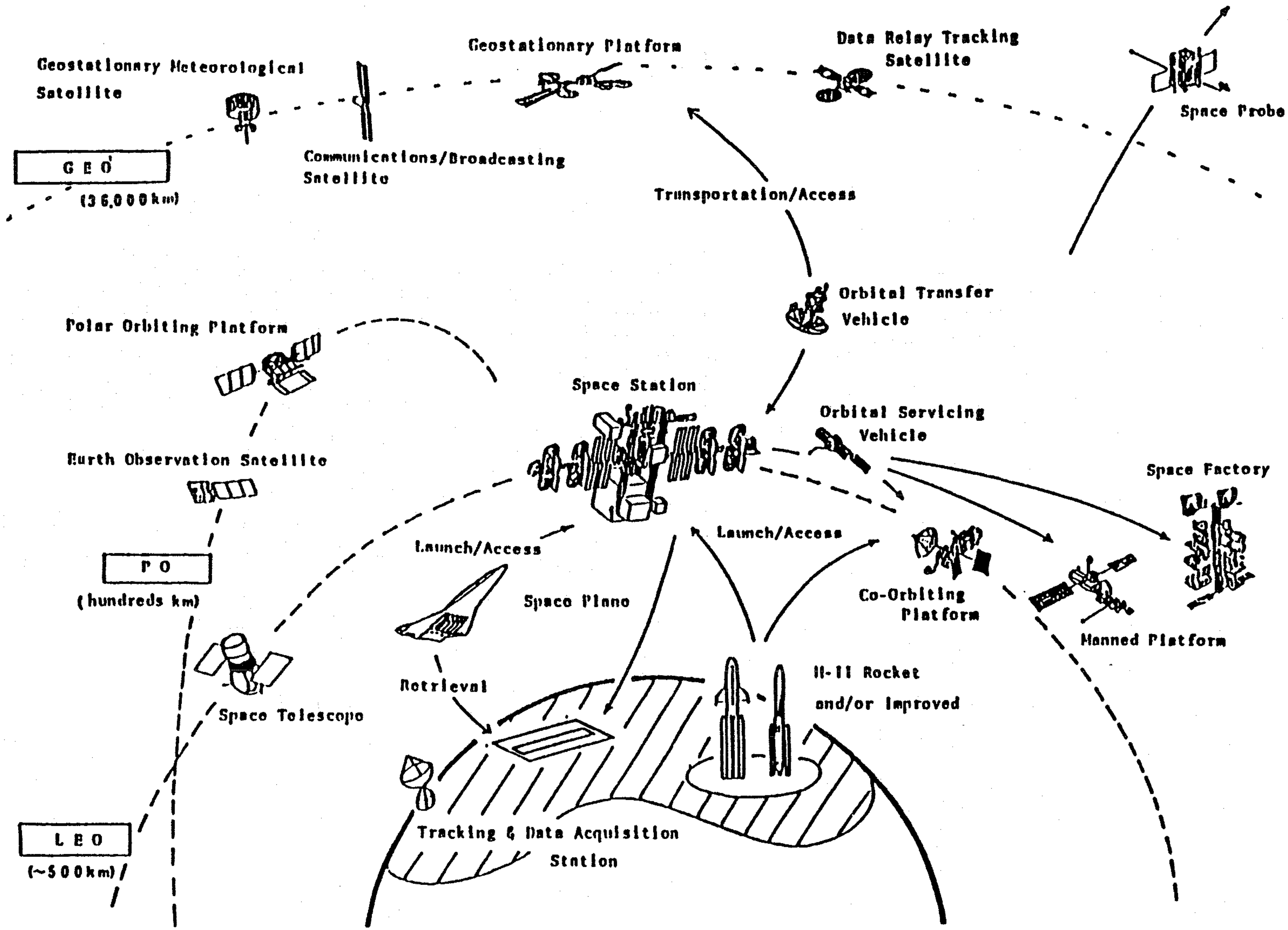

Figure 2.4 Japanese Space Activities in the 21st Century

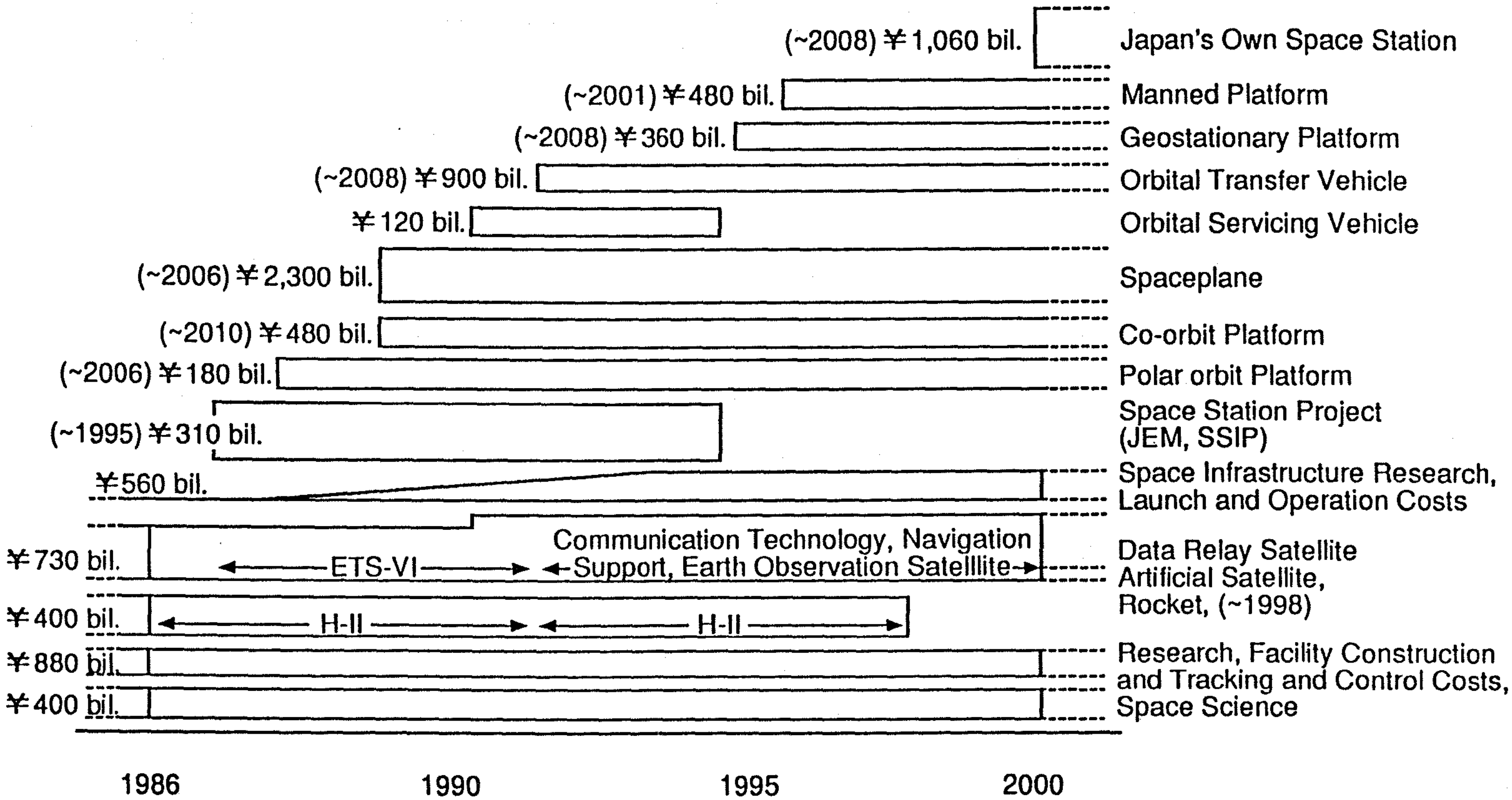

Figure 2.5 Major Japanese Projects

NASDA is focusing on the development of the H-II Orbiting Plane (HOPE), which is an unmanned, reusable, winged space vehicle to be launched by the H-II rocket. The planned missions are recovery of experimental products from the Space Station (Japanese Experiment Module, JEM), supply logistics to and return materials from co-orbiting and polar platforms, conduct experiments and tests in the cargo bay, and establish the technology for a future spaceplane including rendezvous and docking, deorbiting, reentry, and landing. The weight budgeted for propellants includes fluids for orbit maneuver and attitude control, fuel cell reactants, and other consumables used prior to atmospheric reentry. The basic characteristics of the vehicle are shown in Table 2.4.

Table 2.4
HOPE Stage Characteristics

LENGTH	37.7	FT
WINGSPAN	20.5	FT
WING AREA	290	FT^2
LAUNCH WEIGHT	19,400	LB
RETURN WEIGHT	15,400	LB
PAYLOAD WEIGHT	2,600	LB
PROPELLANT WEIGHT	4,000	LB

ISAS has been conducting research and exploratory development activity in support of a Highly Maneuverable Experimental Space Vehicle (HIMES), which is conceived as a multipurpose, reusable, sounding rocket. The primary focus for the vehicle appears to be on testing advanced space transportation system technology. It is equipped with a liquid oxygen and liquid hydrogen main propulsion system using two gimbaled, throttleable rocket engines. The launch concept described is sled, then-sled-plus rocket acceleration to 420 ft./sec., which is sufficient to aerodynamically lift HIMES for powered ballistic flight. References were found which describe HIMES as a test vehicle for a future vertical takeoff stage; however, no launch vehicle descriptions were obtained. Following powered ballistic flight to 130 nautical miles, the mission is completed by reentry and horizontal landing. The basic characteristics of the vehicle are shown in Table 2.5.

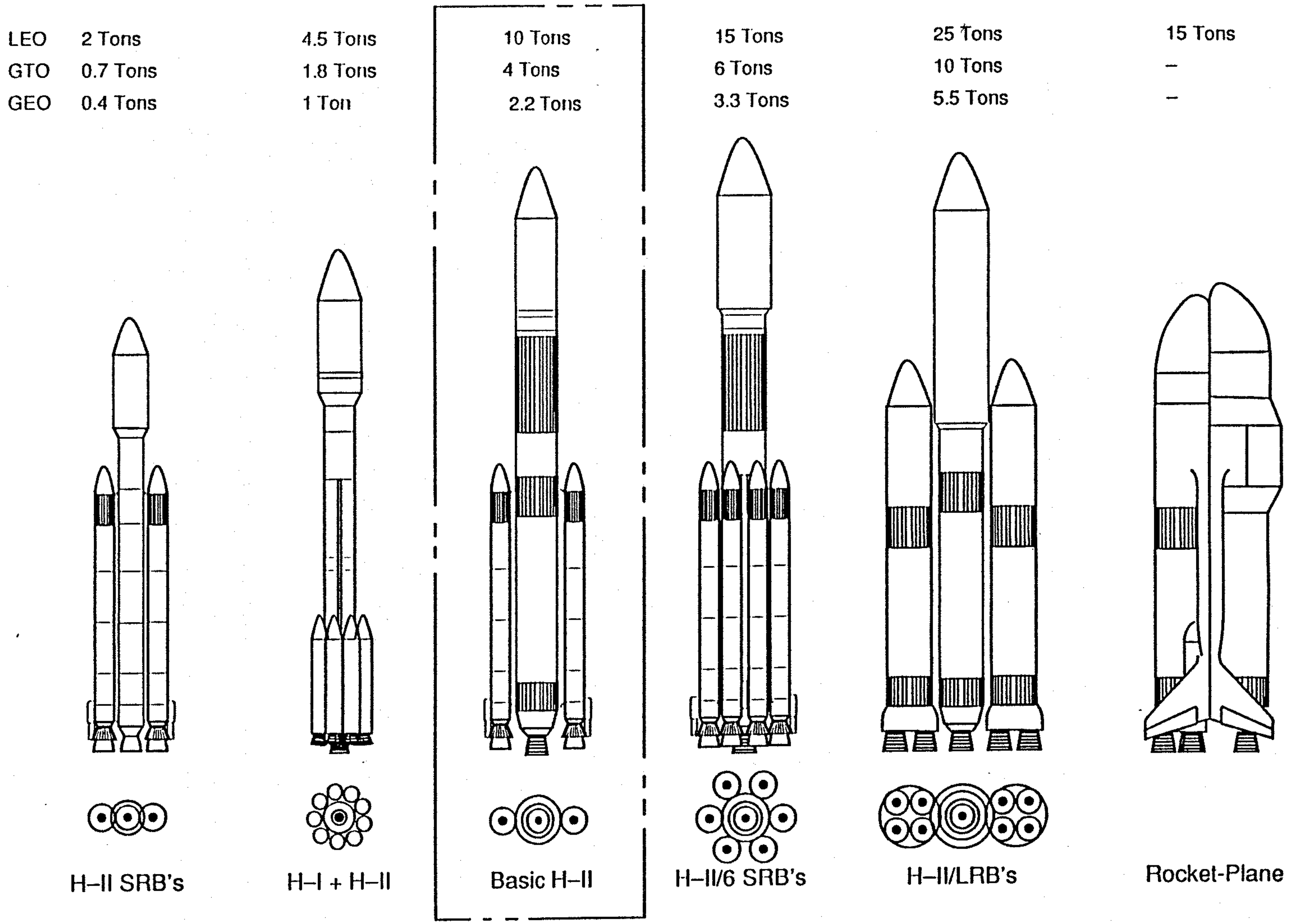

Figure 2.6 Concepts of H-II Derived Vehicles

Table 2.5
HIMES Stage Characteristics

LENGTH	44.6	FT
WINGSPAN	30.6	FT
WING AREA	287	FT^2
LAUNCH WEIGHT	30,300	LB
RETURN WEIGHT	7,200	LB
PAYLOAD WEIGHT	1,100	LB
PROPELLANT WEIGHT	23,100	LB

NAL is investigating a hypersonic flight manned research vehicle focused on a reusable winged vehicle for horizontal takeoff and landing with reliability and operational flexibility comparable to that of an airplane. The target mission requirements are 270 nautical mile orbit, 5-day orbital stay time, and 4,400 lb. payload which includes a crew of four. A baseline single-stage-to-orbit configuration with the basic characteristics shown in Table 2.6 is described, although evaluation of alternatives including two-stage-to-orbit, expendable boosters, and expendable external tanks is identified. A two-step experimental aircraft approach has been described by Yamanaka (Ref. 2.40). The first step would use a liquid air engine based upon the LE-7 technology to evaluate the boost phase flight regime (Mach number of 0 to 8). Another option could be a ducted rocket/ejector rocket. The second step would be to integrate scramjet engines into the experimental aircraft to investigate the orbiter regime.

The evaluation of these alternative approaches to a spaceplane is underway by a liaison group from NAL, ISAS, and NASDA. This liaison group, described by Yamanaka (Ref. 2.41), is to coordinate the research activities as well as assess their results to evaluate concept feasibility. Yamanaka stated the project will not proceed into the engineering model/design feasibility stage until the concept looks possible. It was expected that this assessment will culminate in a government decision in the 1992-1993 time frame.

Table 2.6
Prototype Single-Stage-to-Orbit Stage Characteristics

LENGTH	213	FT
WINGSPAN	81	FT
LAUNCH WEIGHT	773,400	LB
RETURN WEIGHT	222,700	LB
PAYLOAD WEIGHT	4,400	LB
PROPELLANT WEIGHT	538,800	LB

LIQUID ROCKET ENGINES

Early efforts in rocket engine development were collaborative efforts with the United States using technology under license from U.S. manufacturers. For example, the first liquid rocket engine developed by NASDA was the LE-3 engine. It was a pressure-fed engine using nitrogen tetroxide and 50/50 hydrazine/unsymmetrical dimethyl hydrazine blend for propellants, and delivered a thrust of 11,900 lbs. with a specific impulse of 285 lb.-sec./lb. in vacuum. Rocketdyne assisted the development through evaluation and review of the design and test results. The second engine was the AJ10-118FJ used for the second stage of the N-II rocket. It was an improved version of the Aerojet AJ10-118F developed for NASDA to meet the N-II mission requirements.

A preliminary study of cryogenic propulsion, using hydrogen and oxygen as the propellants, was started in 1971 by ISAS. This activity preceded the development of a 15,000 lb. thrust experimental gas generator cycle engine. This cycle was selected to ease the development despite the lower performance. These studies had progressed to where detailed design of the components was initiated in 1975. By 1979, all the major components had been tested, and in 1980, were integrated into an engine system and tested to verify performance. This ISAS project was followed by initiation of the development study of a 22,000 lb. thrust experimental engine in 1979, which was tested in 1981. The gas generator cycle was also used for this higher thrust engine. In 1983, work on a high-pressure expander cycle engine, the HIPEX, was initiated. Following successful component testing, the engine system was integrated and tested in 1987 to verify the concept.

Meanwhile, NASDA started the development of the LE-5 engine in 1977 for use on the second stage of the H-I vehicle. This engine, which was also based upon the gas generator cycle, progressed through the Design Feasibility, Design Verification, and Qualification phases with the final Post-Qualification Review in

November 1985. In September 1985, some feasibility tests of an expander bleed cycle engine built from LE-5 components were conducted. The satisfactory test results verified the concept and supported proceeding with development of the LE-5A for the second stage of the H-II. Qualification was projected to be complete in early 1990, but now is scheduled to be completed in early 1991 because of facility troubles. Based upon the successful LE-5 development, NASDA also decided to proceed with the development of a large booster engine for the final stage of the H-II. This engine, designated the LE-7, employs a staged-combustion cycle. The Component Development, or Design Feasibility, Phase was initiated in 1984 with qualification now projected to be complete in 1992.

LE-5 Rocket Engine

The LE-5 gas generator cycle engine was the first Japanese cryogenic engine qualified for flight. The development and qualification activity, started in 1977, was carried out by NASDA, in cooperation with NAL and ISAS.

The program was accomplished in three phases. The Design Feasibility Phase, started in 1977 and completed in 1982, included component development and system tests of prototype engines. Critical components were tested to obtain design data and confirm design parameters. A breadboard engine using nonflight design plumbing and three prototype engines were tested to establish the functional baseline, including the start sequence and the nominal performance and operating requirements. This was followed by the Design Verification Phase, initiated in 1981 following completion of the preliminary system tests, and completed in 1984. Two series of flight type engines using a total of five engines were tested to establish and verify the flight configuration. Testing included stage and altitude tests to verify compatibility and functional capability including restart. The final Qualification Phase, started in 1983 and completed in 1985, used four engines fabricated in parallel to the first flight engine and included engine firing tests, stage firing tests, engine vibration tests, and component qualification tests. Through qualification, the ground test program was reported to have totaled 387 tests and 32,536 seconds on 11 engines. (Ref. 2.39)

Some of the technical characteristics of the LE-5 are listed in Table 2.7. The engine has a multiple restart capability, but does not include mixture ratio control for propellant utilization. Sequence control is accomplished by a microcomputer sequencer and a pneumatic package to control the pneumatic functions and main propellant valves. A hydraulic pump driven by a hydrogen gas turbine provides power for the thrust vector control actuators.

Table 2.7
LE-5 Engine Characteristics

THRUST (VACUUM)	23,200	LB
SPECIFIC IMPULSE (VACUUM)	445	LB-SEC/LB
MIXTURE RATIO (O/F)	5.5	
MAIN CHAMBER PRESSURE	520	PSIA
AREA RATIO	140	
CYCLE: GAS GENERATOR		
DRY WEIGHT	560	LB
LENGTH	104	IN
DIAMETER	65	IN

The propellant flow schematic is shown in Figure 2.7 (Ref. 2.39). In mainstage, liquid hydrogen from the propellant tank passes through the fuel pump (FTP) raising the pressure to 800 psia, regeneratively cools the combustion chamber (MC) in a downpass mode to an area ratio of 8.5, and then is ducted to the main injector. Liquid oxygen passes through the oxidizer pump (LTP) raising the pressure to 750 psia and is ducted directly to the main injector. Propellants are tapped from the main valves (MFV and MLV), ducted to the gas generator, and are burned at a pressure of 375 psia and a nominal mixture ratio of 0.85. The hot gases generated, about 1.8% of the total inlet propellant mass flow, flow through the fuel and oxidizer pump turbines in sequence, providing the power to pressurize the propellants, and then are injected into the supersonic region at the main chamber, nozzle extension interface. The nozzle extension (NE) is attached to the chamber at the 8.5 area ratio, carrying the overall area ratio of 140. This extension is cooled by approximately 3% of the hydrogen flow which is routed from the outlet of the combustion chamber coolant through the 650 nozzle tubes, exhausting through small supersonic nozzles at the exit of each tube.

The gas generator orifices just downstream of the valves (GFV & GLV) and the oxygen turbine bypass orifice are used to adjust thrust and mixture ratio to the rated values. The required tolerance is $\pm 3\%$ on thrust and $\pm 1\%$ on mixture ratio. Reference 2.39 indicates the repeatability for mixture ratio meets the requirement with a standard deviation of 0.36% (based upon tests on five engines), presumably once steady state is achieved. Reference 2.43 indicates that more than 100 seconds were necessary to achieve steady state on thrust while mixture ratio and specific impulse converged to a constant after 40 seconds, except for a few cases where specific impulse increased continuously for the entire duration of 370 seconds.

For starting, propellants chill down the two turbopumps and the gas generator feedlines by bleeding through the fuel chill valve (FCV) and the oxidizer chill valve (LCV) and draining overboard. Upon receipt of the start signal, the torch igniters of the main chamber and gas generator are initiated by opening the igniter valves (GIV and MIV) with the ignition energy supplied by electric spark igniters. The engine start valve (ESV) and the main fuel valve are opened sequentially, allowing fuel to flow under tank pressure through the main combustion chamber, where it picks up heat from the ambient temperature metal to power up the fuel and oxidizer turbine. About 25% of the hydrogen flowing through the chamber goes through the start valve powering the pump turbines and eventually is exhausted into the nozzle. Thus, the start energy is supplied by tank pressure hydrogen heated in the chamber coolant channels rather than using start tanks or a solid propellant turbine spinner. The main LOX valve (MLV) is opened some 0.5 seconds after the main fuel valve supplying oxygen to the main injector where the oxygen burns with the remaining 75% of the fuel not flowing through the turbines.

These burning gases supply additional heat to the chamber walls, increasing the energy available to power the turbines. The power supplied to the turbines increases the turbopump speeds, which increases the pump discharge pressure and causes the engine power and combustion pressure to increase. Sufficient energy is available in this expander cycle mode to power the engine to 70%. The gas generator valves are time sequenced open by the engine electronic sequencer once a main chamber pressure of 50% of the steady state valve is sensed. Similarly, the sequencer commands the start valve to close at the appropriate time, converting the system to the gas generator power cycle and continuing the transient to the main stage power and chamber pressure setting. The sequencer also closes the gas generator and main chamber igniter valves, terminating flow to the igniters.

In flight, the vehicle acceleration will apparently be sensed to generate the signal to the sequencer to initiate conversion to gas generator power cycle instead of chamber pressure which was used during ground testing. The sequence from start to steady mainstage conditions appears to take about 6 seconds at the nominal fuel inlet pressure. As expected, variation of the fuel inlet pressure from nominal will influence the buildup time to 50% power with a variation of almost one second (Ref. 2.43). This method does not require precooling of the chamber to avoid pump surge. Neither is an auxiliary start energy system required, thereby facilitating restart.

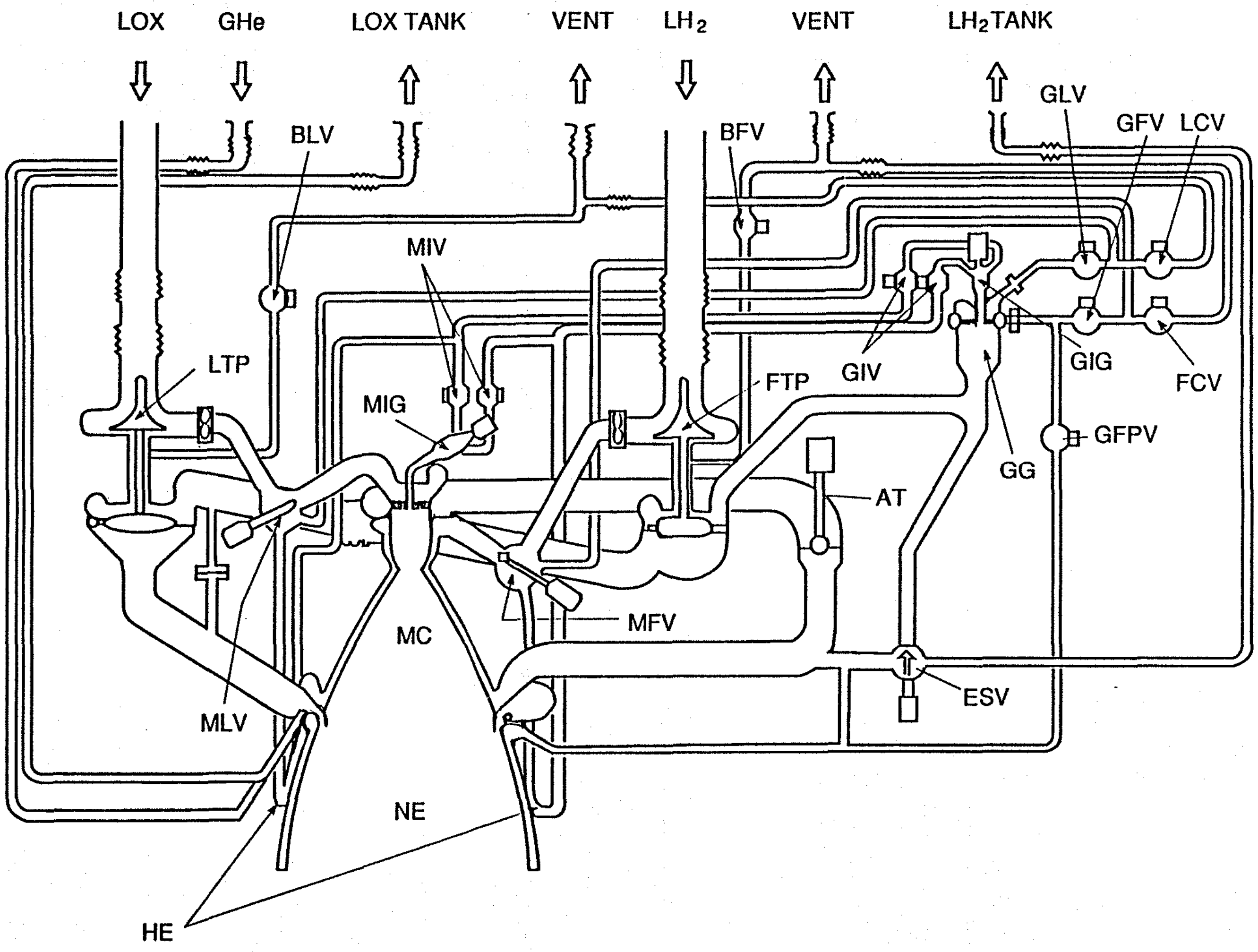

Figure 2.7 System Diagram of LE-5 Rocket Engine

Shutdown is accomplished by closing the gas generator valves upon receipt of the shutdown command, terminating turbine power. Once turbine power is terminated, the flow to the main chamber quickly drops, causing a rapid decay in chamber pressure. The main valves are time sequenced closed after chamber pressure decays, with the fuel valve delayed to provide for a fuel-rich shutdown. The sequence from stop signal to negligible thrust appears to take about 3.5 seconds.

System features worth noting, in addition to those described above, include the following: A hydrogen gas turbine powered hydraulic pump (AT) is used to provide power for the thrust vector control actuators. The main chamber and gas generator igniters operate on gaseous propellants generated by two tubes 23 feet long, circumferentially attached to the nozzle extension. Gas is bled to the gas generator propellant inlets, gaseous hydrogen through the GFPV valve from the chamber coolant exit, and gaseous helium (not shown on Fig. 2.7) to the oxidizer, to prevent ice formation and burnout of the gas generator body.

LE-5A Rocket Engine

The LE-5A is an expander bleed cycle engine derived from the LE-5. It is currently being qualified for use on the second stage of the H-II vehicle. The modification was based upon using original LE-5 engine components, as is, or with minimal changes, but changing the power cycle to achieve a 15% uprating in thrust. The basic feasibility of this approach was confirmed in a series of six tests in 1985 using modified LE-5 hardware and earlier subscale nozzle tests. Since the component designs were so close to the LE-5, this engine test series was used to conclude the Design Feasibility Phase. The Design Verification Phase, started in 1986, is expected to conclude with stage captive firing tests in 1990. Two engines are apparently planned for this phase. The Qualification Phase, also planned for two engines, was started in 1987 and is also expected to conclude in 1990. A total of 106 starts and 10,950 seconds of engine test are projected as being sufficient to achieve a high reliability level comparable to the LE-5 engine.

The basic technical characteristics of the LE-5A are listed in Table 2.8. The new parts needed for the LE-5A are the combustion chamber, the nozzle extension, the waste valve, and some engine ducts. The throat area of the combustion chamber was increased some 10% by reshaping the convergent and throat region to match the remaining portion of the LE-5 tube. In addition, the acoustic cavities used on the LE-5 were deleted. The nozzle extension was redesigned to a two-pass coolant path and shortened to 80% of the LE-5 length to reduce the heat transfer area, with the remaining length being a radiation cooled extension. The turbine exhaust gas manifold was moved from the beginning of the cooled nozzle extension to the end. A waste valve was added to dump turbine power at shutdown. The oxidizer flow area of the main injector was increased 20% by enlarging the orifice diameter to keep the injection velocity low and minimize the load on the oxidizer pump. The turbopumps are basically the same as the LE-5's, except for a small increase in oxidizer impeller diameter and an increase in the number of fuel impeller blades.

Table 2.8
LE-5A Engine Characteristics

THRUST (VACUUM)	26,500	LB
SPECIFIC IMPULSE (VACUUM)	452	LB-SEC/LB
MIXTURE RATIO (O/F)	5.0	
AREA RATIO	130	
CYCLE: EXPANDER BLEED		
DRY WEIGHT	540	LB
LENGTH	104	IN
DIAMETER	65	IN

The propellant flow schematic is shown in Figure 2.8 (Ref. 2.5). The mainstage propellant flowpaths are very similar to the LE-5 except for the elimination of the gas generator and its propellant flow lines. For the LE-5A, the gas necessary to power the turbine in series is tapped from the main chamber coolant discharge, passed through the nozzle and waste valve (WV) and into the fuel turbine inlet. From there, the gas passes through the fuel turbine, the LOX turbine, and into the exhaust manifold at the exit to the cooled nozzle. Thrust control, while keeping the metal of the nozzle extension at an acceptable temperature, is probably accomplished by adjusting the orifice in the nozzle coolant tapoff line in conjunction with the orifice which bypasses both turbines. Mixture ratio control is probably accomplished by the orifice bypassing the oxidizer turbine, as in the LE-5.

Starting the LE-5A is accomplished the same way as the LE-5 up to 50% thrust. Instead of sequencing at that point to gas generator power, the LE-5A continues on to full power mainstage without the issues and potential hazards of igniting the gas generator and phasing the power. This greatly simplifies the engine start and control, eliminating the gas generator ignition system and facilitating restart pump chilldown due to lower turbine temperatures.

Shutdown is accomplished by opening the waste valve (WV), which bypasses the nozzle extension coolant flow around both turbines, thereby terminating turbine power. After pump speeds and pressures have decreased to acceptable levels, the main valves can be sequenced closed.

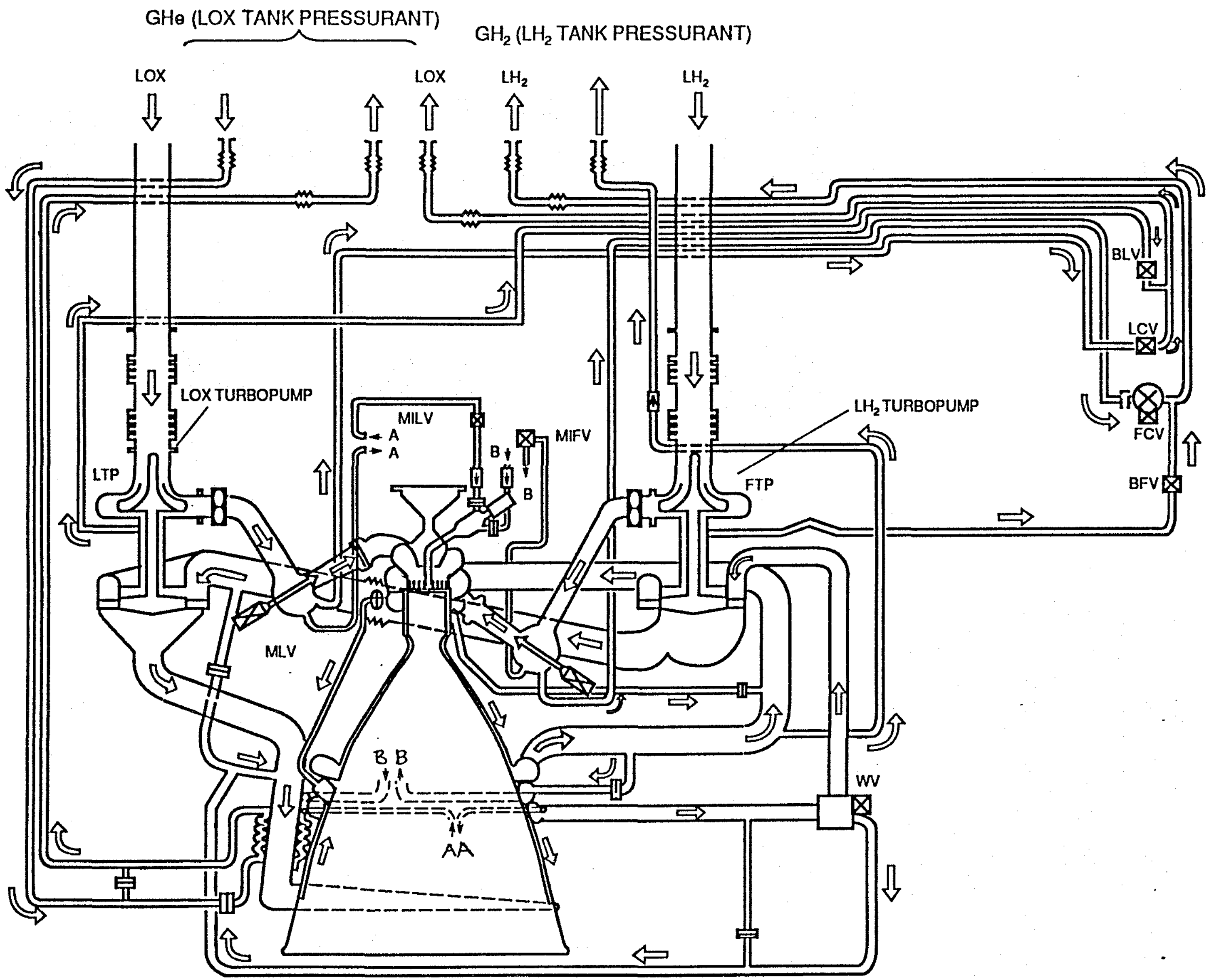

Figure 2.8 System Diagram of LE-5A Rocket Engine

Idle mode operation, which is a low-thrust engine operating point, is useful for producing impulse from the cooldown propellants prior to restarts by burning these propellants in the main combustion chamber under low-pressure conditions. A stable idle mode operating point is achieved by leaving the waste valve open during start. This bypasses the nozzle coolant flow around the turbines and keeps the available pump power low. This operating condition including the transition to normal engine mainstage by closing the waste valve was demonstrated in several engine tests in 1988.

LE-7 Rocket Engine

The LE-7 is a staged combustion cycle, cryogenic engine which is planned for use on the first stage of the H-II rocket. The development and qualification activity, which started in 1984, is being carried out by NASDA in cooperation with NAL. The program employs the phasing concept, similar to the LE-5 and LE-5A programs, but the first phase was split into two--the Design Feasibility Phase and the Prototype Engine Phase--so the total program is composed of four phases instead of three.

The Design Feasibility Phase, started in 1984, was due to be completed in 1987. It includes component development and breadboard system tests to confirm the component designs and performance and establish system feasibility. The system breadboard tests were completed in April 1988, but component testing of the turbopumps is continuing in order to solve the problems identified to date. The Prototype Engine Phase was started in 1986 and is planned to confirm the start and stop sequences and the nominal performance of the engine. This phase was projected to include testing of two engines in the engine test stand and in a battleship stage test facility. Prototype engine tests have been in progress since 1988 and were expected to conclude in 1989. Recent test results (Ref. 2.10) would indicate that additional testing will be required before the start transient can be confirmed.

The Flight Type Engine, or Design Verification, Phase was to start in 1987. Four flight type engines are planned for engine and battleship stage tests to finalize the engine design and confirm the off-design performance. The final Qualification Phase calls for use of four engines fabricated in the same lot with flight engines to accomplish testing of the allocated reliability and flight worthiness. These tests will be accomplished on both the single engine test stand and captive stage firing tests. Major design reviews are scheduled at the end of each phase to confirm the progress and make the necessary decisions regarding the design and improvement modifications for the next phase. Reference 2.2 projected a total of 10 engines, 254 tests and 16,600 seconds would be accomplished to qualify the LE-7 for flight with an acceptable demonstrated reliability level. This had originally been planned for 1991 with the first flight of the H-II planned for January 1992, but component design problems and system transient problems (Ref. 2.41) are likely to postpone these dates by one to three years.

The basic performance characteristics of the LE-7 are listed in Table 2.9. These characteristics were set based upon a feasibility study of the vehicle and engine to optimize vehicle performance and minimize risks, cost, and development schedule. Reference 2.39 reports the selection criteria to be (1) to maximize vehicle payload for minimizing satellite launch cost per unit weight, and (2) to use the LE-5 development technical experience to achieve the cost, risk, and time savings. The engine is a fixed-thrust, fixed-mixture-ratio design with the mainstage steady state values controlled by orifices. Engine sequence control is accomplished by an onboard digital processor, and the control valves are operated pneumatically.

Table 2.9
LE-7 Engine Characteristics

THRUST (VACUUM)	265,300	LB
SPECIFIC IMPULSE (VACUUM)	449	LB-SEC/LB
MIXTURE RATIO (O/F)	6.0	
MAIN CHAMBER PRESSURE	2,132	PSIA
AREA RATIO	60	
CYCLE: PREBURNER		
DRY WEIGHT	3,439	LB
LENGTH	138	IN
DIAMETER	75	IN

The propellant flow schematic is shown in Figure 2.9 (Ref. 2.5). In mainstage, liquid hydrogen from the stage fuel tank passes through the two stages of the LH_2 turbopump where the pressure is increased to 4630 psia. After passing through the main fuel valve (MFV), the flow is split to cool the combustion chamber and the nozzle extension through parallel flow paths. Approximately 46% of the flow cools the combustion chamber in a single up-pass while the remaining 54% cools the two-pass nozzle extension. After recombining the two coolant flows and extracting GH_2 for tank pressurant and other uses, approximately 96% of the hydrogen pump flow remains to produce the engine thrust. This flow is split approximately 84% to the preburner and the other 16% to the turbine bypass orifice for thrust control and the injector face coolant.

Liquid oxygen from the stage oxidizer tank flows through the main impeller of the LOX turbopump where the pressure is raised to 3090 psia. The oxidizer preburner pump, which has a flow rate of approximately 20% of the main pump flow, raises the pressure to 4740 psia. This preburner pump flow is composed of 11% which goes on to the preburner and igniters, and 8% which is recirculated to the preburner pump inlet through the mixture ratio control orifice. The remaining 89% of the LOX flow passes through the main oxidizer valve (MOV) and goes directly to the main injector, except for a small amount extracted as GO_2 for tank pressurant. The preburner is fed with fuel and oxidizer at a mixture ratio of 0.81. After combustion at 3480 psia, the hot products are split into two flows, with 72% flowing to the fuel pump turbine, and the remaining 28% to the oxidizer pump turbine. After expansion through the turbines, the gas streams are ducted into the main injector. The flow through the fuel turbine passes through a heat exchanger to heat the igniter fuel flow and is then combined with the turbine bypass fuel flow before reaching the injector manifold. The oxidizer turbine flow heats the igniter oxidizer flow and the oxidizer tank pressurant flow in separate heat exchangers.

The flow schematics of References 2.16 and 2.39 show the oxidizer tank pressurant as oxygen while the schematic of Reference 2.5 shows it as helium. The thrust and mixture ratio control orifices are adjusted at acceptance test to achieve the targeted values within a thrust tolerance of $\pm 1.5\%$ and a mixture ratio tolerance of $\pm 2\%$. The LE-5 experience was used to set operational limits for component design of $\pm 4.5\%$ for thrust and $\pm 5\%$ of mixture ratio based upon three standard deviation values.

For prestart cooldown, hydrogen flows through the fuel turbopump and is bled overboard from a line shown upstream of the MFV. Oxygen flows through the main pump and through the preburner pump before flowing overboard from the pump recirculation line. The engine control valves for engine start and shutdown are described in Reference 2.31 as the main fuel valve (MFV), the main oxidizer valve (MOV), the preburner oxidizer valve (POV), and the nozzle skirt valve (NSV). The NSV is described as adjusting the hydrogen coolant flow rate to the combustion chamber, but its function relative to start or shutdown or sequence is not defined, and it appears in the schematic of Reference 2.5 as an orifice. All of the valves are described as having nine positions, including closed, which is accomplished by having three pneumatic pistons in series. The stroke, stroke rate, and timing for each piston is adjustable to provide the desired open loop valve sequence. The position of each piston, and therefore the valve position, is controlled by a solenoid valve that controls pressure to the opening port of that piston.

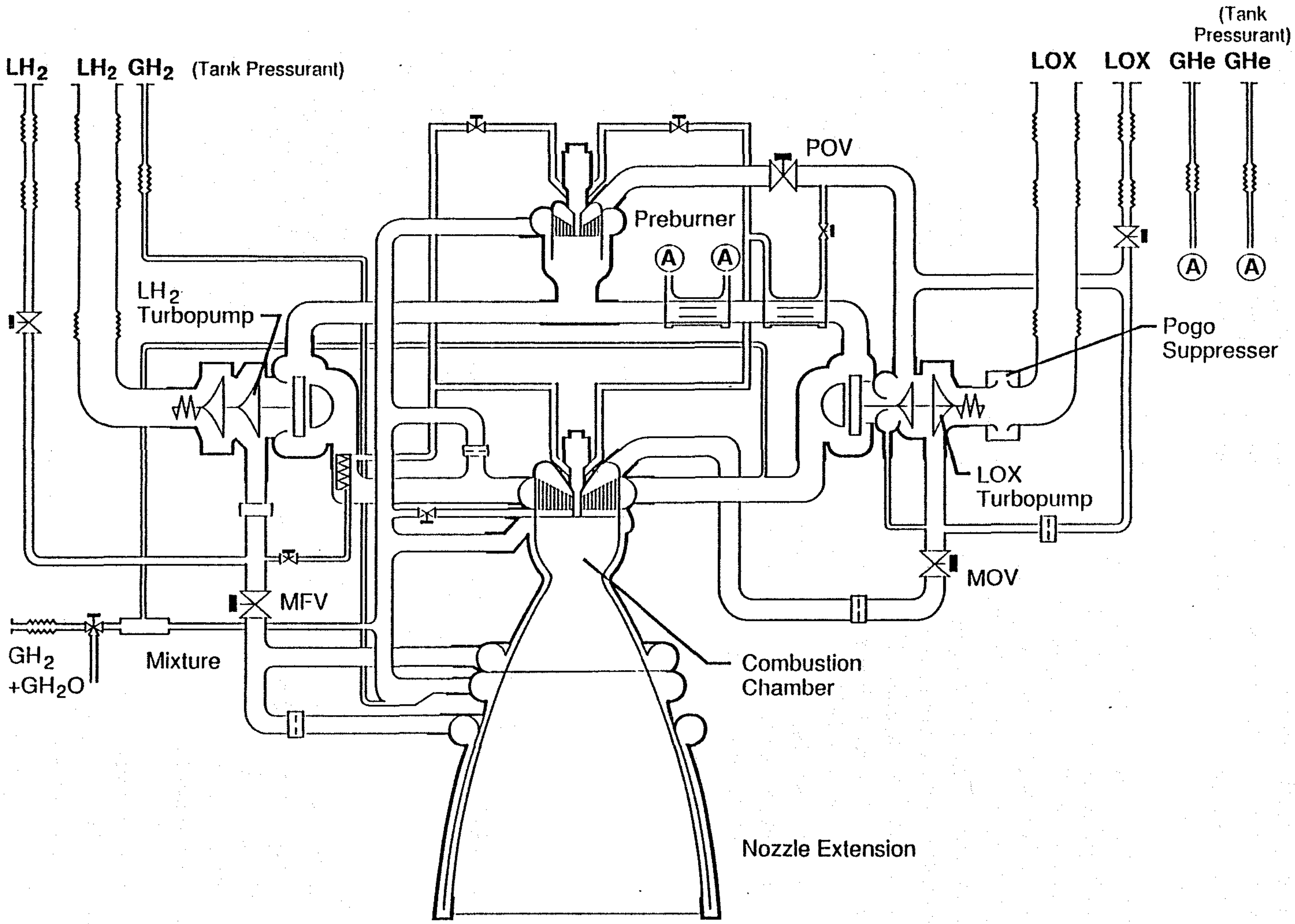

Figure 2.9 System Diagram of LE-7 Rocket Engine

Therefore, three solenoids are used to control the opening sequence of each valve. The MFV, MOV, and POV are ball valves; the NSV type is not described. Also, several other valves are shown, four in the fuel and oxidizer igniter lines and one in the face plate coolant line, implying a start or shutdown function which is not described.

Although the start and shutdown procedures for the LE-7 are still under development, indicators are (Ref. 2.21) that the expander cycle is used to initiate the start. Both the MFV and MOV are apparently opened at about the same time (1.2 seconds) to the first position, allowing flow to start, which begins rotation of both pumps. At about 2 seconds, the MFV starts a ramp to full open at about 3 seconds which accelerates both pumps and increases system pressure. The MOV position is slightly increased in two steps and, at 7 to 7.5 seconds, starts a ramp to full open at 7.5 to 8 seconds. The POV is opened to the initial position at about 6 seconds, so until this time, the engine is operating in an expander cycle mode. At about 9.2 seconds, the POV starts a ramp to full open which is completed at about 10 seconds. At this point, the engine appears to be operating at near mainstage. This start sequence is quite lengthy. Because the engine cycle is closed loop, the start (and shutdown) characteristics are very complicated and Reference 2.42 notes the extreme sensitivity of the MOV and POV opening sequences relative to the buildup characteristics, a sensitivity which seems to be borne out by the test results.

HIPEX Engine

The HIPEX engine is a high-performance expander cycle engine currently under development by ISAS. It has been proposed for use on the HIMES experimental spaceplane and for the orbital transfer vehicle. The preliminary design study was initiated in 1983, and the project has progressed to initial tests of a low-pressure experimental engine, the HIPEX-X01, in 1987. The development activity is projected to progress through a Prototype Engine Phase and a Flight Type Engine Phase prior to the initial test flight in 1995. The Prototype Engine Phase, to accomplish a system performance verification of the engine at full design pressure, was projected to start in 1989 and be complete in 1992. The Flight Development Phase to qualify the engine design for flight is projected to start in 1992 and be complete by the beginning of 1995.

The design characteristics of the HIPEX engine for use on the HIMES vehicle are shown in Table 2.10. These design goals were described in Reference 2.31 as being close to the upper limit attainable by an expander cycle. The nozzle expansion ratio of 35 is based upon optimization to fit the HIMES vehicle flight profile. Specific impulse of the engine could be increased to 460 lbs.-sec./lb. by using a nozzle extension to increase the area ratio to 100.

Table 2.10
HIPEX Engine Characteristics

THRUST (VACUUM)	31,500	LB
SPECIFIC IMPULSE (VACUUM)	435	LB-SEC/LB
MIXTURE RATIO (O/F)	6.0	
MAIN CHAMBER PRESSURE	1,450	PSIA
AREA RATIO	35	
CYCLE: EXPANDER		
DRY WEIGHT	507	LB
LENGTH	66	IN
DIAMETER	24	IN

The flow schematic of the HIPEX-X01 engine is shown in Figure 2.10 (Ref. 2.31). In mainstage, hydrogen flows through the LH_2 turbopump, the fuel valve, the combustion chamber, and the heat exchanger inside the combustion chamber, and then out to the turbine of the LH_2 turbopump. From the fuel turbine, the hydrogen flows through the LOX turbopump turbine and into the fuel manifold of the main injector. Two bypass lines, one containing the thrust control valve and the other containing the mixture control valve, are used to bypass flow around the turbines and control the thrust and mixture ratio of the engine. Oxygen flows through the LOX turbopump and directly to the oxidizer manifold of the main injector. For the subscale HIPEX-X01 engine, the design point chamber pressure was 200 psia with fuel and oxidizer pump discharge pressures of 700 and 300 psia respectively. Flow simulation tests showed excess pressure loss in the heat exchanger due to entrance, exit, and turning losses in the flow passages, but the basic feasibility and heat transfer characteristics were verified. Reference 2.31 shows test data ranging from 23% to 102% of design chamber pressure, and mixture ratios from 4.9 to 7.4.

Start is accomplished simply by opening the fuel valve and oxidizer valve. Data shows that mainstage conditions can be achieved in 6 to 7 seconds with the proper setting on the thrust control valve. Shutdown is accomplished by closing the propellant valves, both cases requiring proper sequencing between the fuel and oxidizer valves to avoid chamber overheating problems.

Design values for the full-scale HIPEX engine compatible with the 1450 psia main chamber pressure, shown in Table 2.10, are as follows: fuel pump discharge pressure, 3650 psia; fuel turbine inlet pressure, 3180 psia; LOX turbine inlet pressure, 2000 psi; fuel turbine inlet temperature, 710°R and LOX turbine inlet temperature, 650°R.

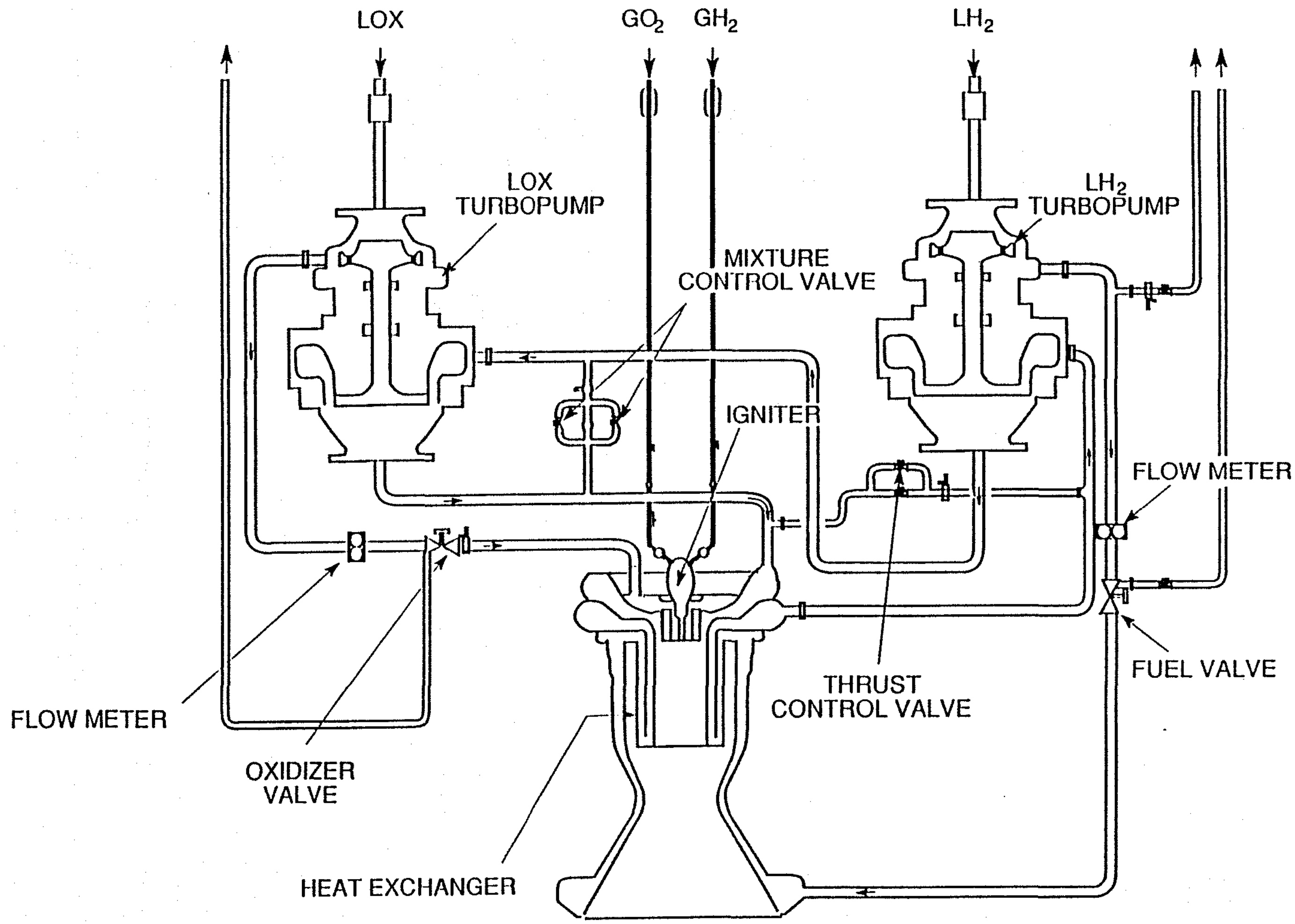

Figure 2.10 System Diagram of HIPEX-X01 Rocket Engine

The basic system differences between the HIPEX design and the HIPEX-X01 are the addition of a nozzle extension to increase the expansion ratio to 35, and the routing of the hydrogen flow through this nozzle extension to cool it and pick up additional heat before going to the turbines. These system differences plus appropriate uprating of the design of several components, primarily the turbopumps and the heat exchanger area, are the changes necessary to achieve the HIPEX design.

Summary of Performance Characteristics

An overall look at the performance characteristics of the Japanese cryogenic engines relative to a number of U.S. rocket engines is shown in Figures 2.11, 2.12, and 2.13. Figure 2.11 displays vacuum thrust-to-weight ratio at comparable installation sizes for the Japanese cryogenic engines just discussed and a number of U.S. rocket engines using different propellant combinations. Figure 2.12 is a similar display for specific impulse for the same set of engines. Figure 2.13 cross-plots thrust to weight and specific impulse on the same chart for the same set of engines. All three figures show that the performance characteristics of the Japanese and U.S. cryogenic engines are very competitive.

Rocket Engine Future Directions

The range of future improvement options, identified as being investigated, were outlined in this Chapter's section, " Vehicle Propulsion System Future Directions," relative to transportation to space. Studies and experimental investigations of rocket engines and components to support these vehicle options were also identified.

One of the key areas was uprating of the H-II to increase its payload capability by a factor of 1.5 to 2.5. The options identified include adding additional solid strap-on motors and converting to a liquid rocket booster. Several engine options listed in Table 2.11 were identified as candidates for the engines of the liquid rocket booster. These propellant combinations and power cycles represent the range of reasonable candidates, based upon past experience in the United States for this type of application. The thrust levels identified would imply that the number of engines per booster could range from two to the four shown on Figure 2.6.

In addition to concept studies of the engine options identified above, experimental investigations were found to be in progress to establish the needed technical base for high-pressure booster type engines. The literature includes articles on main

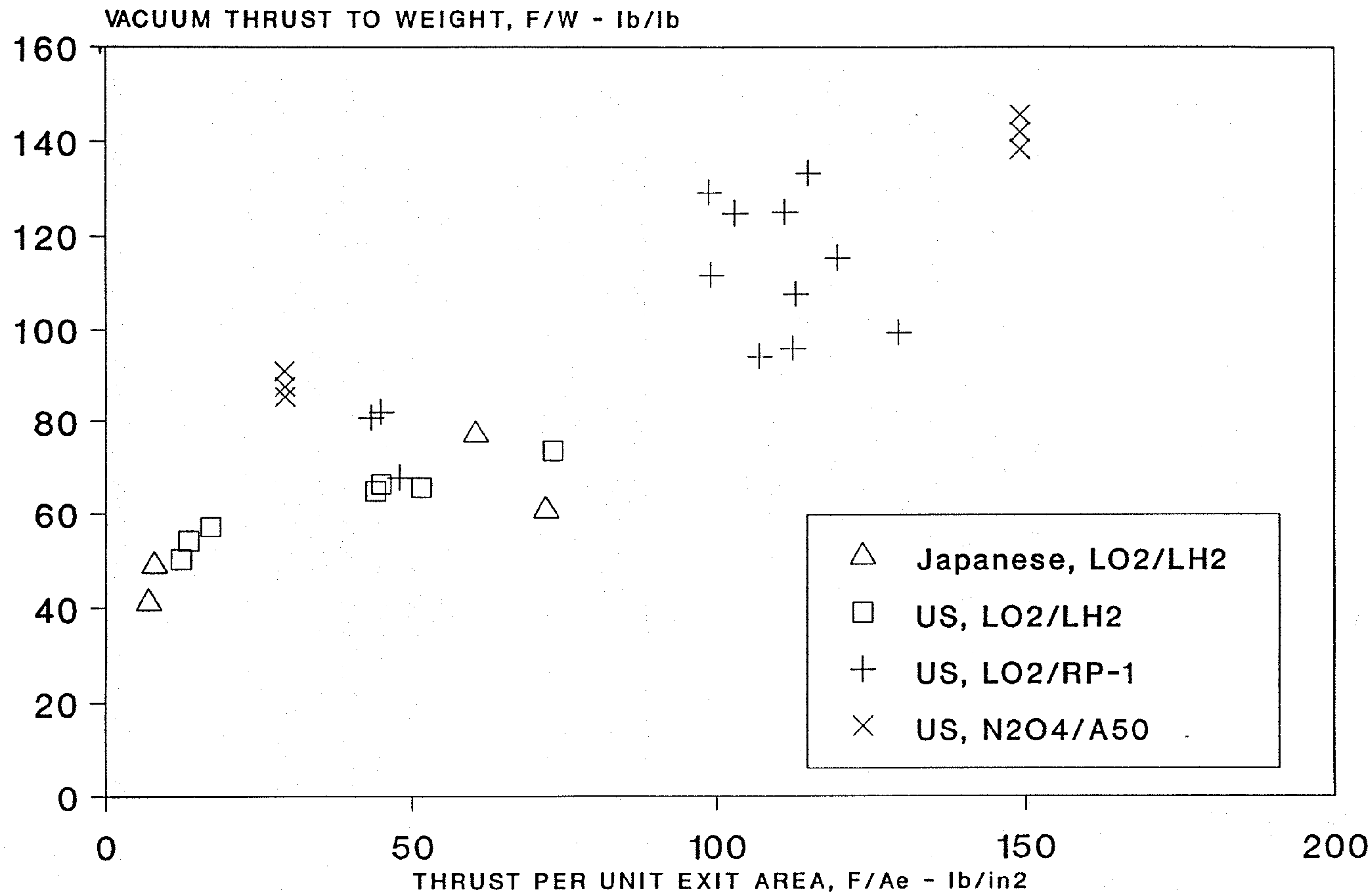

Figure 2.11 Engine Thrust to Weight Comparison

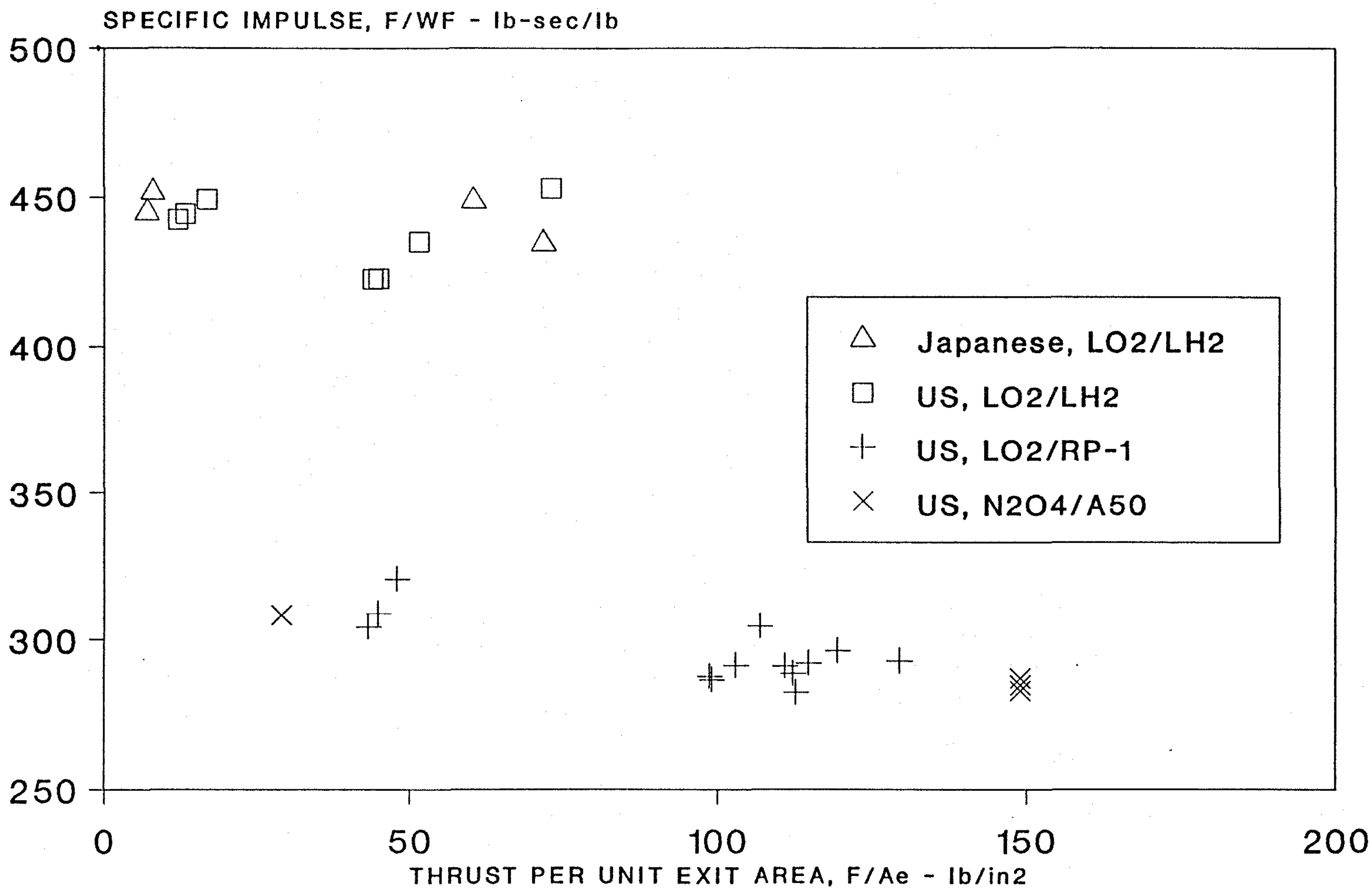

Figure 2.12 Engine Specific Impulse Comparison

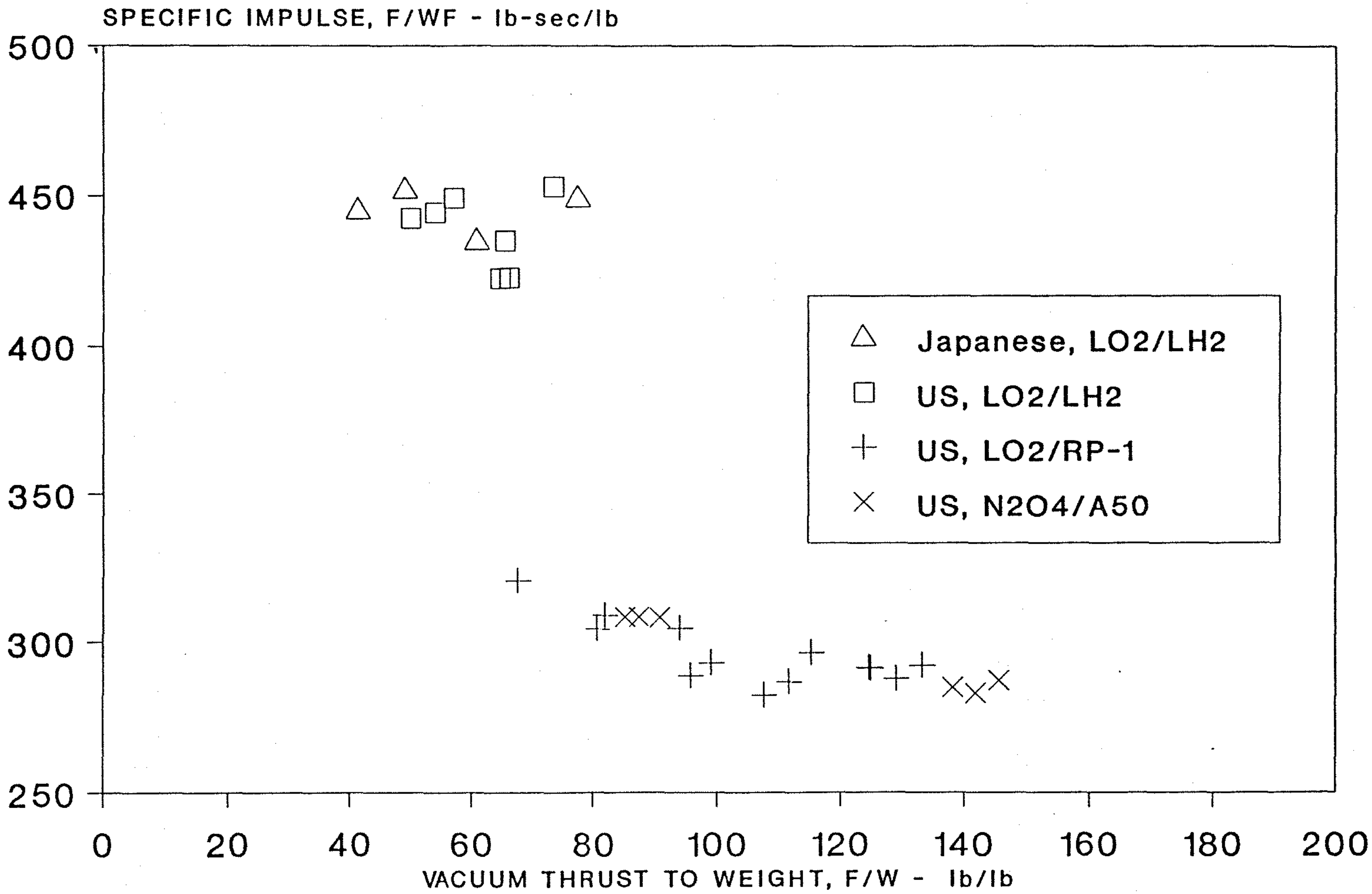

Figure 2.13 Rocket Engine Performance

chamber and preburner combustion studies, performance investigations, and hot gas and coolant side heat transfer studies for the fuels listed in Table 2.11. Discussions also indicated that experimental combustion and stability investigations for up to 20% aluminum powder in RP-1, and main combustor cooling studies using liquid oxygen as the coolant, were in progress.

Table 2.11
Liquid Rocket Booster Engine Options

Thrust Class	Propellant	Power Cycle
220,000 LB	LO_2/KEROSINE	gas generator
220,000 LB	LO_2/CH_4	gas generator
220,000 LB	LO_2/CH_4	staged combustion
550,000 LB	LO_2/LH_2	gas generator
550,000 LB	LO_2/CH_4	gas generator

Similarly, a comprehensive program in rocket engine activities to support spaceplane research is being pursued. The HIPEX engine already described combines design studies, component development, and breadboard systems tests to demonstrate an expander cycle main combustion pressure potential some three times higher than that achieved in the United States to date. *Aviation Week* (Ref. 2.9) has reported on plans by Mitsubishi Heavy Industries to test an experimental liquified air cycle engine (LACE) (Fig. 2.14). Tests of a LACE heat exchanger to support these plans were performed last year. The engine will use the same combustion hardware as the LE-5 engine. While not optimized for performance, it will provide proof of concept and significant operating information. The depth of interest in LACE is perhaps reflected by Yamanaka's comments that "LACE is very simple, it can rely on rocket technology, which Japan has been developing. Our experience with scramjets is very little, almost nothing." (Ref. 2.38.)

The literature also shows an interest and activity on the part of Japan relative to transportation engines beyond H-II uprating and spaceplanes. Studies of propulsion concepts for Mars, fusion rockets, and laser thrusters show a conceptual and experimental activity looking beyond the indicated major focus areas.

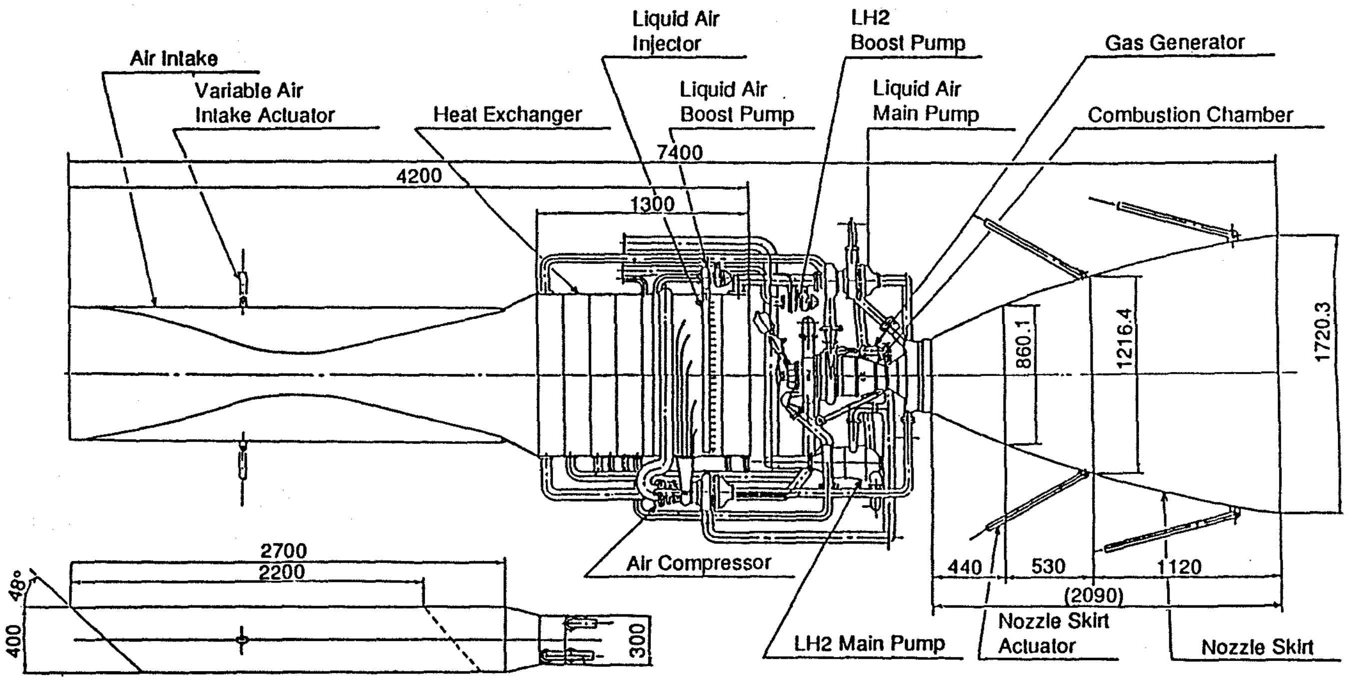

Figure 2.14 Experimental LACE Engine Based upon LE-5

FACILITIES

Tanegashima Space Center

Japan's facilities for launching liquid propellant rockets, N-I, N-II, H-I, and H-II, are located at the southeastern tip of Tanegashima Island. This location is

30.4 degrees north latitude, which is almost the same as KSC at 28.5 degrees. Its facilities include the Takesaki Range, the Osaki Range, and the Yoshinobu Range, as well as radar, tracking, and data acquisition stations. The center occupies approximately 2,100 acres and has launched 106 vehicles as of September 1988. (Note that KSC occupies approximately 140,000 acres and is adjacent to Cape Canaveral Air Force Station with some 17,000 acres.) The Takesaki Range includes a launch control room, a functional test room, an assembly building and a launch position for small rockets, as well as the H-II SRB static firing test facilities.

The Osaki Range (Fig. 2.15) is used to launch the H-I vehicle and includes the necessary operational facilities for the stack and launch process used for the H-I. These include separate stage assembly buildings, a solid motor test building, a spacecraft test and assembly building, a third stage inspection and spacecraft integration building, the mobile service tower for vehicle stacking, the blockhouse, and the Range Control Center, and static firing test facilities for the first stage engine and its major components.

The Yoshinobu Range (Fig. 2.15) currently under construction for the H-II rocket is located approximately 3,300 ft. east of the H-I launch complex. The major launch facilities are the Vehicle Assembly Building, Mobile Launcher, Pad Service Tower, Propellant Storage Facilities, and the Launch Control Building. These facilities are based upon the integrate, transfer, launch process. The LE-7 static test stand and blockhouse are also located in this range. This test stand is only 1800 feet from the launch pad and 550 feet from the Vehicle Assembly Building. The crawler way connecting the Vehicle Assembly Building and the launch pad is 1,750 feet long in comparison to the distances at KSC--VAB to Pad A, 18,160 feet; VAB to Pad B, 22,440 feet.

Tanegashima Space Center does not have port facilities. The rocket components and spacecrafts for launching and test are unloaded at Shemama Port on the other side of Tanegashima Island, and transported overland. By agreement with the local fishermen's union, all launches at Tanegashima must take place within one of two 6-week periods in August/September or January/February. The close confines of the space center, the restricted port availability, and the constrained

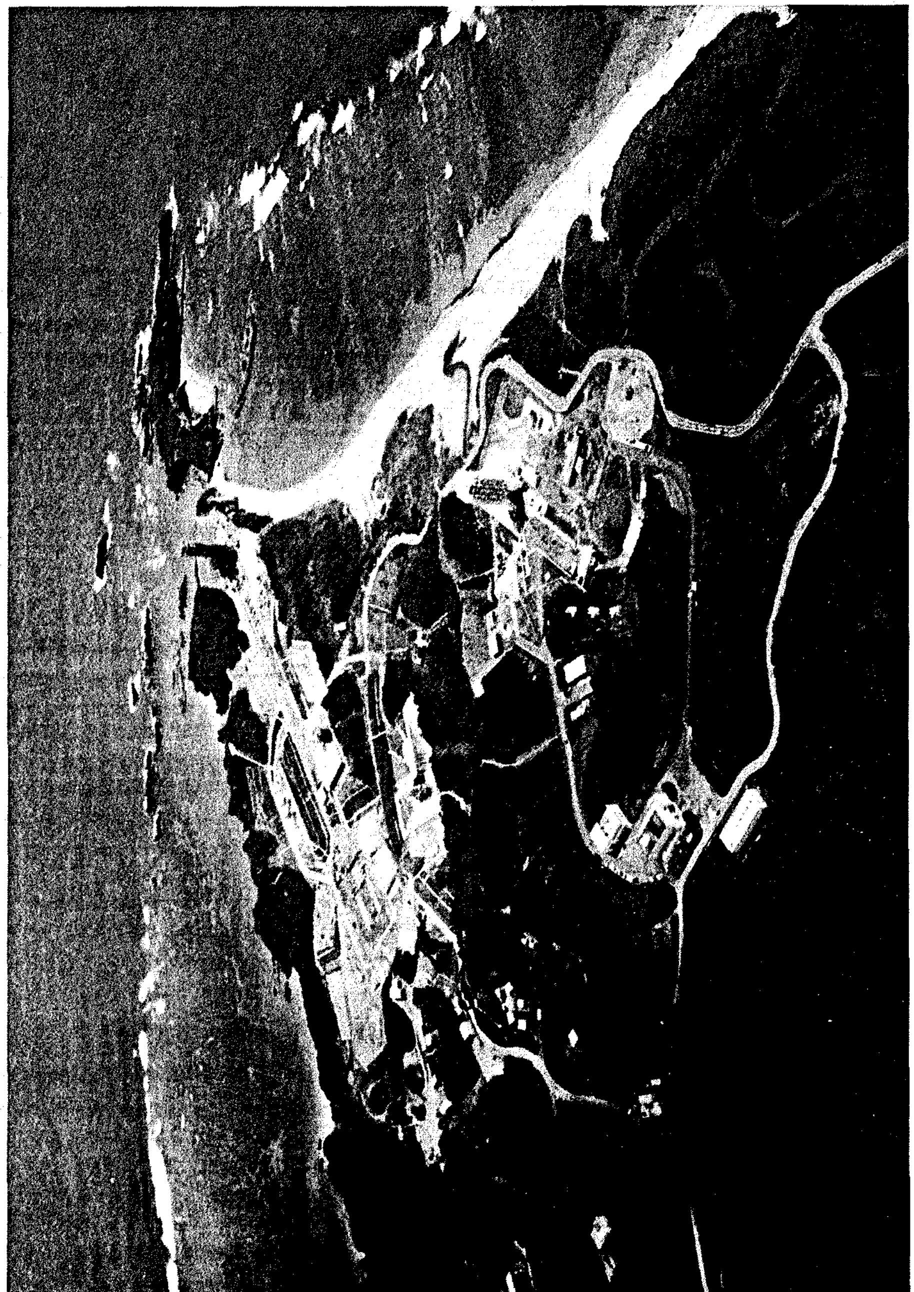

Figure 2.15 Tanegashima Space Center

launch periods would appear to severely limit the launch opportunities of the H-I or H-II vehicles. These limitations appear to be in conflict with the marketing indications unless actions are undertaken to relieve the constraints.

Kakuda Propulsion Centers

The Kakuda Propulsion Center of NASDA was founded in July 1980 as a test center for rocket propulsion systems. It is located on 200 acres in the northern end of Kakuda City. Major facilities at the center include propellant feed system test stands for the LE-5 liquid oxygen turbopumps and the LE-7 liquid hydrogen turbopumps; a simulated high-altitude test stand for cryogenic rocket engines (thrusts to 33,000 lb.); a tank thermal evaluation test stand; and the support equipment for conducting test preparations, test operations, data acquisition, and analyses.

Propulsion test facilities of the National Aerospace Laboratory (NAL) are also located at Kakuda, across the street from those of NASDA. This research area was established in 1965 and appears to be of about the same size as the NASDA center. The major facilities include both sea level and altitude rocket test stands (thrusts to 22,500 lb.); a turbopump test facility for the LE-7 oxygen pump; combustion test facilities for gas generators, preburners, and main combustors; turbopump research facilities; a ramjet, scramjet, and air-breathing rocket test facility; and a solid rocket motor test facility.

ROCKET PROJECT IMPLEMENTATION

NASDA, the National Space Development Agency of Japan, is a quasi-governmental organization established to carry out the development and implementation of Japan's major space applications programs. In carrying out these programs, NASDA employs phased project planning similar to that used by NASA or USAF. Relative to new programs, NASDA accomplishes the conceptual studies and preliminary designs for candidate systems. Advanced research with respect to key technologies is carried out by government research bodies like NAL and ISAS, as well as at NASDA and private industry, under the approval of the Space Activities Commission. The SAC is chaired by the Minister of State for Science and Technology, a member of the Prime Minister's Office. As such he is the highest ranking decision maker for space activities in Japan.

In implementing a new program, a proposal selected as the result of the conceptual studies is made to the SAC after coordination with the appropriate agencies like NAL and ISAS. Normally for a large project like the H-II, a special subcommittee of the SAC is organized. This subcommittee reviews the technical aspects of the proposal. The SAC makes the decision to proceed or not based upon the report of the subcommittee, the budget situation of the government, and other considerations. After the program is approved, the Minister of STA directs NASDA to execute the project and provides the necessary budget authority. Other organizations like NAL may also execute some portion of the project with their own budgets. Typical examples of the related NAL activity are the development of a prototype LE-7 oxygen turbopump and the 22,000 lb. thrust experimental engine to support the LE-5 development.

The selection of contractors to participate in the projects is based upon the technical capability of the companies gained through previous projects. The project share of the various companies for launch vehicles is almost fixed by this historical technical capability. Participation by the contractors in the advanced design activities is accomplished through funded studies from NASDA and independent research activities. The primary focus appears to be to design to cost in cooperation with the government agencies so that the project will get a favorable start decision by the SAC.

Program/project organizations are established within NASDA to accomplish a specific project once approved. For example, the Launch Vehicle Program Office of the Program Management and Control Department manages the H-II program, including launch site, payload, and launch operations. Management includes budget allocations, performance requirements, schedule coordination, and other matters. These are documented in program plans, project plans, system/subsystem engineering specifications, and interface handbooks. These documents are prepared, approved, issued, and controlled under systems engineering and configuration control procedures managed by the Program Planning and Management Department.

Safety, reliability, and quality assurance requirements and guidelines come from the Safety Control Department and the Reliability Assurance Department. These requirements and guidelines appear to be based primarily on the systems and procedures developed and used by NASA and USAF as documented in their various specifications and handbooks, such as NHB 5300.4 and MIL-STD-499. Each project/subproject establishes its own plans by tailoring the guidelines or standards in cooperation with the contractors.

OBSERVATIONS

The proceeding examination of liquid propellant rocket engine and propulsion systems in Japan has focused on the evolution from the early designs based upon technology licensed from U.S. manufacturers, to the design for the H-II rocket. This evolution has produced systems and performance comparable to current U.S. rockets. It is apparent that this evolution has made extensive use of U.S. data, experience, procedures, and technology available in the open literature and technical meetings. It is also evident that this evolution is characterized by a series of carefully planned steps using low-risk, well-characterized options that build upon prior rocket elements. The rocket engines and propulsion systems that make up the evolutionary elements are typically of conservative design since extensive use is made of experimental engines and stages in phased projects. This allows necessary adjustment of designs as the experimental results warrant.

It appears that significant constraints on the current Japanese launch capability exist as a result of the limited area available for test and launching and the limited periods that are available for launches to occur. These constraints will have to be relaxed before the Japanese launch capability will achieve competitive strength in the world. However, there are many locations in the world where commercial interests and agreements could act to relax these constraints.

The examination also found a broad range of options being actively investigated for future evolutionary improvement to the current rockets and launch capability. The earth-to-orbit transportation options being pursued appear to focus on two paths: low cost unmanned cargo transportation, and manned transportation. Japan views continued development of improved systems as fundamental to a sound space infrastructure and indispensable to Japan's autonomous space activities.

These research and advanced development activities are evident in the published technical reports and the activities in progress as observed in Japan. Various configurations are being investigated to accomplish evolutionary improvement of the H-II to double or triple its payload capability. Options include increased numbers of solid motors and several configurations of liquid rocket boosters. Another aspect of these studies is focused on spaceplane research to provide a manned transportation capability. Several paths, including high-performance rocket engines and air-breathing/rocket engine combinations, are evident in the subscale experimental tests, component developments, and experimental rocket engines being prepared for and already in test.

Figures 2.16 and 2.17 provide a picture of the pace of development and introduction of new levels of capability. Figure 2.16 shows the payload-to-liftoff weight ratio of several U.S. expendable launch vehicles and the Japanese rockets examined here as a function of the year of first launch. It shows that with the introduction of new Japanese rocket technology in the H-II vehicle, payload efficiency will surpass that of the most efficient U.S. expendable rocket. Figure 2.17 tracks payload to orbit capability for the same U.S. and Japanese rockets, also as a function of the year of the first launch. It shows the accelerating pace of introduction of Japanese rocket technology, and that with the H-II improvement plans projected for the late 1990s, the H-II will equal or surpass the payload to orbit capability of the largest U.S. expendable, unmanned cargo carriers.

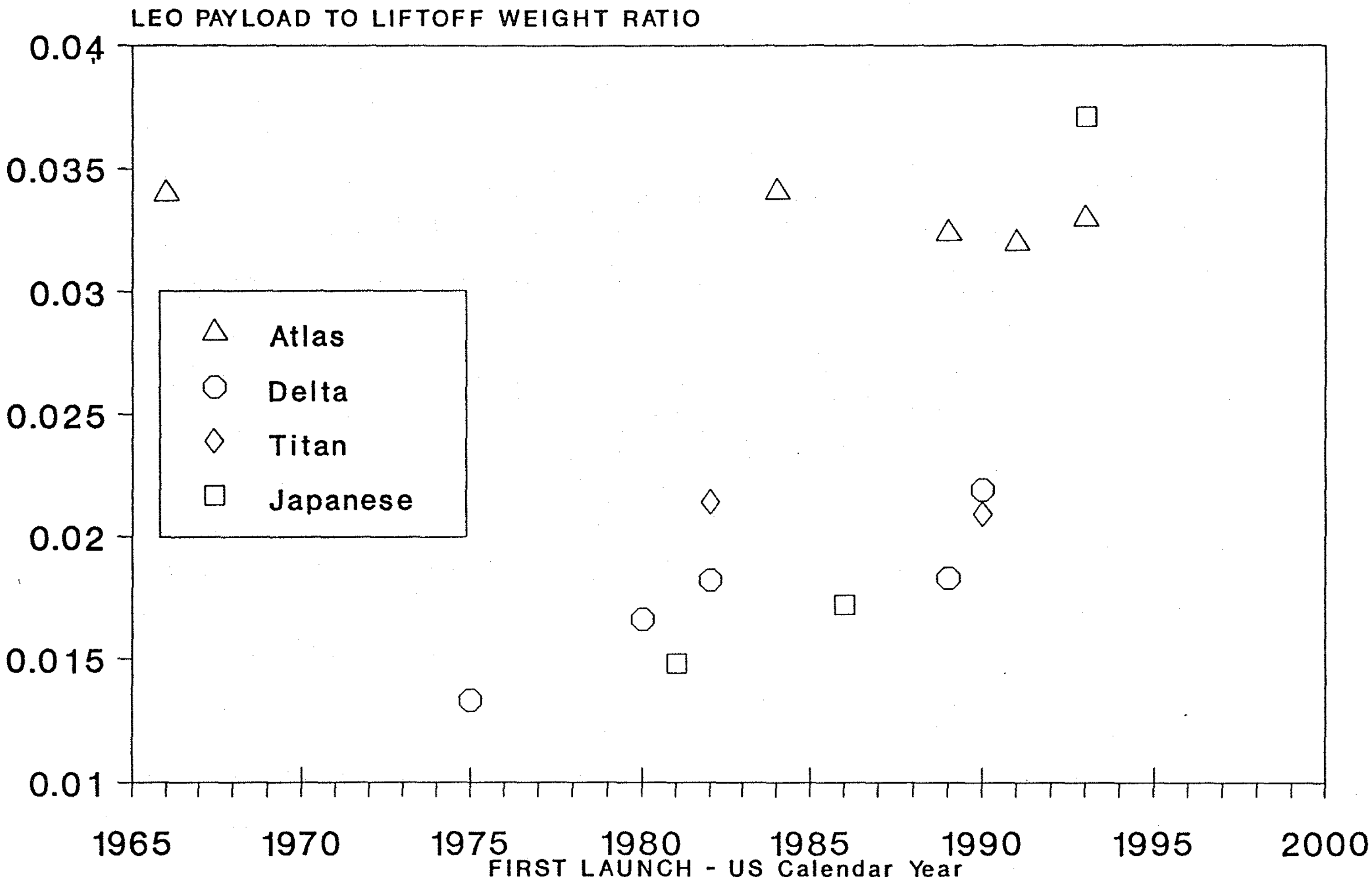

Figure 2.16 Vehicle Payload Efficiency Improvements

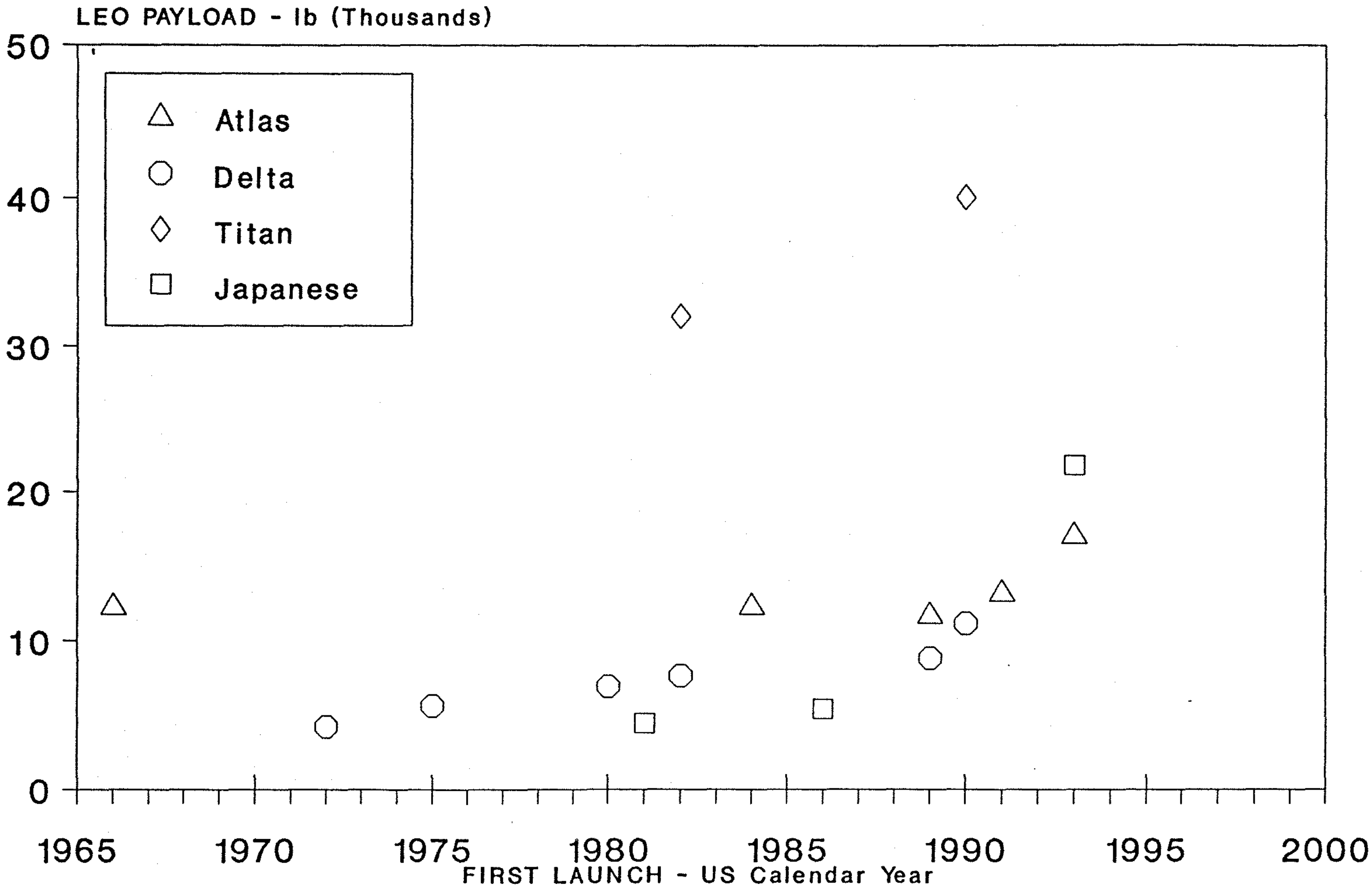

Figure 2.17 Introduction of Payload Capacity

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CHAPTER 3

TURBOMACHINERY SYSTEMS

James R. Brown

INTRODUCTION

The previous chapter has given an overall systems view of Japanese space launch vehicles and their corresponding propulsive engines. In this chapter, we present a similar overview of the turbomachinery systems used for their liquid rocket propellant engines. This systems view of the turbomachinery is followed by a synopsis of the characteristics of the individual turbomachinery components in Chapter 4.

Research efforts on turbopumps for liquid rocket engines were begun in Japan in the early 1970's by NAL and IHI. The actual development of the turbopump system for the second stage engine (LE-5) of the H-I launch vehicle was then started in 1977 using the technology base obtained from this previous research. In addition, a parallel turbopump development for a 15,000 lb. class engine for the MU launch vehicle was carried out at ISAS beginning in 1975 and terminating in the early 1980's.

The LE-5 turbomachinery is developed and produced by Ishikawajima-Harima Heavy Industries (IHI) with the strong support of NAL. It is then integrated into the engine system by Mitsubishi Heavy Industries (MHI). The LE-5 development program and the subsequent production activities were managed by the National Space Development Agency (NASDA) with participation by NAL and ISAS. In parallel with the LE-5 development program, various component technology efforts (e.g., bearings, seals, inducers) to support the design data base were conducted in Japan, primarily by NAL.

The H-II vehicle program was initiated in 1985. It was managed by NASDA with MHI and IHI in the same roles for the LE-7 first stage engine as for the LE-5. At the same time an upgrade program for the LE-5 engine was initiated to modify the engine from a gas generator to an expander-bleed cycle design for the H-II second stage. One of the most significant challenges for the Japanese turbomachinery community is the development of the LE-7 hydrogen turbopump. The turbopump is shown in Figure 3.1 and is discussed later in this chapter.

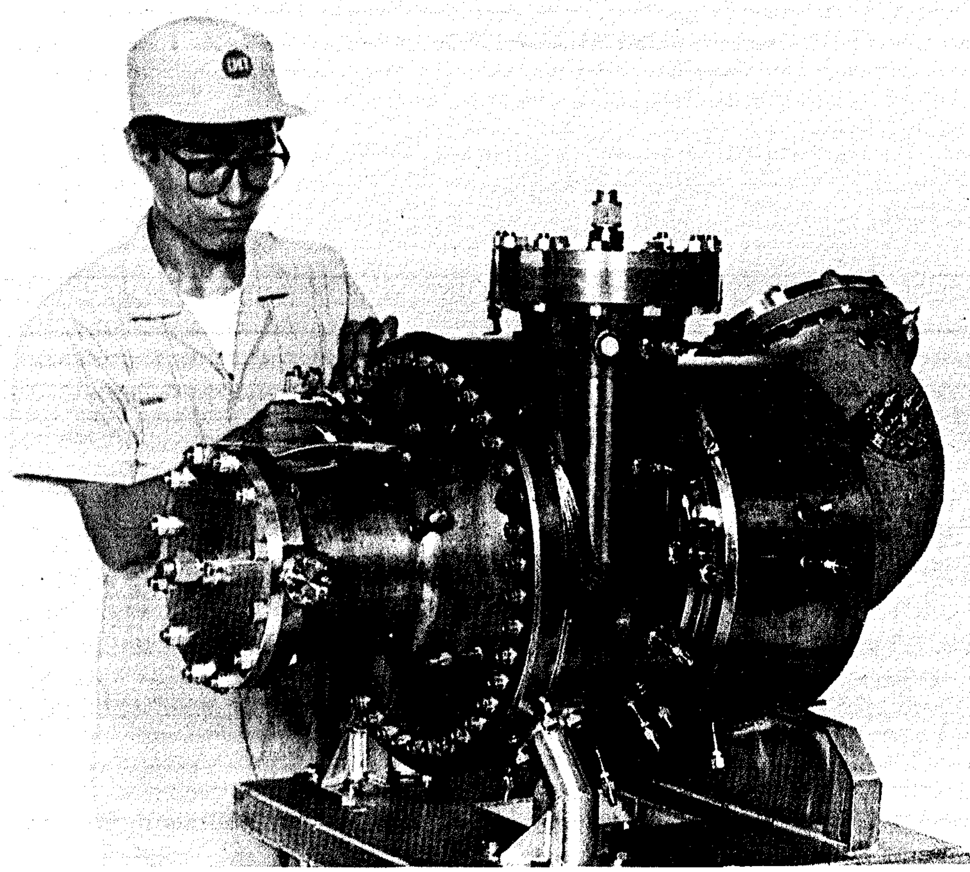

Figure 3.1. LE-7 Hydrogen Turbopump Assembly (Photo Courtesy IHI)

In the United States from 1975 to 1985 a number of turbopump technology programs were conducted on upper stage sized engines and the Space Shuttle Main Engine (SSME) was brought to operational status. While it is not easy to quantify when a component technology is first available, the time an engine using that technology is first used may be a rough guide. Table 3.1 compares the first use of LH_2 engines in the United States and Japan.

Table 3.1
First LOX/LH_2 Engine Launch

	U.S.	Japan
Second Stage	1963	1986
First Stage	1981	1993 (projected)

Table 3.1 should not be used to imply that the difference between the U.S. and Japanese technology level is greater than ten years but rather to show that new launch vehicle programs are not often initiated in either country. To meet possible future needs, component technology programs will continue to advance the state-of-the-art in many areas without a commitment to specific applications. In Japan, this technology advancement is primarily done by NAL and the industrial firms with assistance in some areas by ISAS and the national university system.

One significant difference between the U.S. and Japanese programs is that of competition. In the United States, there are three major rocket engine contractors who actively compete for significant programs. In Japan, however, while there is some competition at the conceptual design level, an engine development program will quite likely have IHI as the turbomachinery contractor and MHI as the overall engine system developer.

DESIGN APPROACH

The design requirements of turbomachinery in Japan, as in any other high technology society, are set by the end product's use. In the United States, the need for man-rated reliability and reusability in addition to high performance drives the design effort related to the SSME turbopumps. In the Expendable Launch Vehicle (ELV) area, the design approach for new or modified designs (based on the U.S. ballistic missile legacy) emphasizes low cost and limited life. In Japan, the turbomachinery requirements have been to build a successful ELV capability with first the LE-5 (H-I second stage engines) and currently with the LE-7 (H-II first stage engines): the challenge was to produce a design using only indigenous capabilities other than information available in the open literature of the United States and Europe.

The engine design requirements of the LE-5 and LE-7 coupled with the design approach philosophy of the Japanese participants have produced a somewhat different result than in the United States For example, in the LE-7 engine the controller is primarily a sequencer rather than one with significant closed-loop capability similar to that of the SSME. The result of this approach is that the turbopumps must be carefully checked out as components to determine their individual performance characteristics prior to being integrated into the engine. In the case of the SSME, turbopumps can be assembled into the engine without prior component performance testing.

Another example of a design approach difference is that the LE-7 fuel pump was selected as a two-stage design because that was within the Japanese state-of-the-art, whereas a technology program would be necessary to confidently select a three-stage design. This decision, combined with some detail constraints (e.g., impeller tip speed, bearing DN, turbine temperature), limited the LE-7 to approximately 2,000 psia chamber pressure rather than the 3,000 psia level of the SSME. Such approach differences do not make the turbomachinery designed in Japan better or worse but only different; however, those differences do impact the engine design characteristics.

LE-7 Turbopumps

In 1985, the Japanese made a commitment to their most challenging space project to date, that of the indigenous development of a launch vehicle in the same class as the U.S. Titan 34D and the European Space Agency Ariane 4. For this vehicle's first stage engine, a staged combustion cycle to achieve high performance in a relatively small size was chosen. (The only other engine in the free world to use this cycle is the Space Shuttle Main Engine.) Component technology programs conducted by NAL, the LE-5 engine experience, and the available data on the SSME, allowed the selection of this cycle by NASDA. However, due to the lack of first-hand experience with high-pressure three-stage hydrogen pumps (as used in the SSME), it was decided to select a two-stage hydrogen pump (Fig. 3.2) with its lower performance and reduced risk.

Table 3.2 compares the major characteristics of the LE-7's hydrogen pump with that of the SSME. Allowing for the difference in the number of stages (and the inherent added difficulty of producing a three-stage turbopump), the technology levels of the two turbopumps are similar. One other design difference between these turbopumps is the use of a helical inducer on the LE-7's main pump shaft to provide the Net Positive Suction Head (NPSH) capability, while in the case of the SSME, pump NPSH requirements are met by the use of a separately driven boost pump.

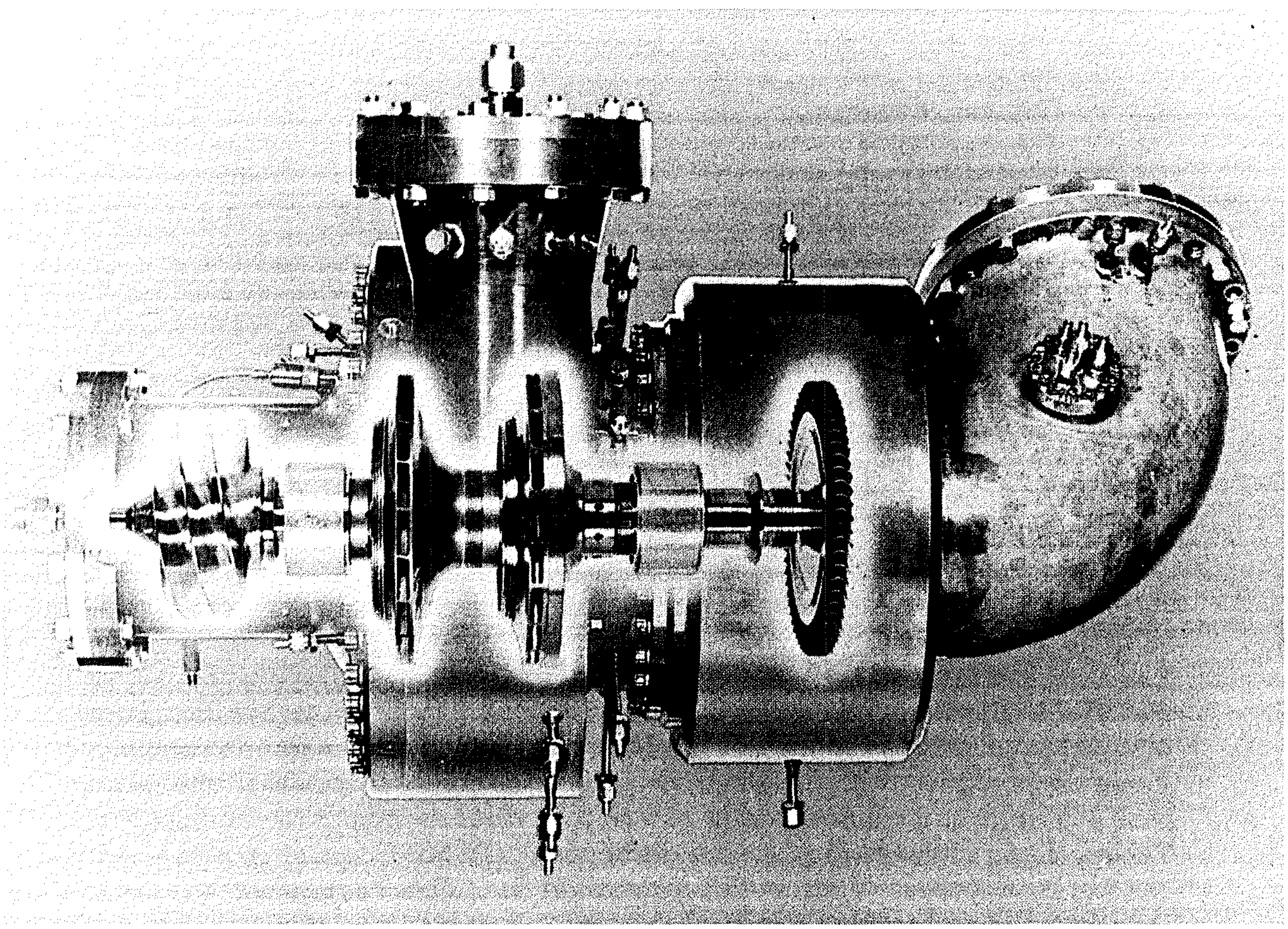

Figure 3.2. LE-7 Hydrogen Pump (Photo Courtesy IHI)

Table 3.2
Comparison of LE-7 and SSME Hydrogen Turbopump Characteristics

	LE-7	SSME
Pump		
Number of stages	2	3
Speed, RPM	46,130	34,164
Required NPSH, ft	445	N/A
Pump flow, lb/sec	87.4	149.2
Pressure rise, P	4700	5800
Impeller tip speed, ft/sec	1970	1790
Efficiency,%	70.9	77.4
Turbine		
Number of stages	1	2
Inlet pressure, P	3520	4922
Inlet temperature, R	1770	1781
Pressure ratio	1.43	1.45
Efficiency,%	72	82

The LE-7's oxidizer pump (Fig. 3.3) is also similar in technology level to that of the SSME. Both turbopumps have a single main impeller (the SSME utilizes a double suction inlet design while the LE-7 configuration has a more conventional single suction inlet) with the single stage preburner pump mounted on the same shaft. The major characteristics of the two oxidizer pumps are compared in Table 3.3. As in the case of the fuel turbopump, the SSME's suction requirements are met by a separately driven boost pump, while the LE-7's requirements are met by a helical inducer on the main shaft.

During the development of the LE-7 some problems have been encountered, such as approximately 3% low fuel pump head performance and some early fuel pump impeller shroud and turbine blade cracking. None of these problems would be expected to pose long-term barriers to the accomplishment of the LE-7 program objectives.

LE-5 and LE-5A Turbopumps

The 25,000 lb. thrust class LE-5 engine powers the second stage of the H-1 launch vehicle. To date, this engine has performed satisfactorily on all of its five H-1 launches. The LE-5A derivative engine is now in its qualification phase which is expected to be concluded in 1990, well before it is operationally needed for the H-II vehicle.

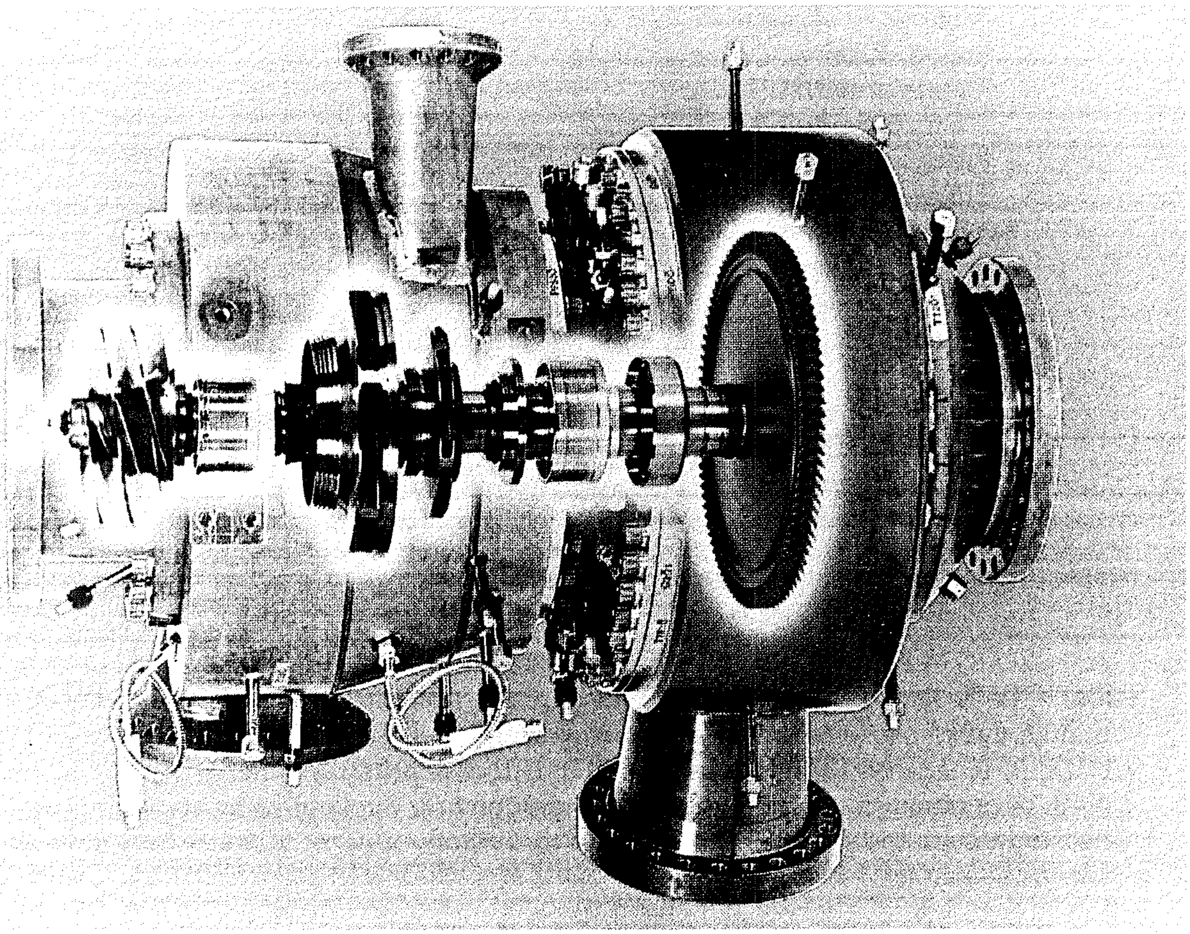

Figure 3.3. LE-7 Oxidizer Pump (Photo Courtesy IHI)

Table 3.3
Comparison of LE-7 and SSME Oxygen Turbopump Characteristics

	LE-7	SSME
Main Pump		
Number of stages	1	1
Speed, RPM	20,000	27,210
Required NPSH, ft	98.5	N/A
Pump flow, lb/sec	505	1070
Pressure rise, P	3030	3730
Efficiency,%	75	67
Preburner Pump		
Pump flow, lb/sec	96.4	100
Pressure rise, P	1650	3030
Efficiency,%	65	80
Turbine		
Number of stages	1	2
Inlet pressure, P	3410	4985
Inlet temperature, R	1750	1455
Pressure ratio	1.43	1.51
Efficiency,%	48	79

Since 1975, hydrogen and oxygen turbopump systems in the 15,000 to 25,000 lb. pound thrust class have been under evaluation in Japan. This activity involved NASDA, NAL, and ISAS as well as MHI, IHI, and possibly others. By 1981, when NASDA committed to the H-I vehicle development, applicable turbopump systems had demonstrated full power, closed-loop operation. One of these turbopumps is shown in Figure 3.4.

The principal characteristics of the LE-5 hydrogen turbopump compared with those of the current RL10 model are presented in Table 3.4. Allowing for the differences due to the respective engine cycles (LE-5 gas generator versus RL10 expander), such as the LE-5's supersonic, high-temperature, low-flow rate turbine compared to the RL10's low-pressure ratio, cool high-flow turbine, the two turbopumps' technology levels are quite similar. The LE-5's hydrogen pump is shown in Figure 3.5.

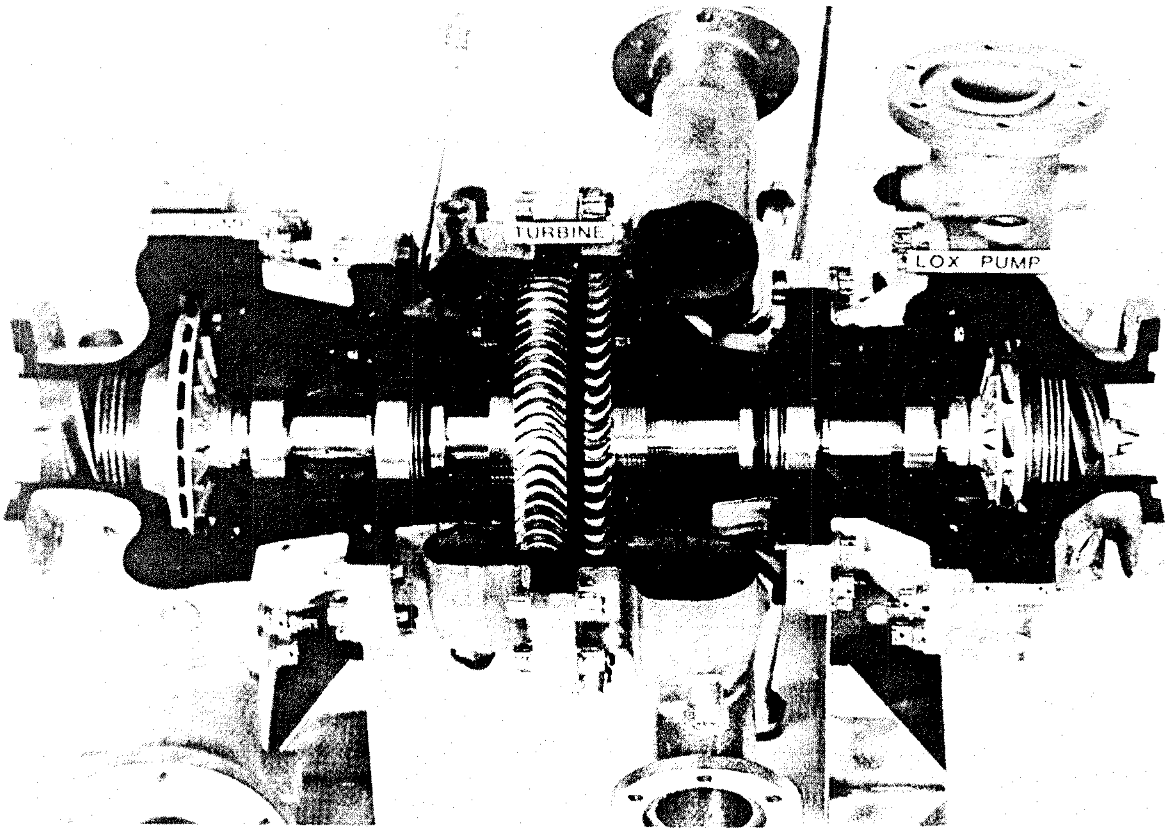

Figure 3.4. Early $H_2/0_2$ Engine Turbopump for 7 to 10 Ton Class Engines
(Photo Courtesy IHI)

Table 3.4
Comparison of LE-5 and RL10-3-3A Hydrogen Turbopump Characteristics

	LE-5	RL10-3-3A
Pump		
Number of stages	1	2
Speed, RPM	50,000	32,800
Required NPSH, ft	184	100
Discharge pressure, P	823	1120
Pump flow, lb/sec	7.76	6.2
Efficiency,%	59	58
Impeller tip speed, ft/sec	1250	1010
Turbine		
Turbine flow, lb/sec	0.93	6.1
Inlet pressure, P	353	845
Inlet temperature, R	1516	380
Pressure ratio	4.8	1.5
Efficiency,%	48	72
Number of stages	2	2
Materials		
Pump impeller	Titanium alloy	AL(AMS 4135)
Pump shaft	INCO 718	SS(AMS 5667)
Housing	A-356	AL(AMS 4215)
Turbine rotor	INCO 718	AL(AMS 4127)
Bearing (Ball)	440C	440C
Seal ring	carbon	carbon

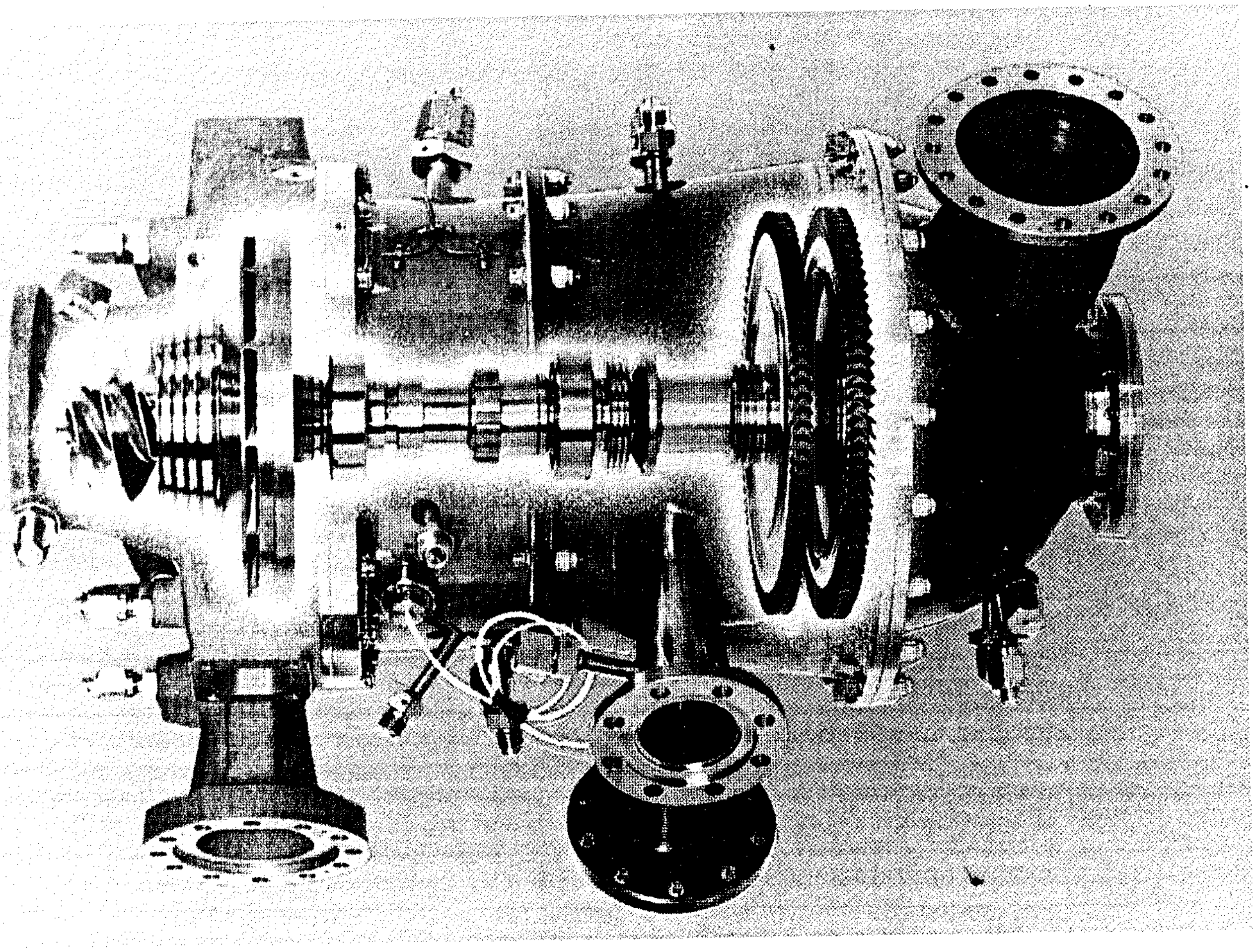

Figure 3.5. LE-5 Hydrogen Pump (Photo Courtesy IHI)

The principal characteristics of the LE-5 and RL10 oxidizer turbopumps are presented in Table 3.5. The oxidizer pumps again are comparable. The LE-5's oxidizer turbine is similar to its fuel turbine, while the RL10 oxidizer pump is driven by its fuel turbine via a gear train. This difference does not imply differences in technology but rather different design approaches. The LE-5 oxidizer pump (prototype model) is shown in Figure 3.6.

Specific data was not found on the LE-5A turbomachinery; however, References to configuration changes imply that turbomachinery modifications were not significant.

Table 3.5
Comparison of LE-5 and RL10-3-3A Oxygen Turbopump Characteristics

	LE-5	RL10-3-3A
Pump		
Number of stages	1	1
Speed, RPM	16,500	13,100
Required NPSH, ft	25	11
Discharge pressure, P	742	670
Pump flow, lb/sec	42.7	30.8
Efficiency,%	66	63
Impeller tip speed, ft/sec	315	250
Turbine		
Turbine flow, lb/sec	0.86	N/A*
Inlet pressure, P	72	N/A*
Inlet temperature, R	1250	N/A*
Pressure ratio	1.87	N/A*
Efficiency,%	39	N/A*
Number of stages	2	N/A*
Materials		
Pump impeller	A-606	SS(AMS 5646)
Pump shaft	INCO 718	SS(AMS 5667)
Pump housing	AC4C	AL(4130)
Turbine rotor	INCO 718	N/A*
Bearing (Ball)	440C	440C
Seal ring	carbon	carbon

* The RL10 oxidizer pump is driven via a gear train by the fuel turbine.

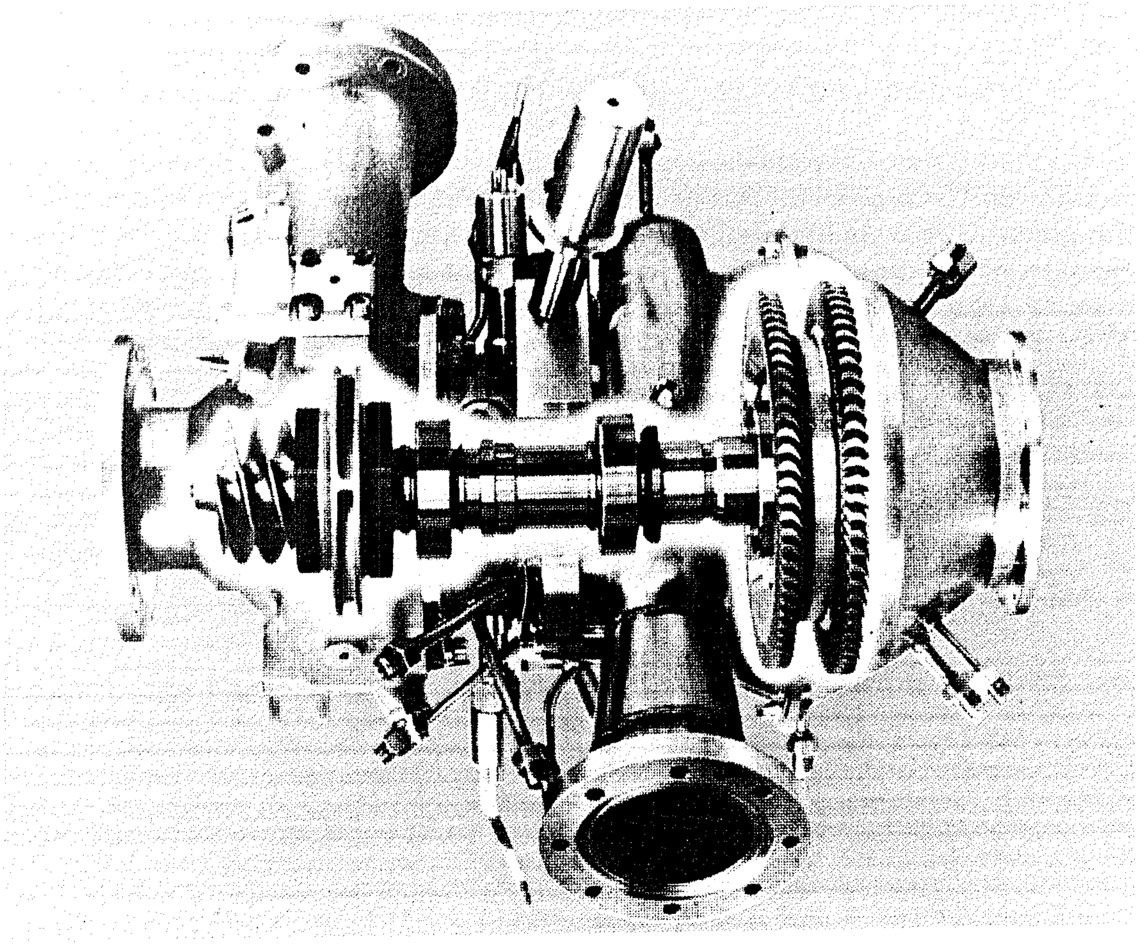

Figure 3.6. LE-8 Oxidizer Pump (Photo Courtesy NAL)

Small Advanced Turbopumps

Two demonstrator engine programs were conducted during the 1980s that might be considered for future full-scale development. Neither of these programs is currently active in Japan but both have verified the concepts sufficiently to allow a development commitment, should the need arise. The first of these programs is a "1-Ton" (2200 lb thrust) engine which has a potential application to a small space transfer vehicle. Versions of this engine were individually fabricated by IHI and MHI. Figure 3.7 shows the IHI engine.

The IHI version was tested at IHI's Aioi test center, and the MHI version (called the RE-6) was tested at the Tashiro Field Laboratory. The IHI engine was modified and retested in 1987. For this latest test series, the turbopumps were substantially modified to improve their performances. The first version was configured with the oxidizer and fuel pumps mounted on a single shaft with the drive turbine in the center of the shaft. This version has fewer parts but does not provide as efficient a system as one with individually optimized turbopumps.

In the United States, the smallest comparable turbopump is currently in a prototype development test phase by Aerojet TechSystems under contract to the Astronautics Laboratory. This engine, designated the XLR-134, has a design thrust level of 500 pounds. The hydrogen pump currently under test is a two-spool design with three stages per spool (for a total of six stages). Only one spool has been tested so far. The IHI latest hydrogen pump characteristics and the XLR-134 first spool results are presented in Table 3.6.

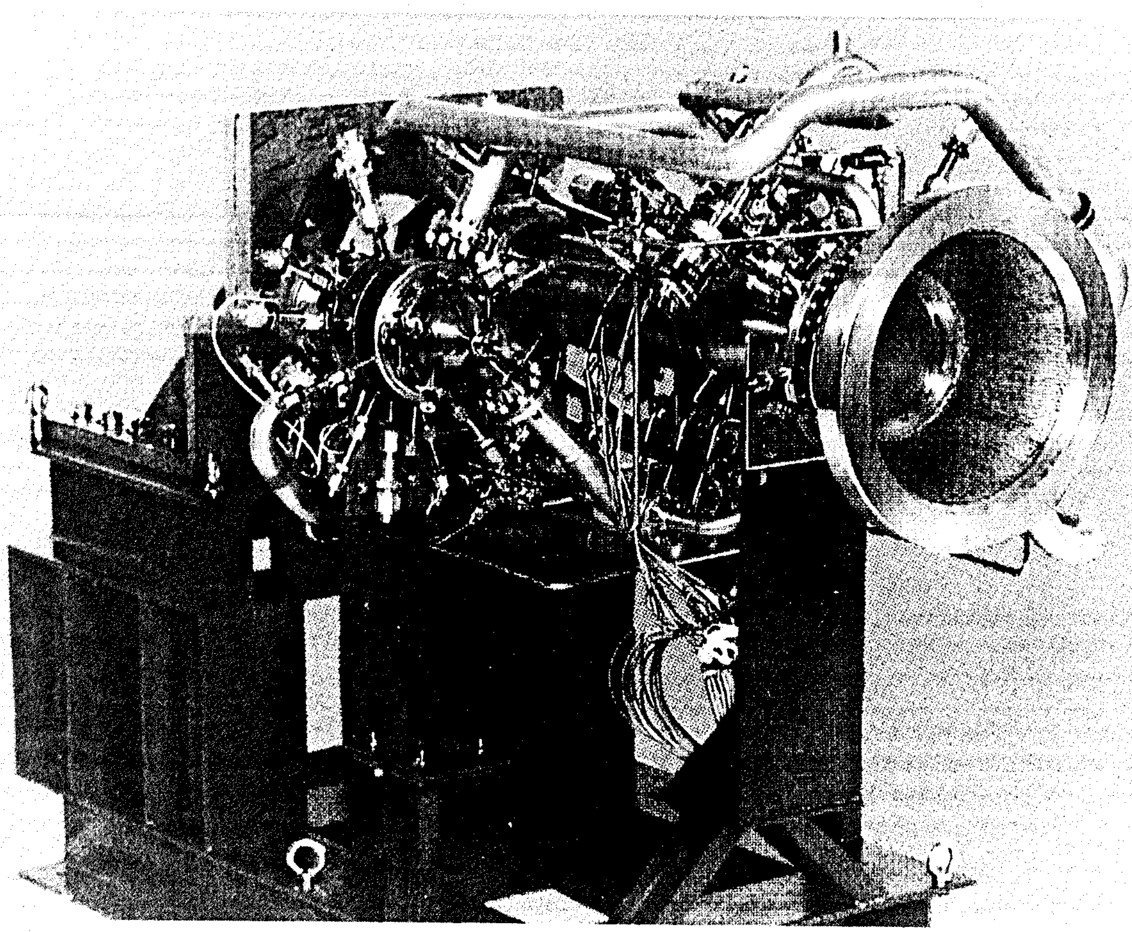

Figure 3.7. "1-Ton" Demonstrator Engine (Photo Courtesy IHI)

Table 3.6
Comparison of Small Space Transfer Engine Turbomachinery

	"1 Ton"	XLR-134*
Hydrogen pump		
Number of stages	2	3+
Mass flow rate, lb/sec	0.9	0.15
Speed, RPM	84,500	60,000
Discharge pressure,. PSIA	1000	420
Efficiency,%	40	N/A
	"1 Ton"	"Small OTV"#
Oxygen pump		
Number of stages	1	2
Mass flow rate, lb/sec	4.1	5.5
Speed, RPM	43,200	70,000
Discharge pressure, PSIA	760	4100
Efficiency,%	45	59

* Aerojet TechSystems under contract to the Astronautics Lab.

\+ Dual spool, six stage pump; data shown is for the one spool tested to date.

\# Aerojet TechSystems under contract to NASA-LeRC.

The oxygen pump of the XLR-134 has not yet been tested; however, under a NASA-Lewis Research Center (LeRC) contract, Aerojet TechSystems has tested an oxygen pump designed for a 3750 pound thrust space transfer engine. The characteristic of the IHI "1-Ton" oxidizer pump and the "small space transfer engine" pump are also compared in Table 3.6.

The second Japanese demonstrator engine in the High Pressure Expander (HIPEX) 30,000 pound thrust engine under a program conducted by ISAS and IHI. The complete engine system was assembled and tested at Noshiro in 1987. Design studies show a three-stage hydrogen pump with a two-stage turbine is needed to meet the requirements of 1500 PSIA chamber pressure. Since the demonstrator engine employed a single stage hydrogen pump, the high-pressure levels were not expected in this test phase. In the United States, comparable component design performance has been demonstrated by Rocketdyne with the RS-44 turbopump under NASA-LeRC contract. The characteristics of the HIPEX demonstrator engine and the RS-44 pumps are shown in Table 3.7. Note the demonstrators engine featured here has reduced capabilities as compared to the design cycle.

Table 3.7
Comparison of Medium Space Transfer Engine Turbomachinery

	HIPEX*	RS-44+
Hydrogen pump		
Number of stages	1	3
Mass flow rate, lb/sec	1.6	6.2
Speed, RPM	44,000	95,000
Pump discharge pressure, PSIA	700	4560
Oxygen pump		
Number of stages	1	1
Mass flow rate, lb/sec	10.2	37.3
Speed, RPM	12,000	70,000
Pump discharge pressure, PSIA	300	4320

* Engine System demonstrator.

\+ Individual component demonstrations.

This data on the turbopumps designed for small- and medium-sized advanced engines (500 lb. to 30,000 lb. thrust) indicates that while Japanese demonstrated technology levels are not up to the U.S. state-of-the-art, turbopump systems could be developed to meet engine needs in this thrust range.

TURBOPUMP TEST FACILITIES

Because of the limited land available for rocket system test activities, the facilities used in Japan are widely scattered across the country (Fig. 3.8). Using the LE-7 oxidizer turbopump as an example, the detail components are fabricated in IHI's Tanashi plant outside Tokyo. These components are then transferred to the IHI Mizuho plant (Fig. 3.9) for clean-room final assembly.

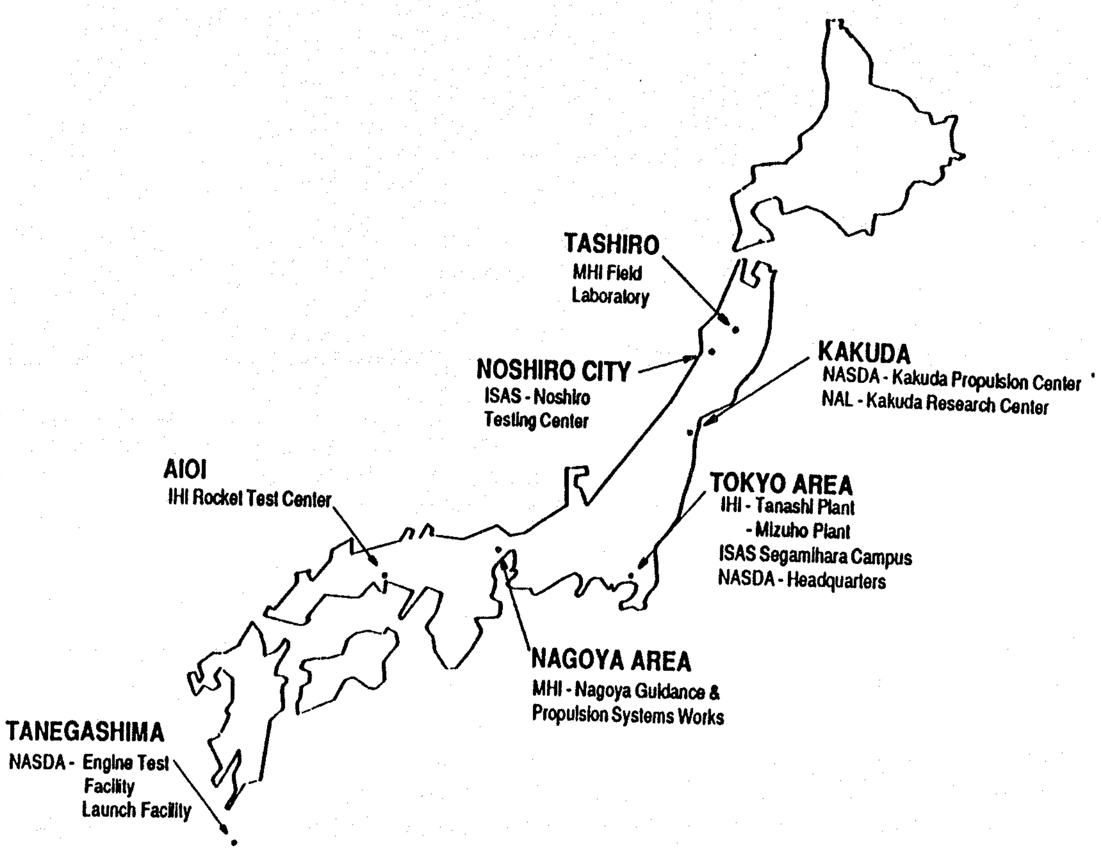

Figure 3.8. Centers of Liquid Rocket Activity in Japan

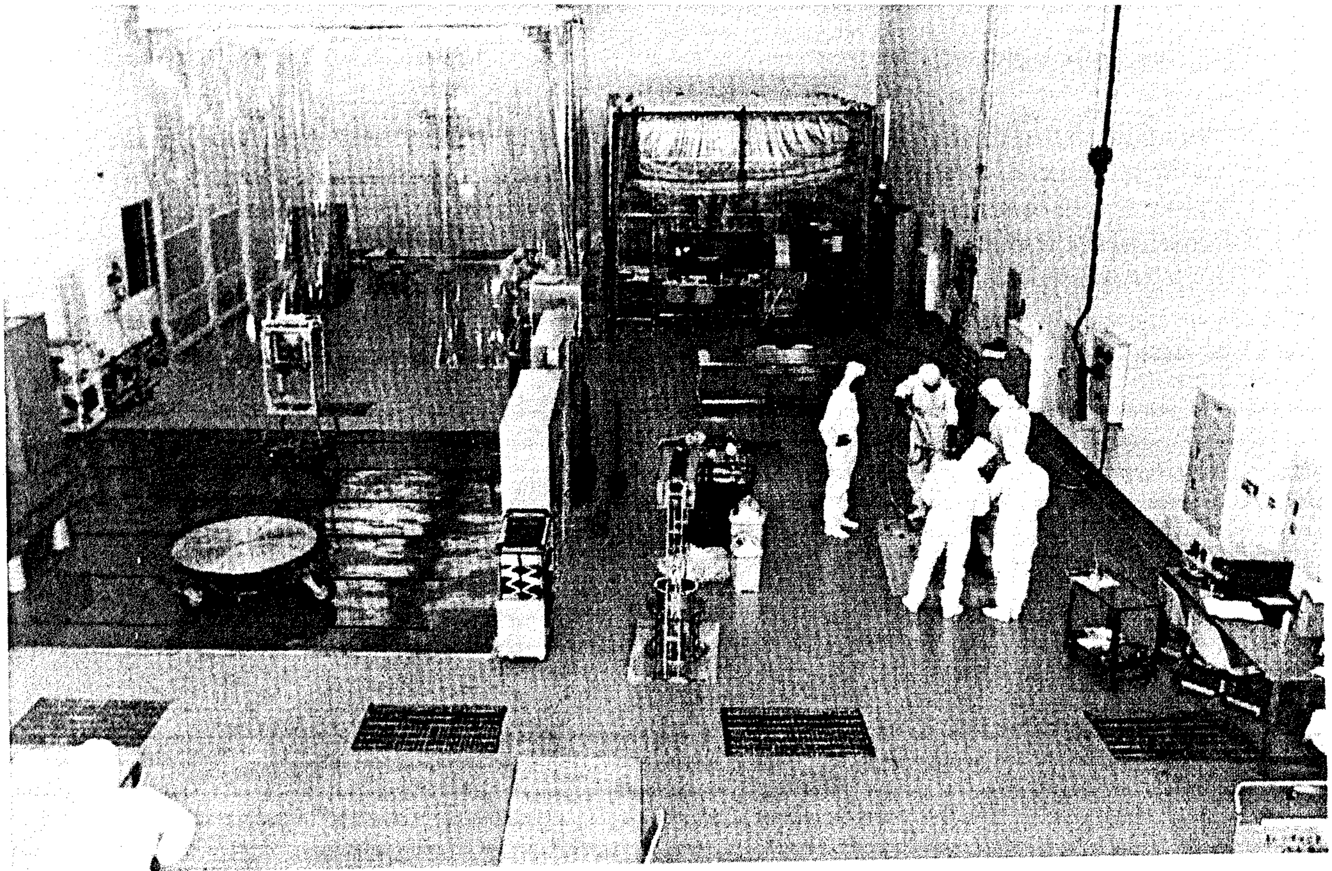

Figure 3.9. IHI's Clean Room Facility at the Mizuho Plant
(Photo Courtesy IHI)

The assembly is then shipped north to the NAL's Kakuda Research Center for testing. This testing is required to determine individual turbopump characteristics that will be used to adjust the engine start transient sequence. After component testing at Kakuda is completed, the turbopump is shipped south to Nagoya to be integrated by MHI into the LE-7 engine. Following engine assembly, the hardware is shipped either to Tashiro in the north or Tanegashima in the south, depending on the engine test program objectives. The flow process is illustrated in Figure 3.10.

The fuel turbopump process flow is similar except that after assembly at the IHI Mizuho plant, the turbopump is first sent to IHI's Aioi test facility for a partial check-out (to approximately 50% pressure), then to the NASDA Kakuda Propulsion Center prior to being sent to MHI Nagoya for engine assembly. The Aioi test facility is shown in Figure 3.11, and the NASDA hydrogen pump test facility is shown in Figure 3.12.

The capabilities of the NAL and NASDA turbomachinery test facilities were specifically expanded considering the needs of the LE-7 turbopumps, and appear to be fully capable of supporting the program's requirements. Both test facilities are capable of full power testing of the oxidizer and fuel turbopump (at the NAL and NASDA facility respectively) for substantial durations.

In general, component and engine test facilities in the United States are located near the contractor's assembly facility. In Japan, however, the test facilities, whether belonging to the government agencies or to the contractors, are not located near the MHI or IHI assembly facilities. This difference, while probably not significant in the long term, does seem as if it would slow down the typical developmental test/teardown/analysis/rebuild/retest process.

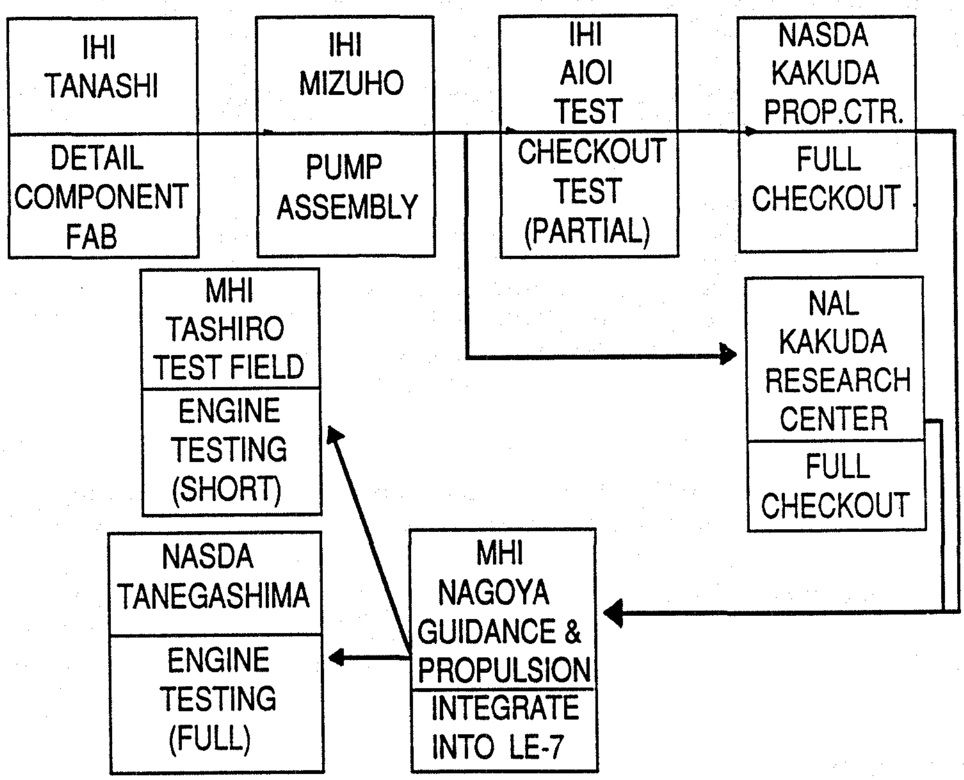

Figure 3.10. LE-7 Turbopump Assembly Flow

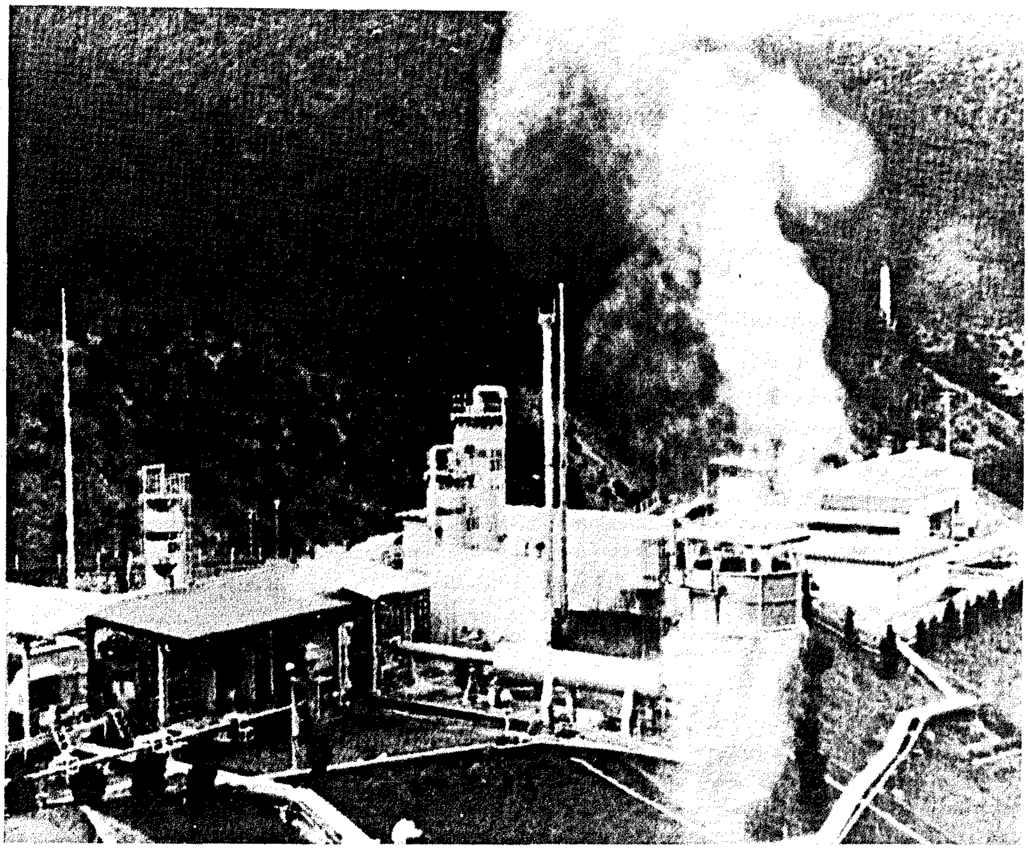

Figure 3.11. LE-7 Hydrogen Pump Test at IHI's Aioi Test Facility
(Photo Courtesy of IHI)

One other difference between the two countries' programs is that currently there is no full-scale turbopump test facility (SSME-sized) in the United States. Early in 1990 Pratt & Whitney, under a NASA-Marshall Space Flight Center contract, will bring such a facility on line adjacent to its turbopump assembly facility for the Alternate Turbopump Development Program. NASA is also considering building a similar facility at the Stennis Space Center in Mississippi.

POSSIBLE FUTURE TRENDS

According to the available literature and from discussions with knowledgeable individuals in Japan, the next evolutionary steps in rocket technology may come in three areas. The first is a growth version of the H-II vehicle. Options for this

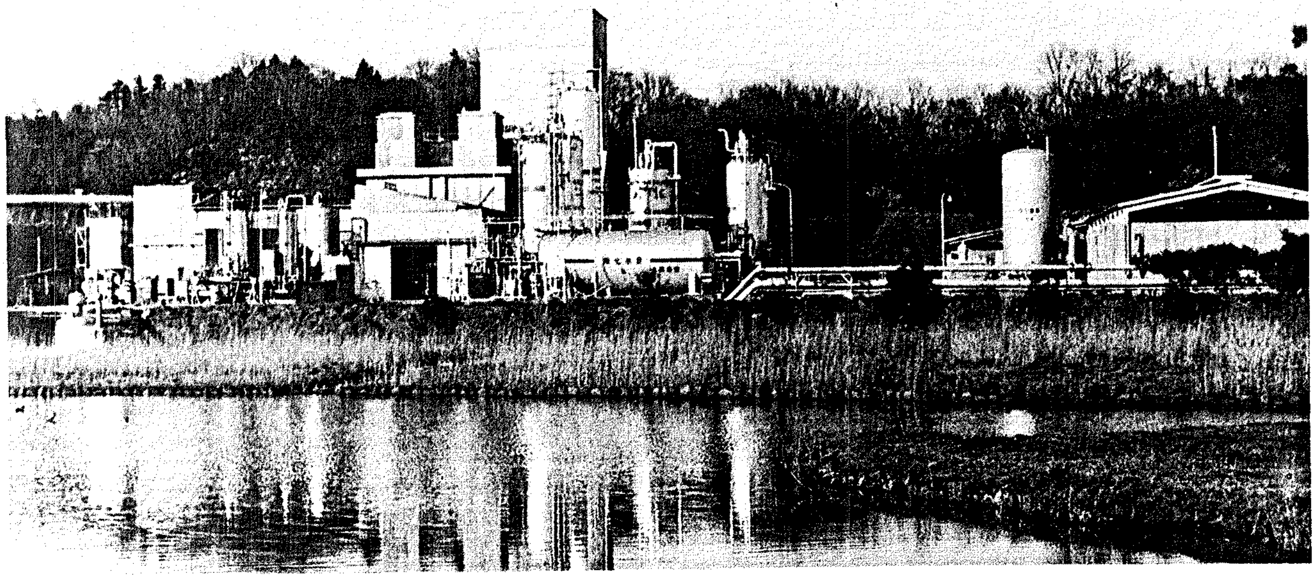

Figure 3.12 LE-7 Hydrogen Pump Test Facility at NASDA-Kakuda Propulsion Center
(Photo Courtesy of IHI)

growth may include the use of either Liquid Air Cycle Engines (LACE) or LOX/hydrocarbon engines, replacing the solid motor strap-on boosters of the current H-II vehicles. In either of these options the development of appropriate turbomachinery systems will play a significant role. The technology capability, while based on the LE-5/LE-7 turbopumps, has not as yet been demonstrated.

A second evolutionary area concerns the launching of man-rated systems. It is a stated objective of the Japanese space policy to "promote fundamental and advanced research and development work to establish technologies for manned space transportation systems on a long-term basis" (Ref. 3.12). A possible implication of this objective for technology is to emphasize longer life and better diagnostic measurements during operation of turbomachinery systems. Such an effort would likely also start from the current LE-5/LE-7 data base.

The third possible area of technology evolution involves space operations. Current Japanese space policy calls (Ref. 3.12) for "basic and advanced research development work" on both an Orbital Servicing Vehicle and an Orbital Transfer Vehicle (OTV). While it is possible that either vehicle could be configured without turbopump-fed engines, it is likely that the OTV will be configured with pumps for performance and packaging reasons. It may be expected that such a technology program will start from the designs and hardware base of the "1-Ton" and HIPEX engines, and will emphasize long life, reusability, and operational flexibility. All of these programs at the technology level will likely involve the IHI and MHI corporations, as well as NAL and perhaps ISAS.

In the United States, turbomachinery technology thrusts are similar to those mentioned above. The need for improvement in launch vehicle systems has spurred the ALS component technology program activities. The overall emphasis on reduced cost has resulted in technology programs focusing on Lox/hydrocarbon engines, system diagnostics, health monitoring, and improved durability. In the orbit transfer arena, the technologies emphasized are long-term autonomy and reusability in addition to high performance. Technology programs are currently in place in these areas.

One area that may produce programs not yet addressed or committed to in either country's space policy is that of the "Space Exploration Initiative" being considered in the United States. Relevant to turbomachinery, the requirements of such a program imply turbopumps that are capable of operating over a wide throttle range (perhaps 20:1) and are of rugged (durable) design with onboard condition monitoring systems.

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CHAPTER 4

TURBOMACHINERY COMPONENTS

M. L. Stangeland

In this Chapter, we discuss the performance characteristics and the current state of the art of rocket engine turbomachinery in Japan. We begin by discussing pumps and turbines, then move to bearings, seals, and rotordynamics, followed by an overview of materials and flow diagnostics. Our scope includes turbomachinery components for the engine systems discussed in the previous chapters, as well as turbomachinery components which were developed as part of technology programs culminating in these engine systems.

PUMPS

Overall Pump Performance

Japanese pump experience. Since 1975 the Japanese have designed, built, and tested six LOX pumps and seven LH_2 pumps in the development of liquid rocket engine turbopumps for the LE-5, LE-5A, LE-7, and HIPEX engine systems. In addition, one LOX pump and two LH_2 pumps are currently being designed for engine systems under development. Japanese experience also includes testing two of the pumps designed for LOX and LH_2 in LN_2 to investigate two-phase pumping capability. Tables 4.1 and 4.2 summarize the overall characteristics of the pumps used for the development of engine system turbopumps; the investigation into two-phase pumping capability will be discussed later. The data were obtained from References 4.1 to 4.9.

DESIGNATION	FLOW (GPM)	DISCH. PRES. (PSI)	SPEED (RPM)	HEAD (FT)	# OF STAGES	OPERATING NPSH	OPERATING NSS	IMP. D(tip) OUT	EFFICIENCY (TEST)	STAGE Ns
LE5	792	823	50,000	26,108	IND+1	183.7	29,831	5.75	60	685
LE7	8,916	4679	46,130	153,563	IND+2	446.2	44,866		66	944
SHSC D	832	783	50,000	26,100	IND+1	183.7	28,903	5.75	59	702
SHSC D'	832	783	50,000	26,100	IND+1	183.7	28,903	5.28	62	702
SHSC E	797	783	50,000	26,100	IND+1	183.7	28,289	5.50	61	687
SHSC F	650	856	45,000	28,473	IND+2	111.5	33,436	4.80	70	880
SHSC H	696	3756	80,000	123,382	IND+2	183.7	42,297	5.10	59	539
HIPEX	1,073	3655	83,200	118,826	IND+3				64	971
ATREX	62	285		9,720						

1 PREDICTED

Table 4.1
Japanese LH_2 Pump Experience

DESIGNATION	FLOW (GPM)	DISCH. PRES. (PSI)	SPEED (RPM)	HEAD (FT)	# OF STAGES	OPERATING NPSH	OPERATING NSS	IMP. D(tip) OUT	EFFICIENCY (TEST)	STAGE Ns
LE5	274	771	16,500	1,535	IND+1	24.6	26,431	4.38	65	1114
LE7	3,238	3089	20,000	6,236	IND+1	98.4	37,333	7.72	80	1622
LE7 (PREB.)	619	4743	20,000	3,401	1			5.65	73.5	1117
SHSC A	285	1697	20,000	3,520	IND+1	42.7	20,213	5.28	67	739
SHSC B	285	1697	35,000	3,520	IND+1	42.7	35,373	3.27	66	1293
SHSC C	273	696	16,500	1,462	IND+1	24.6	24,680	4.38	66	1153
SHSC G	254	3539	45,000	7,308	IND+1	65.6	31,108	3.40	63	907
HIPEX	405	2480	27,000	5,041	IND+1				66	908

1 PREDICTED

Table 4.2
Japanese LOX Pump Experience

The Japanese design experience is for LOX/Hydrogen fed systems which use separate turbopumps for each propellant because of the density difference. All the LOX pumps are single stage pumps and the LH_2 pumps vary from one to three stages, depending on the discharge pressure requirements. All pumps have inducers in front of the first stage impeller to satisfy the suction performance requirements without the use of boost pumps. The Japanese have used shrouded impellers exclusively to eliminate the large tip clearance losses associated with unshrouded impellers in combination with a mixture of vaned and vaneless diffuser designs. Axial thrust control on all turbopump designs has been achieved using single acting balance pistons on the final stage impeller. The impeller shroud leakage flow has been controlled by stepped seals with smooth lands instead of the more conventional labyrinth configurations. Plastic wear ring inserts have been used to achieve less than 0.002 inch clearances for some of the smaller pump designs.

The most mature of the designs are the pumps for the LE-5 engine. Initial development of these pumps started in 1975, with LN_2 testing and first turbopump test in 1980. The only other pumps which have been built for a specific engine system are the pumps for the LE-7 engine. Similar to the Space Shuttle Main Engine (SSME), the LOX pump in this system has two pumping elements - a main pump and a preburner pump which delivers a portion of the total engine LOX flow at the high pressure required by the preburner. Development of these pumps started in 1984 with LN_2 testing and the first turbopump test in 1986. Currently being designed are pumps for the high-pressure expander cycle engine (HIPEX) and an LH_2 pump for an air-breathing system (ATREX).

The other pumps reported in References 4.1 and 4.2 were built and tested as part of a structured technology development program to develop small high-speed centrifugal pumps (SHSC) to support designs for the LE-5, LE-7 and HIPEX engine systems. Nine pumps have been tested in this program (SHSC A,B,C,D,D',E,F,G,H), with designs SHSC C,D,D', and E made specifically for the development of the LE-5 engine.

Comparison with U.S. experience. Figures 4.1 and 4.2 show a comparison of the Head and Flow requirements of the LH_2 and LOX pumps designed in Japan and the United States (log-log scales are used in these figures). It can be seen that the early Japanese LH_2 pump designs (SHSC D,D',E,F,LE-5) are very similar in requirements to the RL-10 LH_2 pump; however, unlike the RL-10 pump, all except the SHSC F are single stage designs. Figures 4.1 and 4.2 show that in terms of flow and overall head, the Japanese designs lie within U.S. experience.

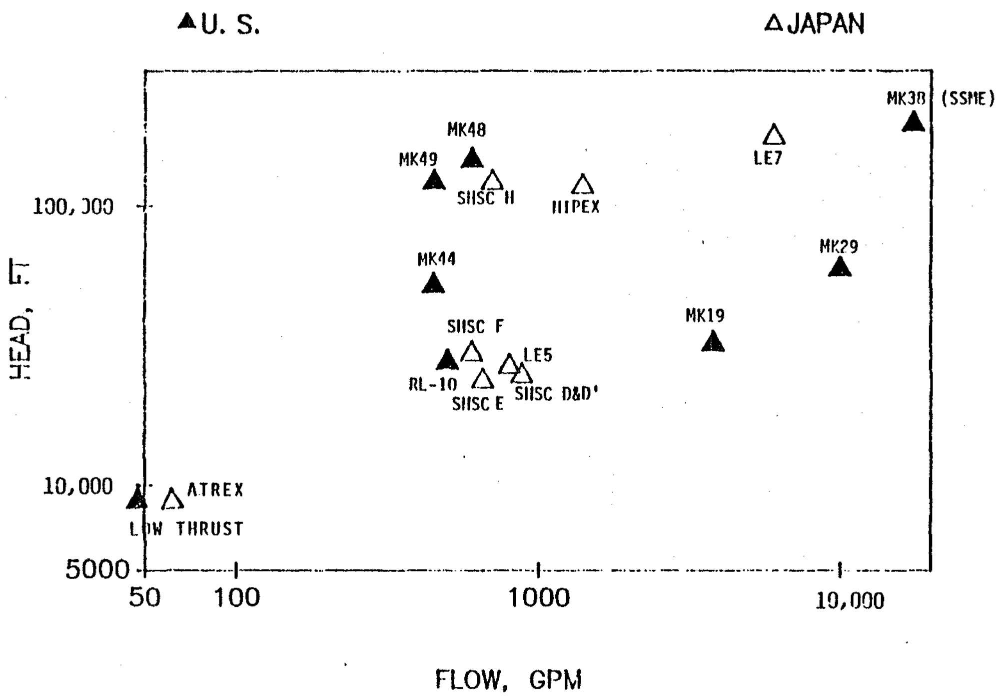

Figure 4.1. U.S. & Japanese LH_2 Centrifugal Pumps Head vs. Flow

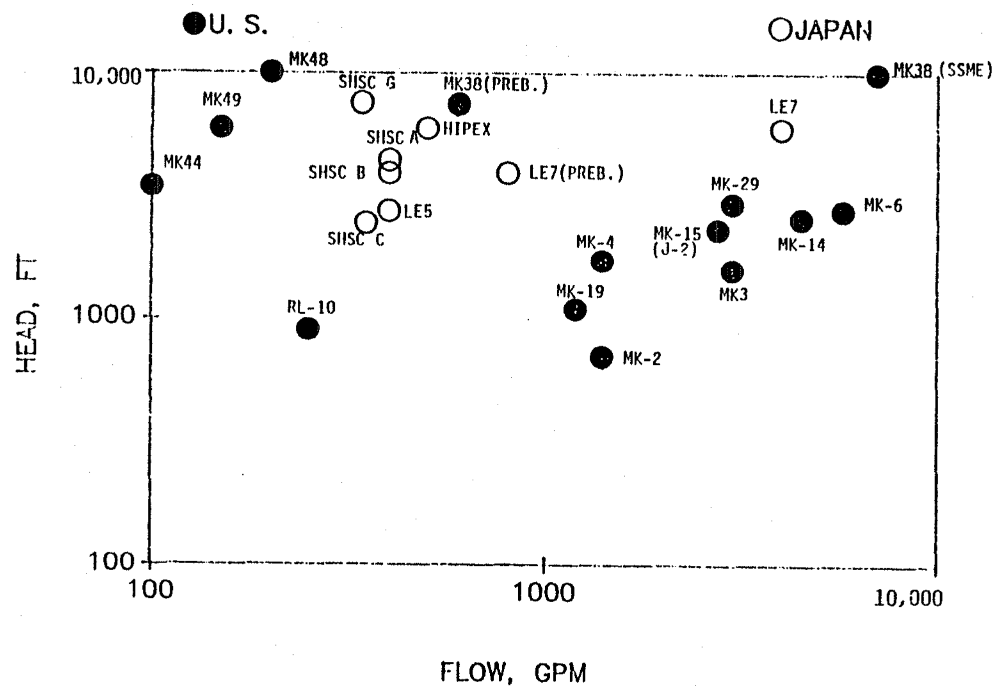

Figure 4.2. U.S. & Japanese LOX Centrifugal Pumps Head vs. Flow

Figure 4.3 shows stage head as a function of flow for Japanese and U.S. LH_2 pumps (semi-log scale). Here it can be seen that, although overall head requirements are within U.S. experience, the LE-7 stage head requirement is about 10,000 ft higher than the maximum stage head achieved in the United States (SSME High-Pressure Fuel pump--66,700 ft).

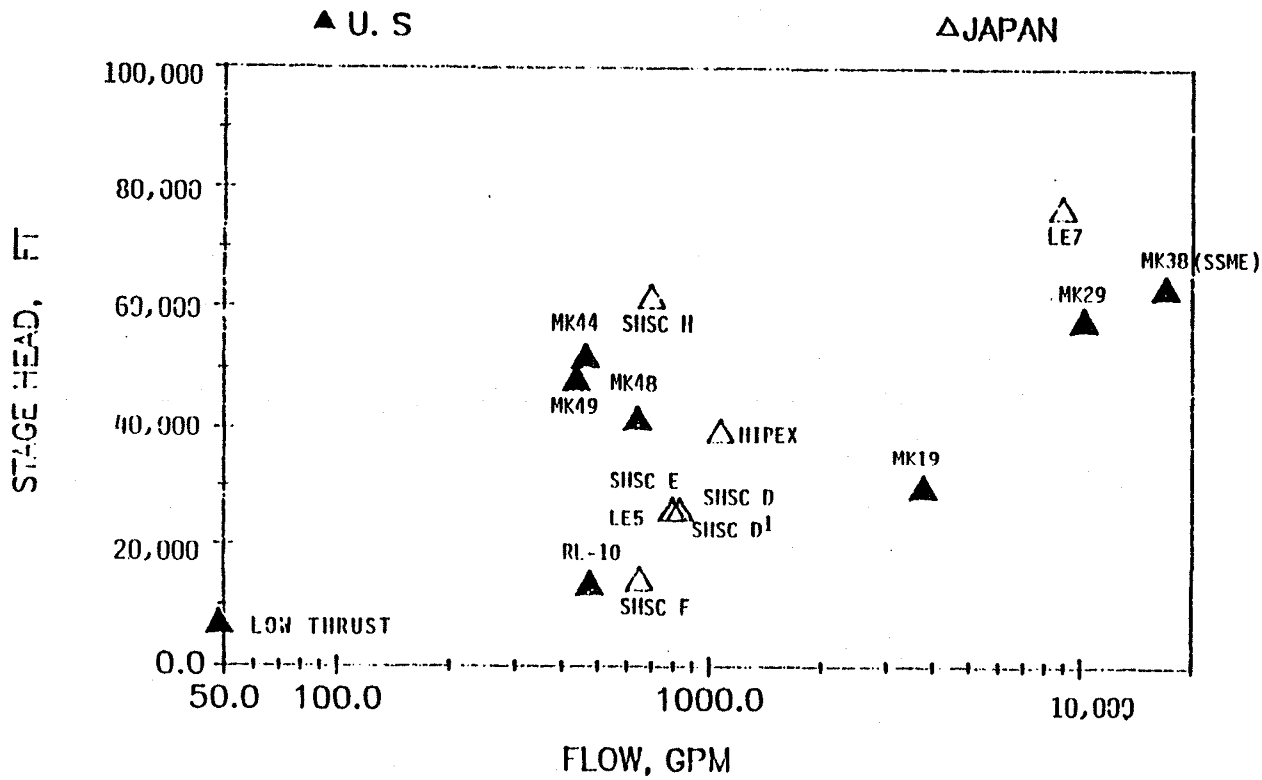

Figure 4.3. U.S. & Japanese LH_2 Centrifugal Pumps
Stage Head vs. Flow

Figure 4.4 compares the achieved efficiency as a function of specific speed (only U.S. pumps for which reliable efficiency data is available have been plotted). Pump size, which has a strong influence on efficiency, is not considered in Figure 4.4.

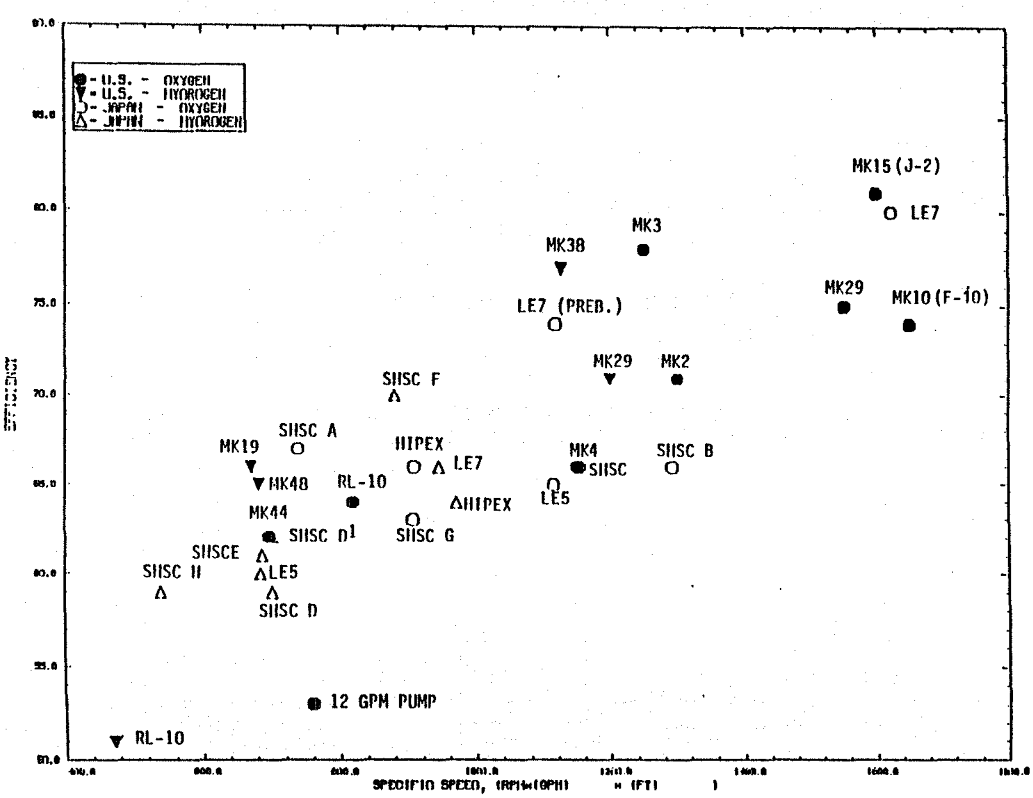

Figure 4.4. Japanese Turbopump Experience

Figures 4.5 and 4.6 show a comparison of the efficiencies either achieved or, in the case of the HIPEX pumps, predicted by the Japanese with those predicted using U.S. correlations taking into account both specific speed and size. Two correlations were used, one reflecting 1970s technology from reference 4.10, and one reflecting 1980s technology.

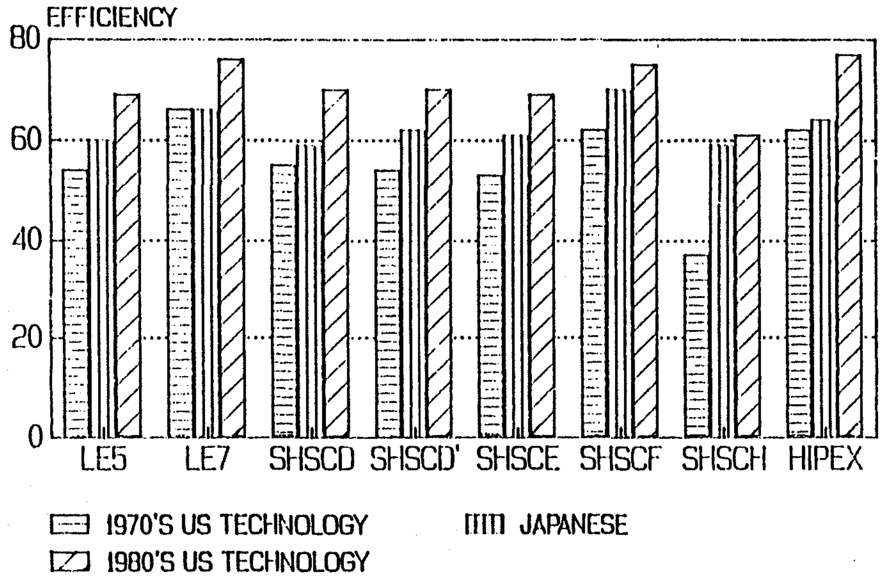

Figure 4.5. Japanese Pump Experience (LH_2)

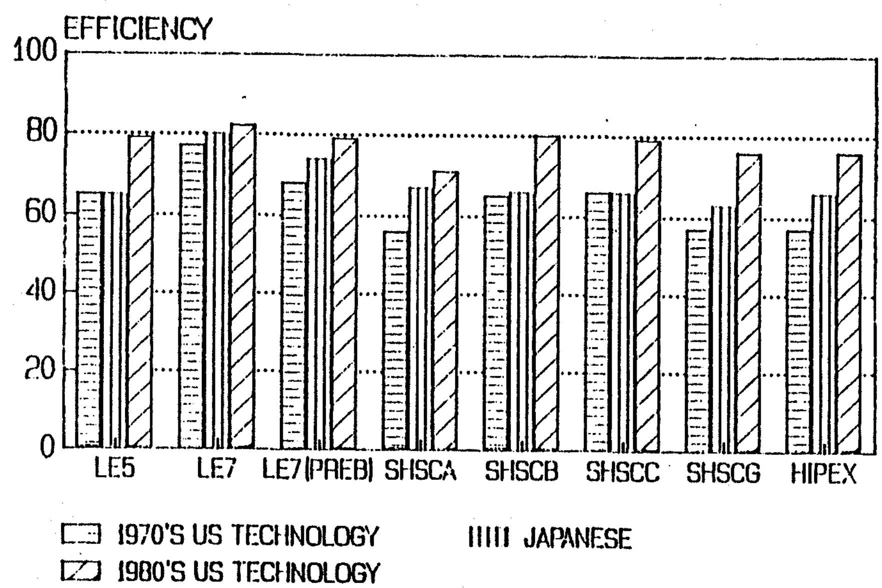

Figure 4.6. Japanese Pump Experience (LOX)

As can be seen from these figures, the current state of the art in Japan is generally ahead of 1970s U.S. technology but behind 1980s U.S. technology. The figures show, however, that the latest Japanese designs - LE-7 LOX, SHSC, F and H - are approaching the efficiencies achievable in the United States.

Cavitation and Suction Performance

Inducer design. Tables 4.3 and 4.4 summarize the parameters quoted for the Japanese inducer designs in the available literature. The design flow coefficients and inlet hub-to-tip ratios are typical for high-suction performance inducers. The quoted head coefficient for the LE-5 LH_2 pump is very low, but test data in the same paper (Ref. 4.3) shows that an inducer head coefficient of 0.2 was actually achieved which is similar to those of U.S. high-suction performance inducers. Inducer design incidence angles are slightly higher than current U.S. design practice and most recent U.S. inducer designs have used four blades rather than three. An odd number of blades can lead to high dynamic loading due to alternate blade cavitation, as was experienced on the Mark 10 (F-1 engine) LOX inducer.

Figures 4.7 and 4.8 compare the required operation suction specific speeds of the Japanese inducer designs to the capability in both water and propellant for a U.S. inducer design of the same flow coefficient and inlet hub-to-tip ratio. In most cases, there is a significant margin between the required operating point and the predicted U.S. capability. This may reflect a more conservative design philosophy but in part reflects a lower suction performance inducer design capability.

Also shown in Tables 4.3 and 4.4 are the achieved suction specific speeds at 10% inducer head loss for the LE-5 LH_2 pump and the SHSC A pump. The LE-5 was tested in LH_2 and achieved a suction specific speed of 45,039, which is below the U.S. predicted capability of 52,178. The SHSC A pump was tested in water and achieved 34,421, which is also below the U.S. predicted capability of 41,209.

Detailed design information in Reference 4.3 suggests that the LE-5 inducers were designed based on meanline geometries and very high solidity blading which is behind that of the United States. It is not known if the technology has advanced in subsequent designs. This lag in technology is supported by the lower suction performance achieved by their inducers.

Two-phase pumping capability. Tables 4.3 and 4.4 include two pumps that were tested in LH_2 to demonstrate two-phase pumping capability (Ref. 4.11). Both pumps appear to be very similar to other designs except for the reported discharge blade angles. Both inducers experienced significant head loss at about

DESIGNATION	FLOW (GPM)	SPEED (RPM)	TIP DIA. (IN)	Dh/Dt (INLET)	PHI-IN	PSI	ACHIEVED Nss	OPERATING Nss	BLADE ANGLES INLET/OUTLET	BLADE NUMBER	INCIDENCE ANGLE
LE5	792	50,000	2.70	0.3	0.1	0.092[1]	45,039[2]	29,831	9.93/ 11.4	3	5.2
LE7	8,916	46,130		0.29				44,866	/		
SHSC D	832	50,000	2.69	0.3[3]	0.088[4]			28,903	9.9 /	3	4.9
SHSC D'	832	50,000	2.69	0.3[3]	0.088[4]			28,903	9.9 /	3	4.9
SHSC E	797	50,000	2.68	0.3[3]	0.085[4]			28,289	9.0 /	3	4.1
SHSC F	650	45,000	2.56	0.3[3]	0.089[4]			33,436	9.9 /	3	4.8
SHSC H	696	80,000	2.48	0.3	0.059			42,297	7.0 /	3	3.6
HIPEX	1,073	83,200		0.29					/		
ATREX	62	17,800							/		
TWO PHASE	792	50,000	2.68	0.3[3]	0.085[4]				9.9 / 13.4		5.0

1 QUOTED DESIGN VALUE, 0.2 ACHIEVED IN TEST
2 10% INDUCER HEAD LOSS IN FULL PUMP TEST IN LH2, 2% PUMP HEAD LOSS ABOVE 100,000
3 ASSUMED - NO DATA AVAILABLE
4 BASED ON ASSUMED HUB TO TIP RATIO

Table 4.3
Japanese Inducer Experience (LH_2)

DESIGNATION	FLOW (GPM)	SPEED (RPM)	TIP DIA. (IN)	Dh/Dt (INLET)	PHI-IN	PSI	ACHIEVED Nss	OPERATING Nss	BLADE ANGLES INLET/OUTLET	BLADE NUMBER	INCIDENCE ANGLE
LE5	274	16,500	2.57	0.3	0.1	0.160		26,431	10.0/12.25	3	4.3
LE7	3,238	20,000	5.91	0.27	0.077			37,333	7.25/	3	2.7
SHSC A	285	20,000	2.56	0.3[2]	0.087[3]	0.2	34,421[1]	20,213	10.3/	3	5.3
SHSC B	285	35,000	2.36	0.3[2]	0.064[3]			35,373	8.0 /	3	4.4
SHSC C	273	16,500	2.57	0.3[2]	0.1[3]			24,681	10.0/	3	4.3
SHSC G	254	45,000	1.86	0.3	0.090			31,108	8.5 /	3	4.2
HIPEX	405	27,000		0.29					/		
TWO PHASE	264	16,500	2.57	0.3[2]	0.097[3]				10.5/ 12.8	3	5.0

1 WATER TEST
2 ASSUMED - NO DATA AVAILABLE
3 BASED ON ASSUMED HUB TO TIP RATIO

Table 4.4
Japanese Inducer Experience (LOX)

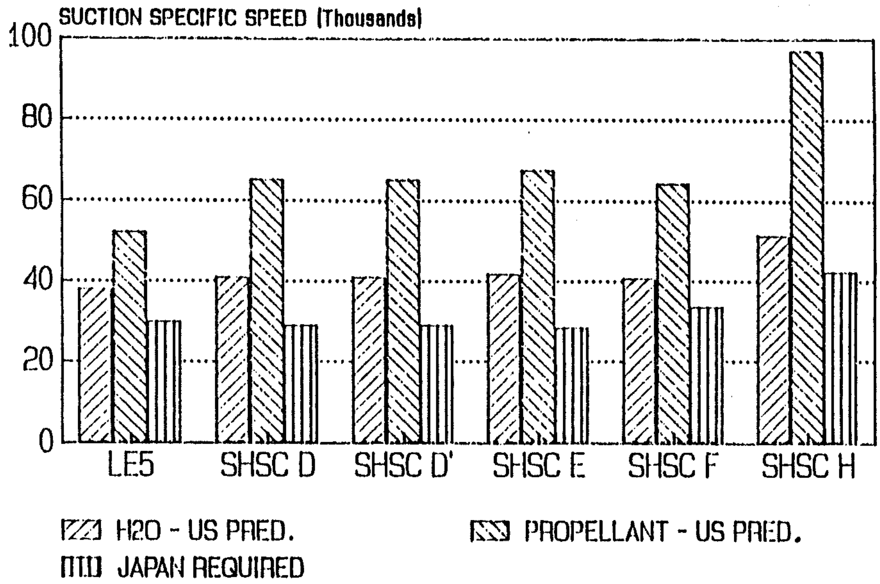

Figure 4.7. Japanese LH_2 Pump Experience

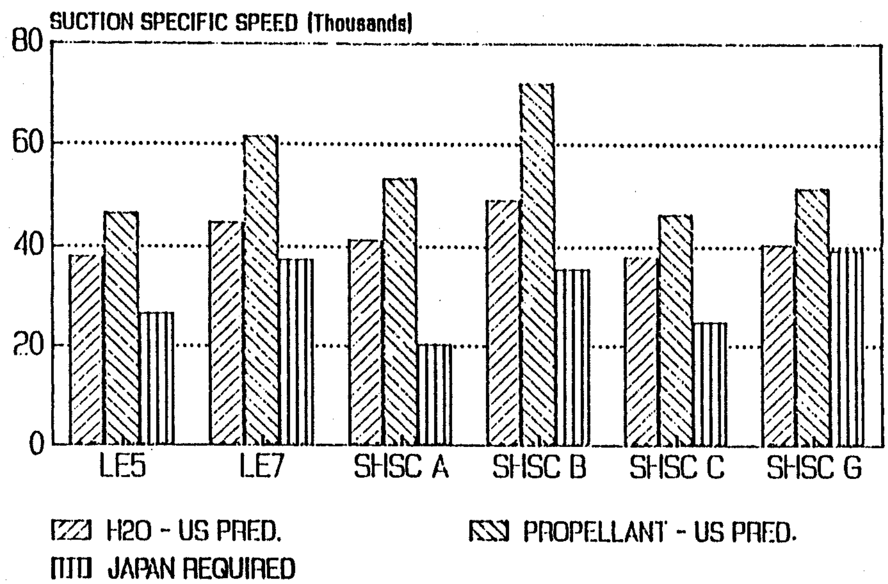

Figure 4.8. Japanese LOX Pump Experience

35% vapor fraction. The overall head for the LOX pump, however, did not decrease even at 40% vapor fraction, as the impeller appears to have made up for the head loss in the inducer. The same was not true for the LH_2 pump where the overall head dropped with the inducer head loss. The United States demonstrated two-phase pumping capability in the early 1970s, achieving vapor fractions of 30%-45%. Although no significant work has been done since that time, U.S. two-phase technology is thought to be leading due to the analytical effort that was undertaken in parallel with the testing program.

Cavitation. The Japanese are conducting extensive research to understand the fundamental physics of cavitation bubble formation and collapse (Refs. 4.12-4.14). Due to the complexity of cavitating flow in an inducer, however, it is likely that it will take five to ten years before this research can be applied to improve inducer design methodology.

Some analytical work is also being done that is more relevant to turbomachinery flows: Reference 4.15 describes the flow of cavitation bubbles through an impeller. The analysis solves the equations of motion for a bubble and the Rayleigh equation for bubble size in a flowfield predicted by a 3D potential flow finite element Computational Fluid Dynamics code. The capability to do this type of analysis exists in the United States but has not yet been applied.

Impeller Design

Tables 4.5 and 4.6 summarize the Japanese impeller design parameters. Figure 4.4 shows that, in general, the Japanese have designed for lower specific speeds than in the United States, which is reflected by the high head coefficients and low flow coefficients of their designs.

The head coefficients achieved by the Japanese in LH_2 are slightly higher than those achieved in the United States. Although not enough data was available to calculate the head coefficient for the LE-7 LH_2 pump, using the maximum impeller tip speed of 2000 ft/sec results in a required head coefficient of 0.65. The maximum head coefficient achieved in the United States in LH_2 is 0.62 (Mark 19), though 0.67 was achieved in RP-1 in the Mark 3 pump. Current U.S. design practice would not attempt such a high head coefficient because the pump head-flow characteristic tends to become very flat, which is detrimental to engine system stability. The head coefficients achieved in LOX by the Japanese are similar to those achieved in the United States.

The low flow coefficients are similar to many U.S. designs, and are driven by the high head coefficient and small envelope requirements of rocket engine pumps.

Table 4.5
Japanese Impeller Experience (LH_2)

DESIGNATION	FLOW (GPM)	SPEED (RPM)	STAGE HEAD (FT)	# OF BLADES (IMPELLER)	BLADE ANGLES IN/OUT (IMP.)	IMP. TIP BLADE WIDTH	IMP. D(tip) OUT	IMP U tip	STAGE HEAD COEFF.(TEST)	PHI [2] DISCH.	STAGE Ns (PREDICTED)
LE5	792	50,000	26,108	6+6	/	0.180	5.75	1,254	0.56	0.062	685
LE7	8,916	46,130	76,782		/60 [1]						944
SHSC D	832	50,000	26,100	6+6	11.4/35	0.150	5.75	1,254	0.57	0.079	702
SHSC D'	832	50,000	26,100	6+6+12	14.2/45	0.193	5.28	1,152	0.63	0.072	702
SHSC E	797	50,000	26,100	6+6	12.7/35	0.142	5.50	1;20		0.087	687
SHSC F	650	45,000	14,237	6+6	12.0/35	0.169	4.80	942	0.56	0.087	880
SHSC H	696	80,000	61,691	6+6+12	9.21/45	0.114	5.10	1,780	0.62	0.069	539
HIPEX	1,073	83,200	39,609		/						971
ATREX	62		9,720		/						

1 PROTOTYPE, FLIGHT PUMP ANGLE INCREASED TO 70 DEG. DUE TO PROTOTYPE HEAD BEING 3% BELOW DESIGN VALUE
2 CALCULATED WITHOUT BLADE BLOCKAGE

Table 4.6
Japanese Impeller Experience (LOX)

DESIGNATION	FLOW (GPM)	SPEED (RPM)	STAGE HEAD (FT)	# OF BLADES (IMPELLER)	BLADE ANGLES IN/OUT (IMP.)	IMP.TIP BLADE WIDTH	IMP. D(tip) OUT	IMP. U tip	STAGE HEAD COEFF.(TEST)	PHI [1] DISCH.	STAGE Ns (PREDICTED)
LE5	274	16,500	1,535	6	/		4.38	314	0.50		1114
LE7	3,238	20,000	6,236	6	/22.5	0.58	7.72	674	0.48	0.110	1622
LE7 (PREB.)	619	20,000	3,401	5	/20	0.30	5.65	493	0.45	0.076	1117
SHSC A	285	20,000	3,520	6	13.8/25	0.165	5.28	461	0.52	0.072	739
SHSC B	285	35,000	3,520	6	11.25/25	0.362	3.27	499	0.48	0.049	1293
SHSC C	273	16,500	1,462	6	14.0/25	0.268	4.38	315	0.50	0.075	1153
SHSC G	254	45,000	7,308	6	11.0/25	0.150	3.40	668		0.082	907
HIPEX	405	27,000	5,041		/						908

1 CALCULATED WITHOUT BLADE BLOCKAGE

Blade spacing and partial blade rows used to accomplish the high head coefficient for LH_2 pumps are similar to U.S. designs. In the LOX designs, the Japanese do not use partial blades to avoid the problem of correct placement of the partials, but this can lead to large blade blockage at the eye of the impeller, or large deviation angles at the discharge. Based on the cross-sections presented in the various references, the blockage at the eye is reduced by starting the blade further into the passage so the leading edge is at a high radius at the hub. This is common practice in pumps for commercial application in the United States. The lower head coefficients of the LOX designs compared to the LH_2 designs means that the partial blades are not essential to achieve the required head.

Reference 4.3 states that the impeller blades are "optimized on two surfaces." This is the same approach used in the United States for the definition of impeller blade geometries, where the hub and shroud profiles are optimized and connected with straight line blade elements.

Diffusion Systems

The limited information obtained on the crossovers, diffusers, and volutes used in the Japanese pump designs is summarized in Table 4.7. Both vaned and vaneless diffusers have been used, though it is not possible to determine whether vane island or double circular arc diffuser vanes have been utilized. Double tongue volutes are reported for both LE-5 pumps, which both reduces radial loads with the vaneless diffuser configuration, and provides structural support for the volute separating loads. Based on the fact that the Japanese have achieved reasonable efficiencies in high head coefficient designs, it appears that Japanese diffuser technology is similar to that in the United States.

The internal interstage crossovers shown on the available pump cross-sections appear to have bladed upcomers and downcomers with an annular turning section. Testing in the design phase of the SSME High-Pressure Fuel Pump showed that higher efficiency could be obtained with the continuous passage design which has been used in all subsequent multistage pump designs in the United States.

Analytical Capabilities

No papers were found on preliminary design methods in Japan, though experience with one company suggests that the Japanese use two-dimensional deviation angle and loss correlations that do not consider blade variations from hub to tip.

Table 4.7
LE-5 TURBINES

TURBINES		FUEL	OXIDIZER
POWER (1)	HP	665	182
SPEED	RPM	50,000	16,500
TORQUE	FT-LBF	70	58
FLOWRATE	LBM/SEC	0.953	0.860
INLET TEMPERATURE	R	1561	1247
INLET PRESSURE	PSIA	353	71.6
OUTLET PRESSURE	PSIA	73.2	38.4
PRESSURE RATIO	-	4.82	1.87
VELOCITY RATIO (2)	-	0.187	0.10
EFFICIENCY	PERCENT	47.6	39.2
STAGING	-	2RVC(3)	2RVC(3)
ADMISSION	-	PARTIAL	FULL
SHROUDED ROTORS	-	NO	NO
ISENTROPIC AVAILABLE ENERGY (1)	BTU/LBM	1037	382
ISENTROPIC VELOCITY	FT/SEC	7206	4376
PITCH VELOCITY	FT/SEC	1350	440
PITCH DIAMETER	INCH	6.2	6.1

(1) Calculated using J-2 Gas Properties
(2) Read from EFF'Y Curves at Design Pres. Ratio and Efficiency
(3) 2 Rotor, Velocity Compounded, Impulse Staging

Reference 4.3 states that the methods outlined in Reference 4.17 are used for inducer analysis. This approach is an axisymmetric control volume analysis with simplified calculations of the blade loading. More advanced analytical techniques are now used in the United States.

The impellers are analyzed using a quasi-3D potential stream function approach (Ref. 4.18) similar to the Katsanis programs used in the United States (Ref. 4.19). This method can still be used effectively for analysis, but other quasi-3D methods or fully 3D Computational Fluid Dynamics (CFD) codes are currently being used for analysis in the United States.

As mentioned earlier, only one paper was found on the application of CFD to turbomachinery (Ref. 4.15) which described a 3D potential flow finite element analysis of an impeller. As indicated in Chapter 7, Japan's capabilities in CFD are on a par with those of the United States, so it will probably begin CFD analysis of turbomachinery for future applications.

Pump Conclusions

Japanese technology is at a similar level to that in the United States in impeller design, diffusion system design, and overall pump performance capability. The Japanese are behind in inducer design technology, although they are ahead in the basic understanding of cavitating flows.

Regarding the pump designs that are currently underway in Japan, both pumps for the HIPEX engine are within the current Japanese experience. The ATREX LH_2 pump requires a much lower flowrate than prior experience and may require some development effort.

TURBINES

Japanese Turbine Experience

The Japanese turbine experience includes both subsonic and supersonic, both partial admission and full admission, and both impulse and reaction blading. Turbines for the gas generator cycle LE-5 engine LH_2 and LO_2 turbopumps both have two-row velocity-compounded (2 RVC) impulse blading arranged in series similar to the turbines for the J-2 engine.

Both LE-7 turbines are single stage turbines arranged in parallel to extract the energy from the relatively low-pressure ratio high flow rate staged combustor cycle, similar to the SSME turbines. The expander cycle HIPEX turbines are arranged in series to extract the energy from the heated hydrogen, similar to the OTV MK48 turbines.

A comparison of the Japanese rocket engine turbine experience and turbines developed for U.S. rocket engines is presented in Figure 4.9. Although the Japanese turbines fall well within U.S. experience, the design methodology utilized and the efficiencies obtained are equivalent to those in the United States.

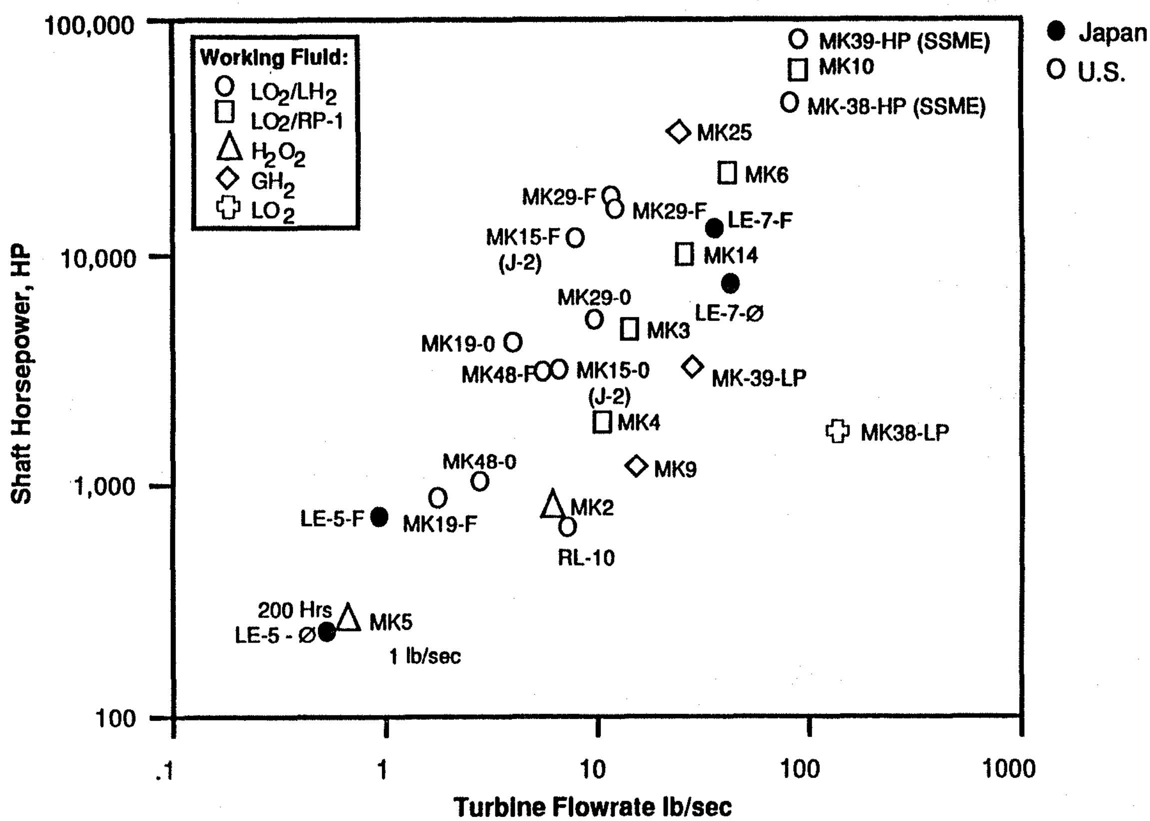

Figure 4.9. U.S. & Japanese Rocket Engine Turbines

LE-5 Turbines

The LE-5 turbines are arranged in series for the gas generator engine cycle, (Fig. 4.10), similar to the J-2 engine. The LE-5 turbines are both two-rotor, velocity-compounded (2 RVC), with supersonic partial admission fuel blading and subsonic full admission oxidizer blading. Both turbines have unshrouded, constant section blading and were designed based on M-1 turbine design reports (Ref. 4.3). Efficiencies of the LE-5 turbines and the J-2 turbines are compared in Figure 4.11. The LOX turbine efficiencies are comparable, but the partial admission LE-5 fuel turbine has significantly lower efficiency than the full admission MK15 fuel turbine.

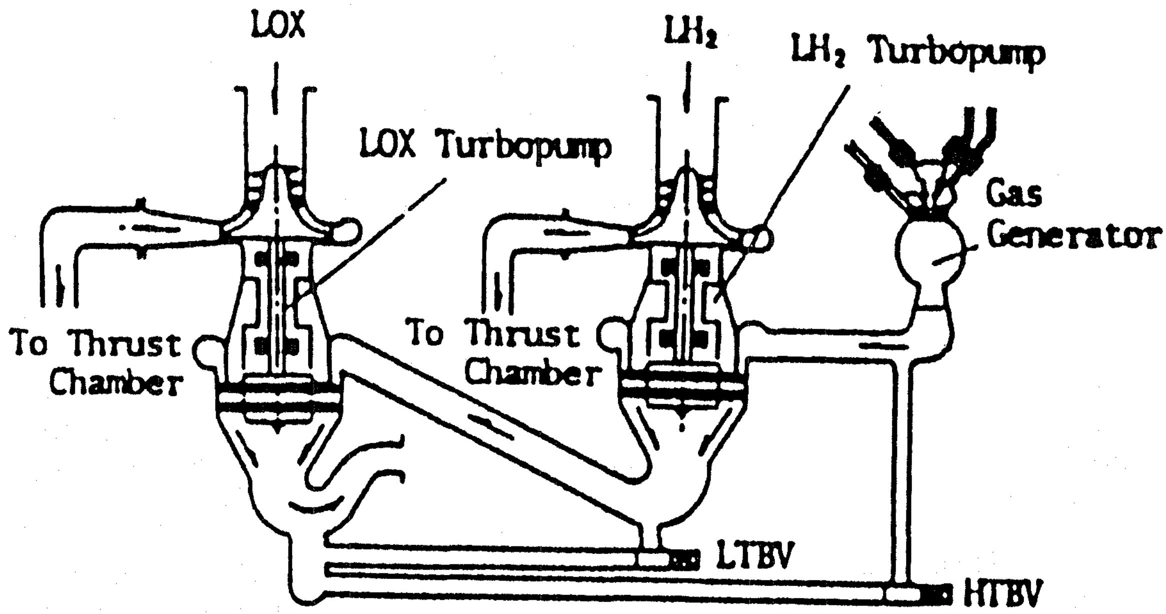

Figure 4.10. Turbopump System Schematic

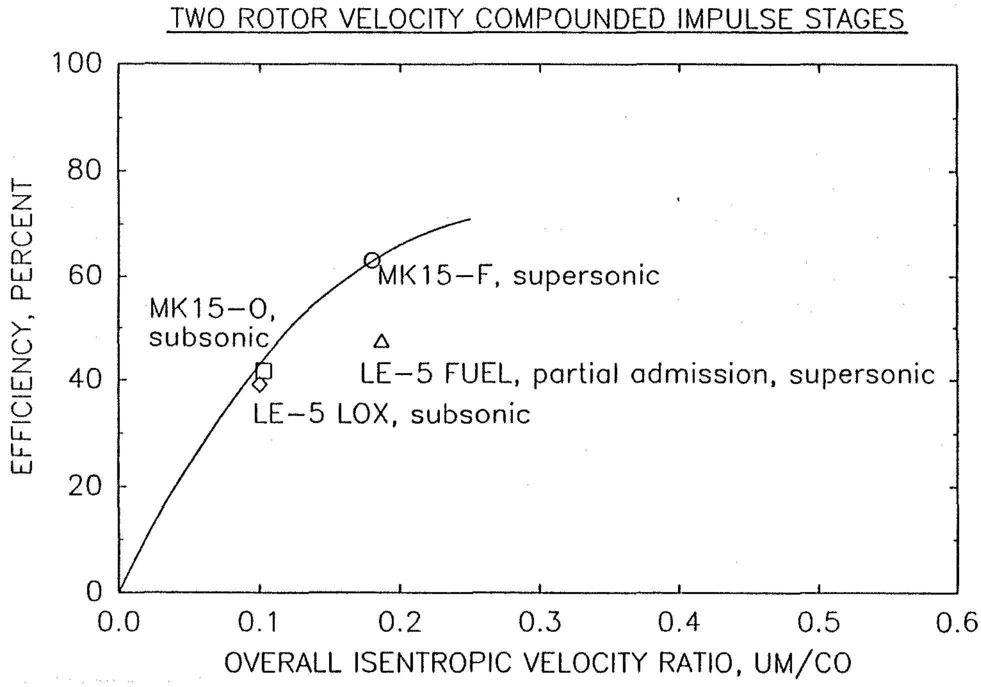

Figure 4.11. JTEC Turbine Comparisons

The LE-5 turbine parameters are listed in Table 4.7. Turbine powers and available energy were determined using J-2 turbine gas properties for the specified turbine pressures and temperatures in Reference 4.3. The turbine velocity ratios, blade speeds, and diameters were determined from the efficiency curves and the design efficiencies in Reference 4.3.

A cross-section of the LE-5 fuel turbine is shown in Figure 4.12 and the blade profiles are shown in Figure 4.13. Leading edges for the stator and second rotor are more blunt than would be selected in the United States for impulse blading. Test performance (Fig. 4.14), varies primarily with velocity ratio and secondarily with pressure ratio for the supersonic stage.

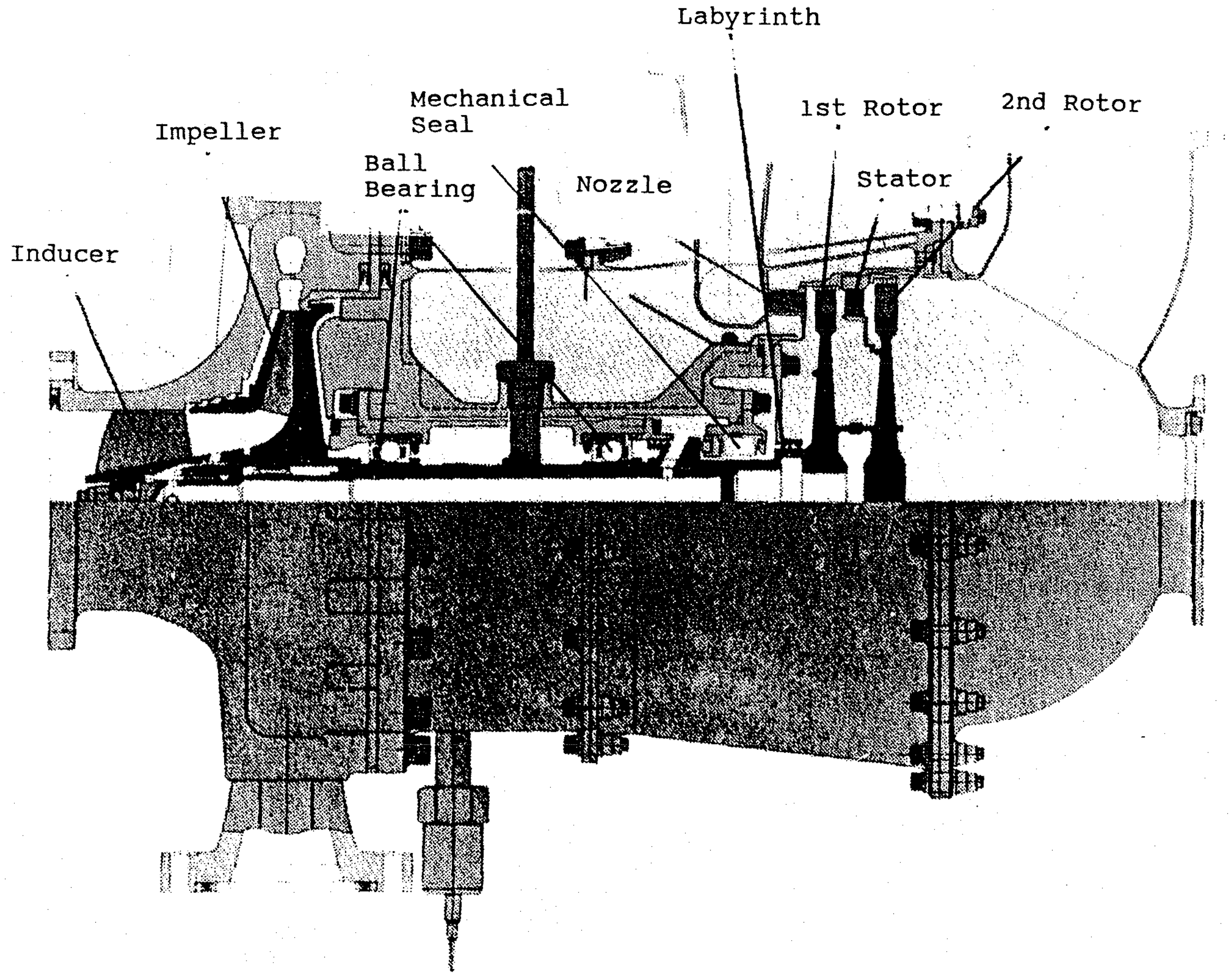

Figure 4.12. LE-5 Liquid Hydrogen Turbopump

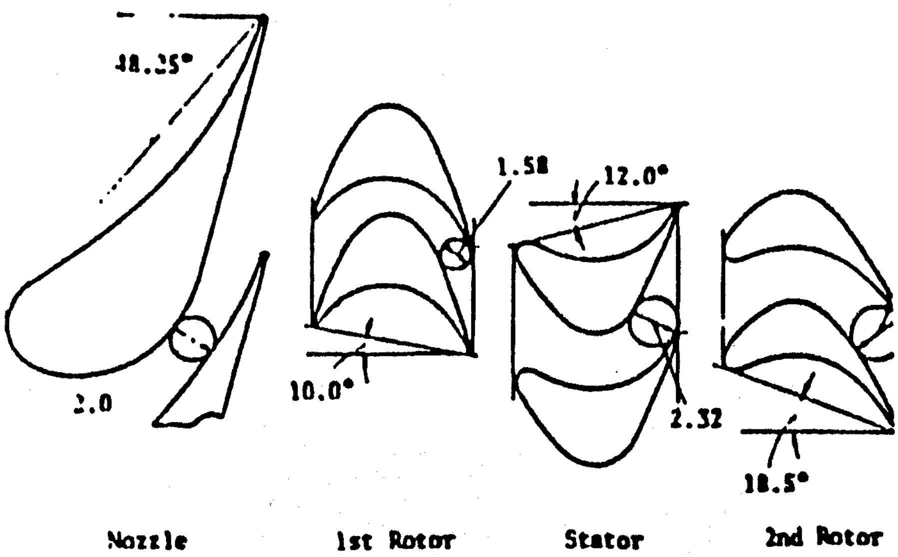

Figure 4.13. LE-5 Blade Profiles of LH_2 Turbopump Turbine

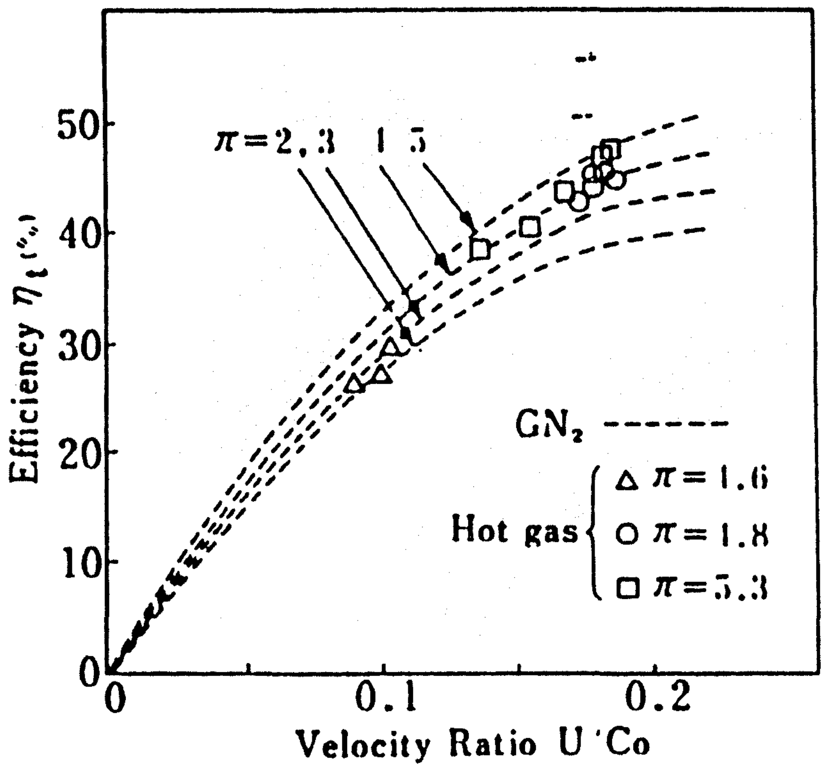

Figure 4.14. LE-5 LH_2 Turbopump Turbine Efficiency

A cross-section of the LE-5 LOX turbine is shown in Figure 4.15 with a single disk and two rotating blade rows. Blunt leading edges were also used for the subsonic design, as shown in Figure 4.16. Test performance, shown in Figure 4.17, varies with velocity ratio, but not significantly with pressure ratio for the subsonic stage.

The LE-5A engine uses a split expander thrust chamber, heating part of the hydrogen flow in a hydrogen bleed cycle. Reference 4.20 reports that the same turbines are used to power the LE-5A as the LE-5 engine.

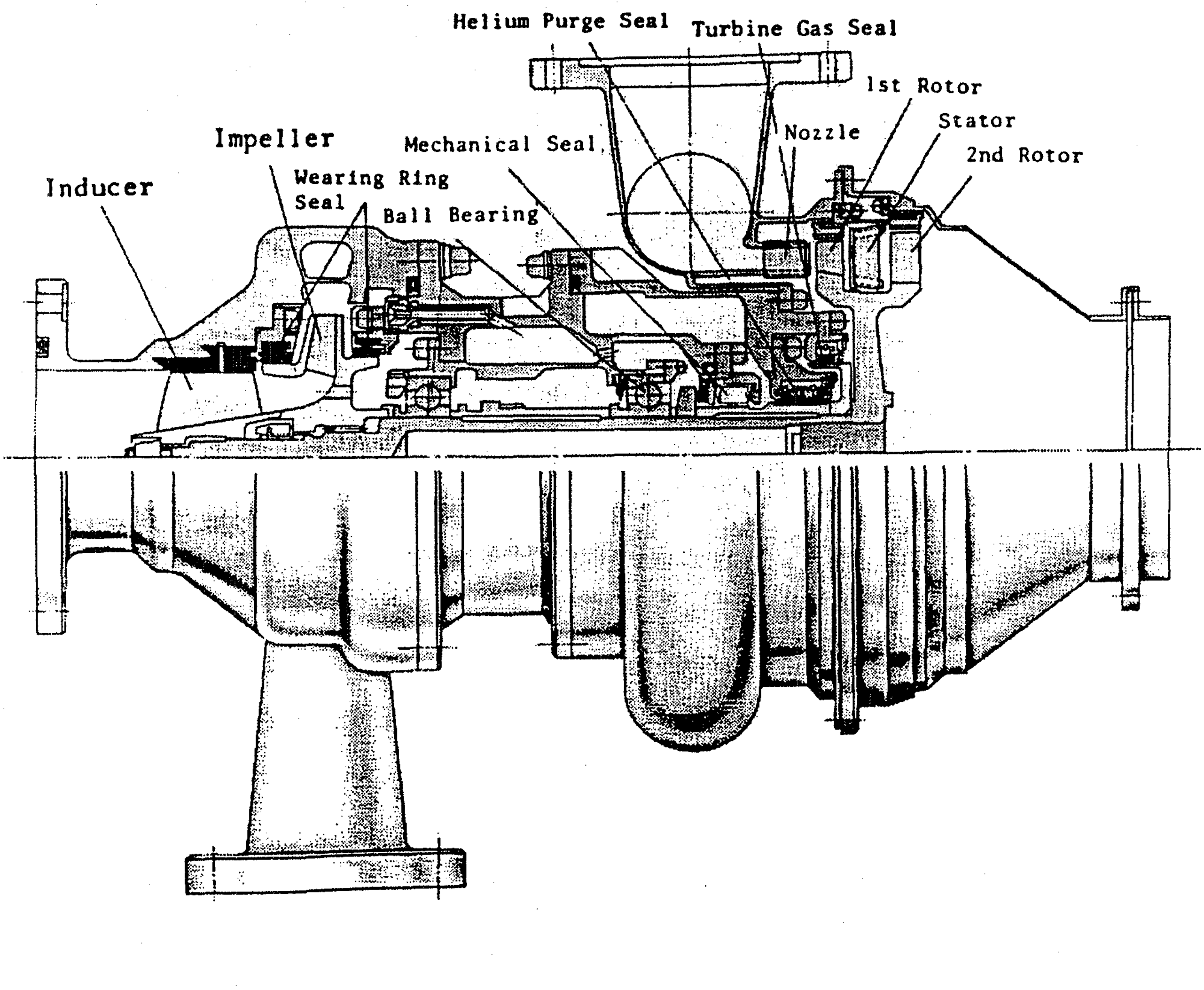

Figure 4.15. LE-5 Liquid Oxygen Turbopump

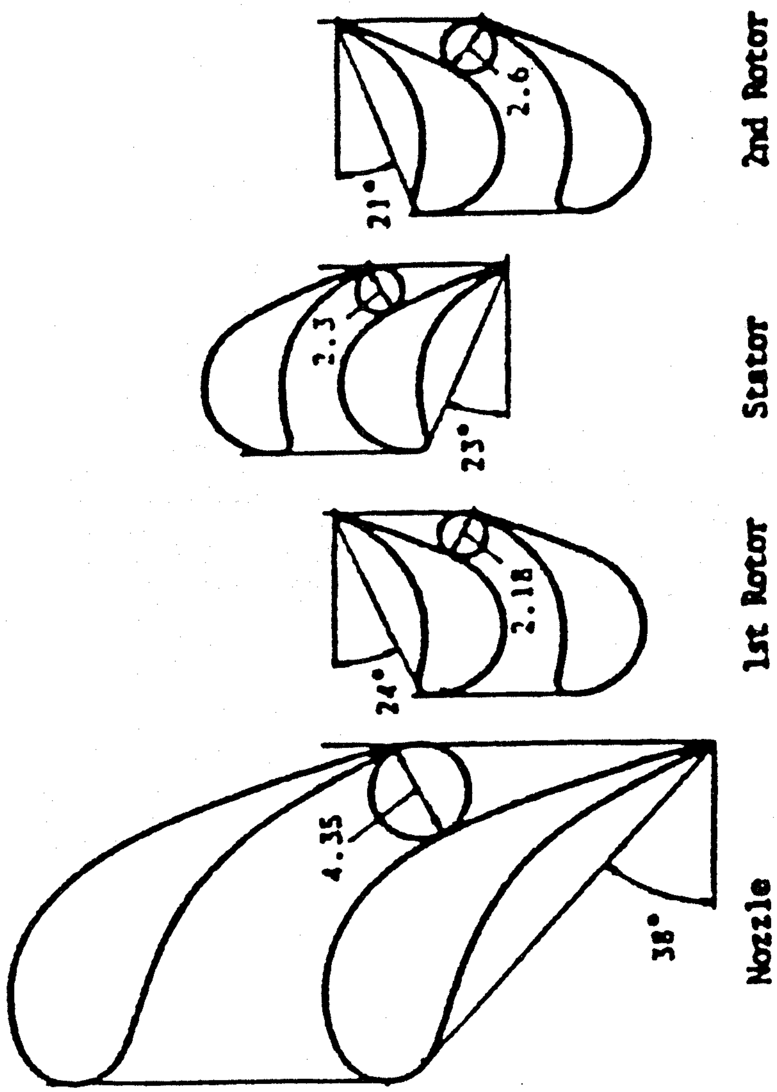

Figure 4.16. LE-5 Blade Profiles of LOX Turbopump Turbine

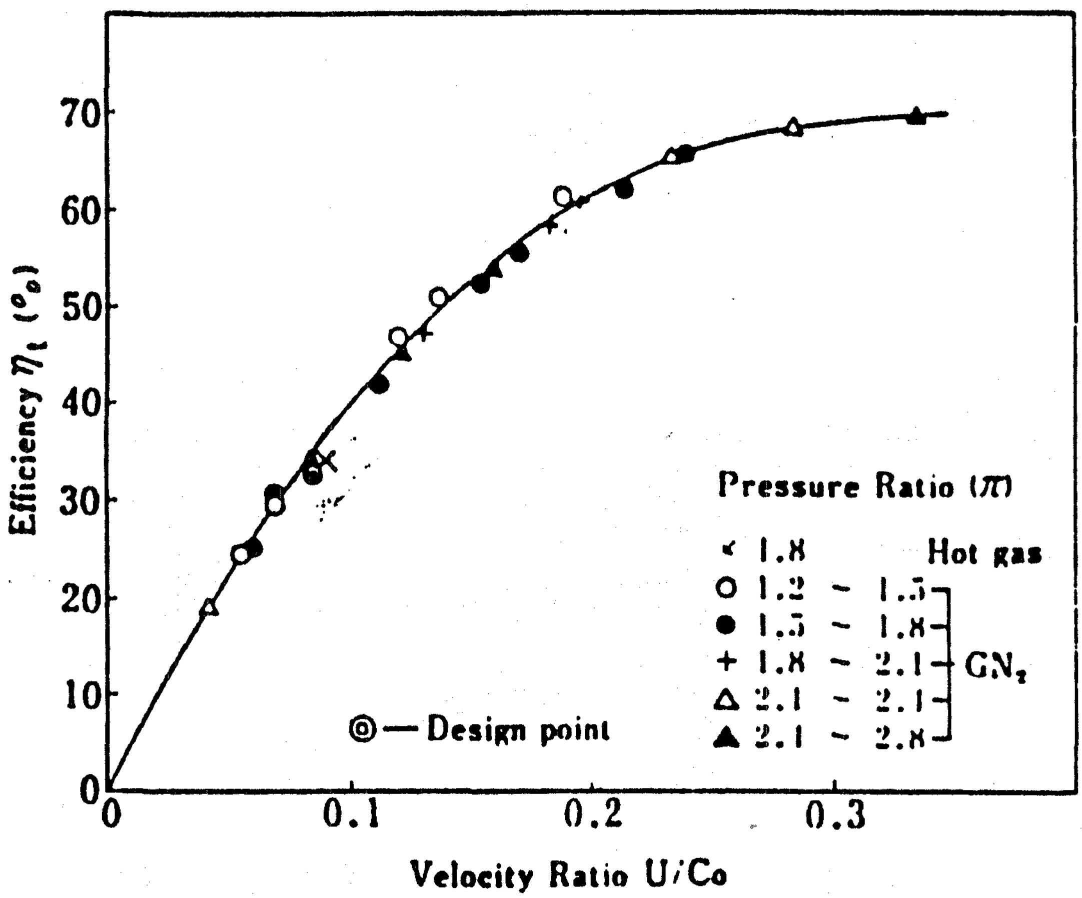

Figure 4.17. LE-5 LOX Turbopump Turbine Efficiency

LE-7 Turbines

The LE-7 turbines are arranged in parallel for the staged combustion engine cycle. A schematic of the LE-7 engine system is shown in Figure 4.18 (Ref. 4.5). Both turbines are single stage for simplicity of design and rotordynamic considerations resulting in lower performance than the two-stage SSME turbines. The efficiencies plotted along a characteristic for single rotor turbines are shown in Figure 4.19. The two-stage SSME turbine efficiencies are higher for similar overall velocity ratios, as shown in Figure 4.20. Unshrouded integrally bladed (blisk) rotors were initially utilized for the LE-7 turbines but were modified for fir-tree blades to reduce transient thermal stress at the blade roots. The LE-7 LOX turbine, initially designed as partial admission, was modified to full admission to reduce dynamic blade loads with a slight reduction in performance for the shorter blade heights per Reference 4.6. LE-7 turbine parameters are presented in Table 4.8.

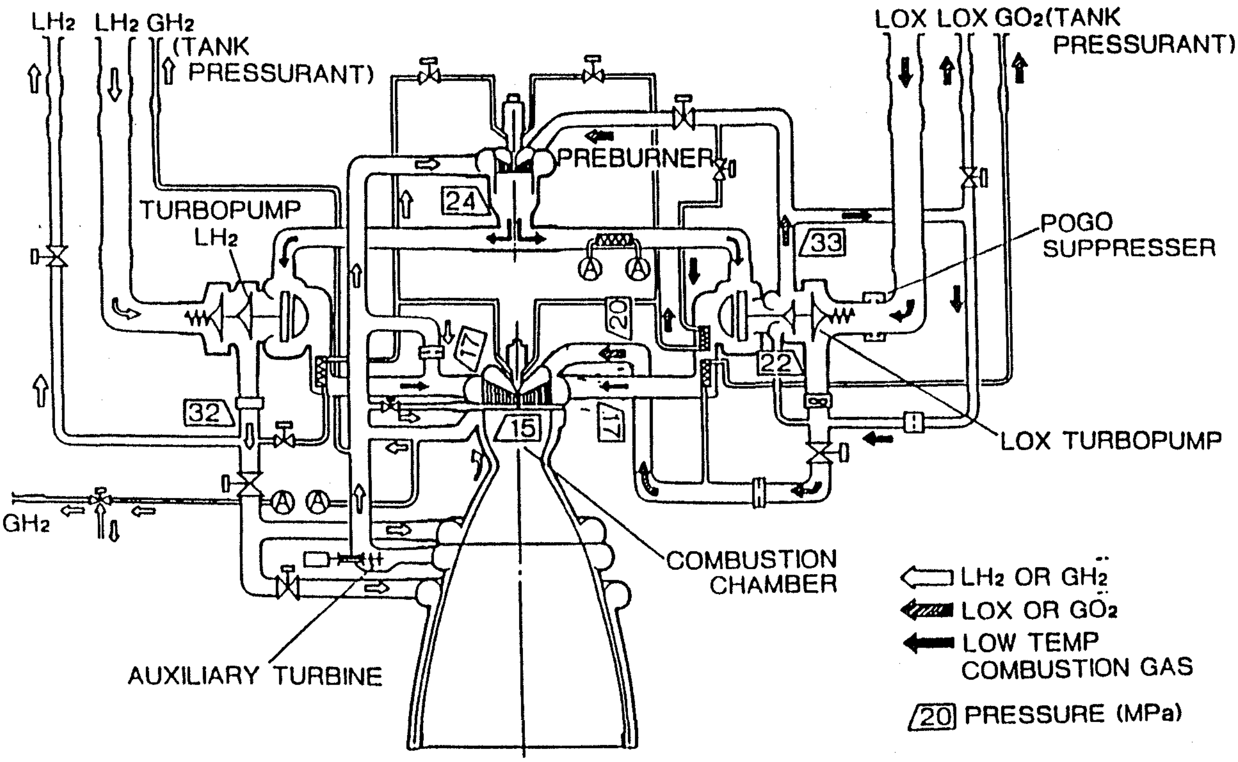

Figure 4.18. LE-7 Engine Schematic

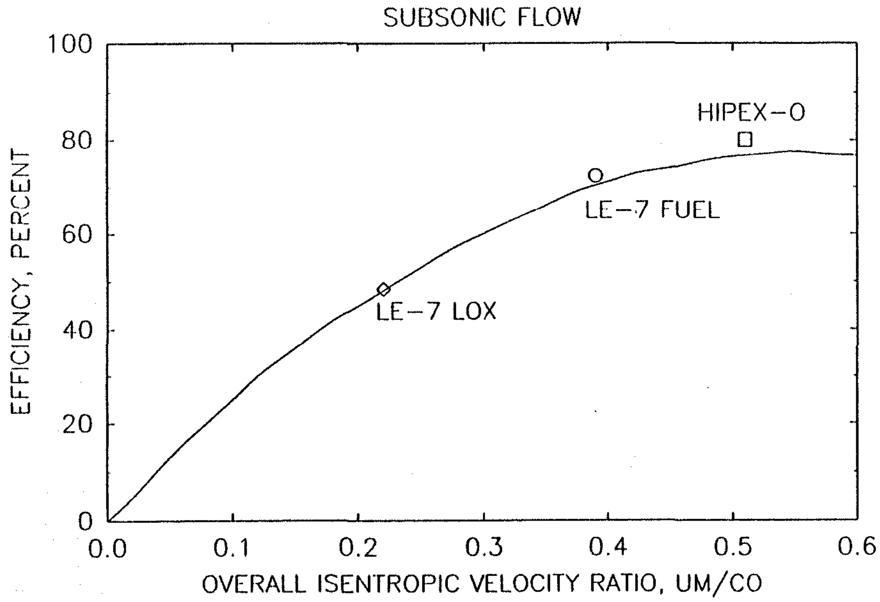

Figure 4.19. JTEC Turbine Comparisons Single Rotor Stages

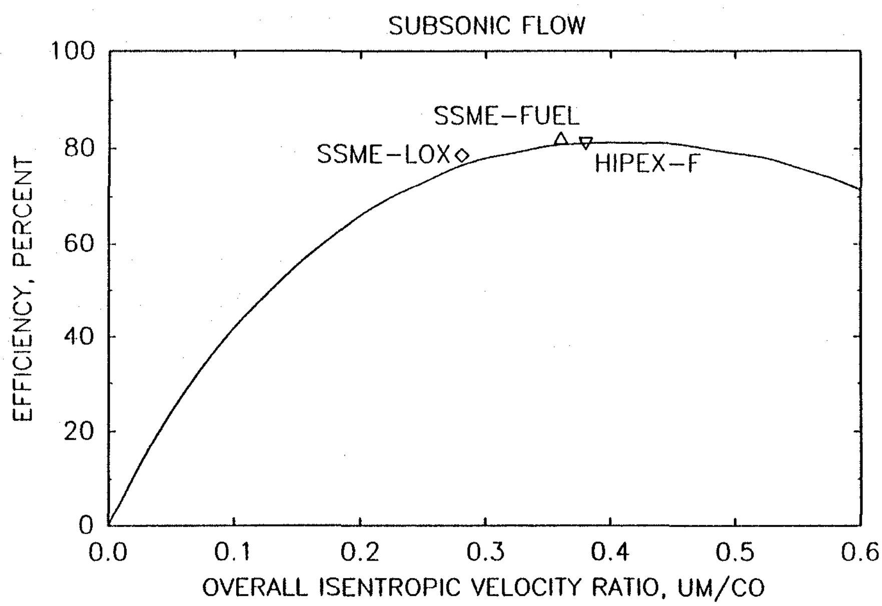

Figure 4.20. JTEC Turbine Comparisons Two Rotor Stages

Table 4.8
LE-7 Turbine Parameters

TURBINE	FUEL	OXIDIZER
POWER	30,777 HP (22,960 KW)	8579 HP (6400 KW)
SPEED	46,130 RPM	20,000 RPM
TORQUE	3504 FT-LBF	2253 FT-LBF
INLET TEMPERATURE	1769 R (982.7K)	1746 R (970 K)
INLET PRESSURE	3414 PSIA (24.29 MPa)	3303 PSIA (23.5 MPa)
PRESSURE RATIO	1.48	1.43
OUTLET PRESSURE	2307 PSIA	2310 PSIA
NUMBER OF ROTORS	1 WITH STR. VANES	1 WITH STR. VANES
ADMISSION	100 PERCENT	100 PERCENT (WAS 60)
VELOCITY RATIO	0.39	0.22
EFFICIENCY	72.3 PERCENT	48.5 PERCENT
GH2 TEST EFFICIENCY	75.9 PERCENT (PLUS 5%)	48.5 PERCENT
TIP SPEED	1640 FT/SEC (500 M/S)	930 FT/SEC
TIP DIAMETER	8.15 INCH	10.6 INCH
BLADE ATTACHMENT	FIR TREE (WAS BLISK)	FIR TREE (WAS BLISK)
REMARKS	OVERSPEED TO 51,000 RPM, NO CRACKS	
BLADE MATERIALS	MAR M-247	MAR M-247

Cross-sections of the LE-7 fuel turbine are shown in Figures 4.21 and 4.22 (Ref. 4.4). The turbine blades are uncooled while the hub portion of the disk is cooled with gaseous hydrogen on both sides. The LE-7 fuel turbine was designed for a tip speed of 1640 feet/second which is comparable to U.S. designs at 1800R operating temperature. A three-dimensional flow analysis program was used to design the blades with varying reaction from hub to tip. The blades, which are directionally solidified MAR-M-247, have experienced cracking problems due to resonant vibration and are in the process of being redesigned. The test efficiency versus velocity ratio is shown in Figure 4.23 (Ref. 4.4).

The LE-7 LOX turbine cross-section is shown in Figures 4.24 and 4.25 (Ref. 4.6). GH_2 is used to cool the turbine disk on both sides, as shown in Figure 4.25. The turbine test efficiency is shown in Figure 4.26 for both the full and partial admission stages.

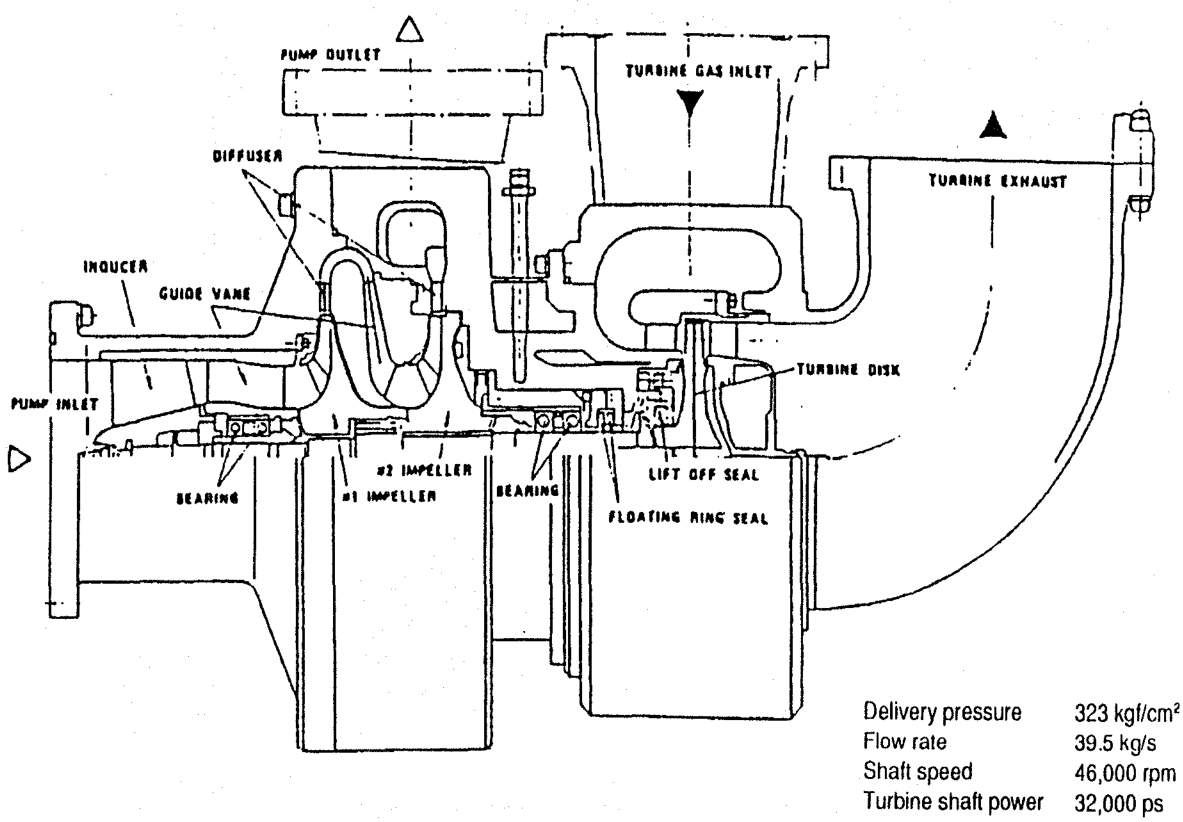

Figure 4.21. LE-7 Fuel Turbopump

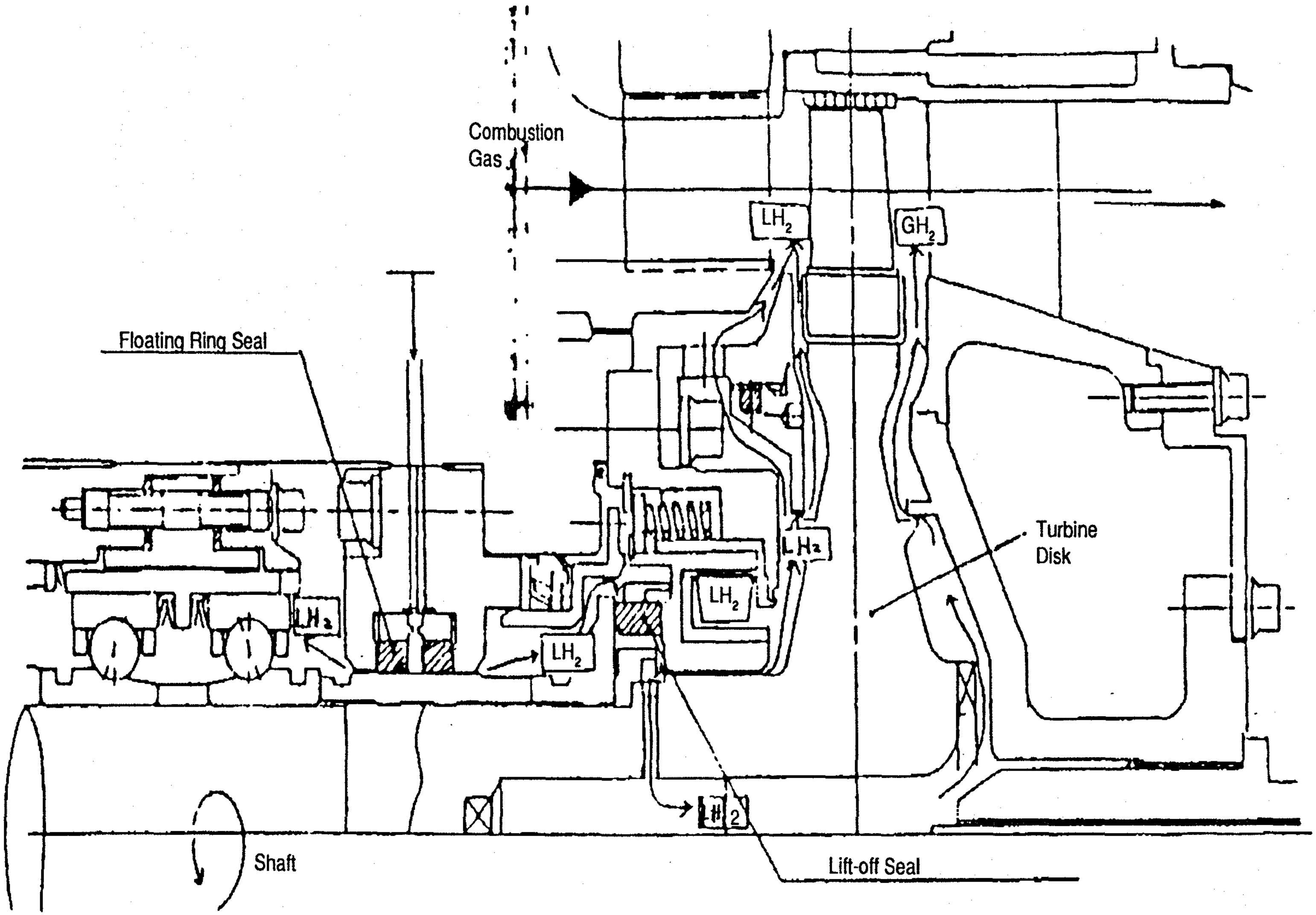

Figure 4.22. Seal System and Disk Coolant Flow

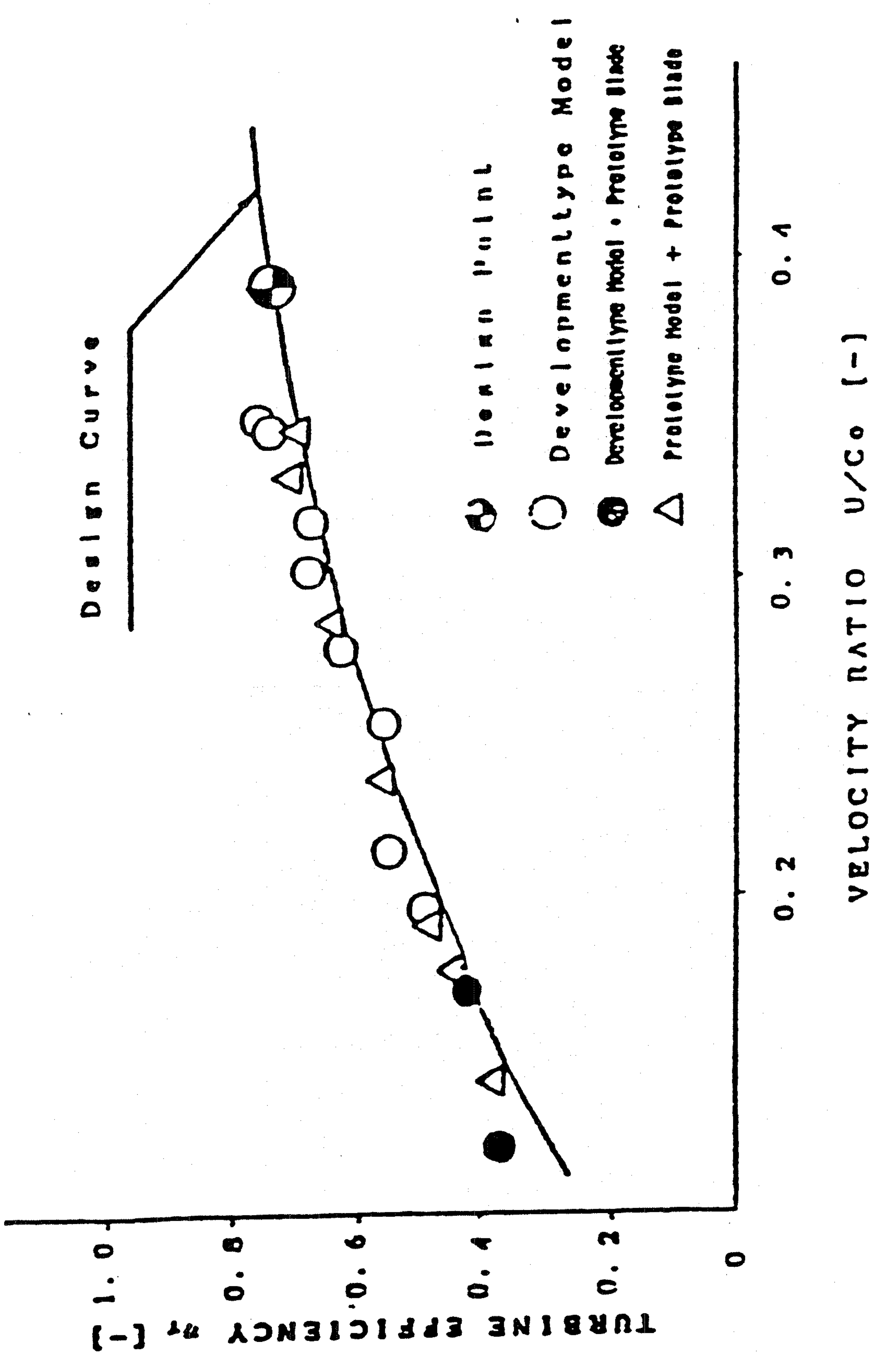

Figure 4.23. FTP Turbine Efficiency

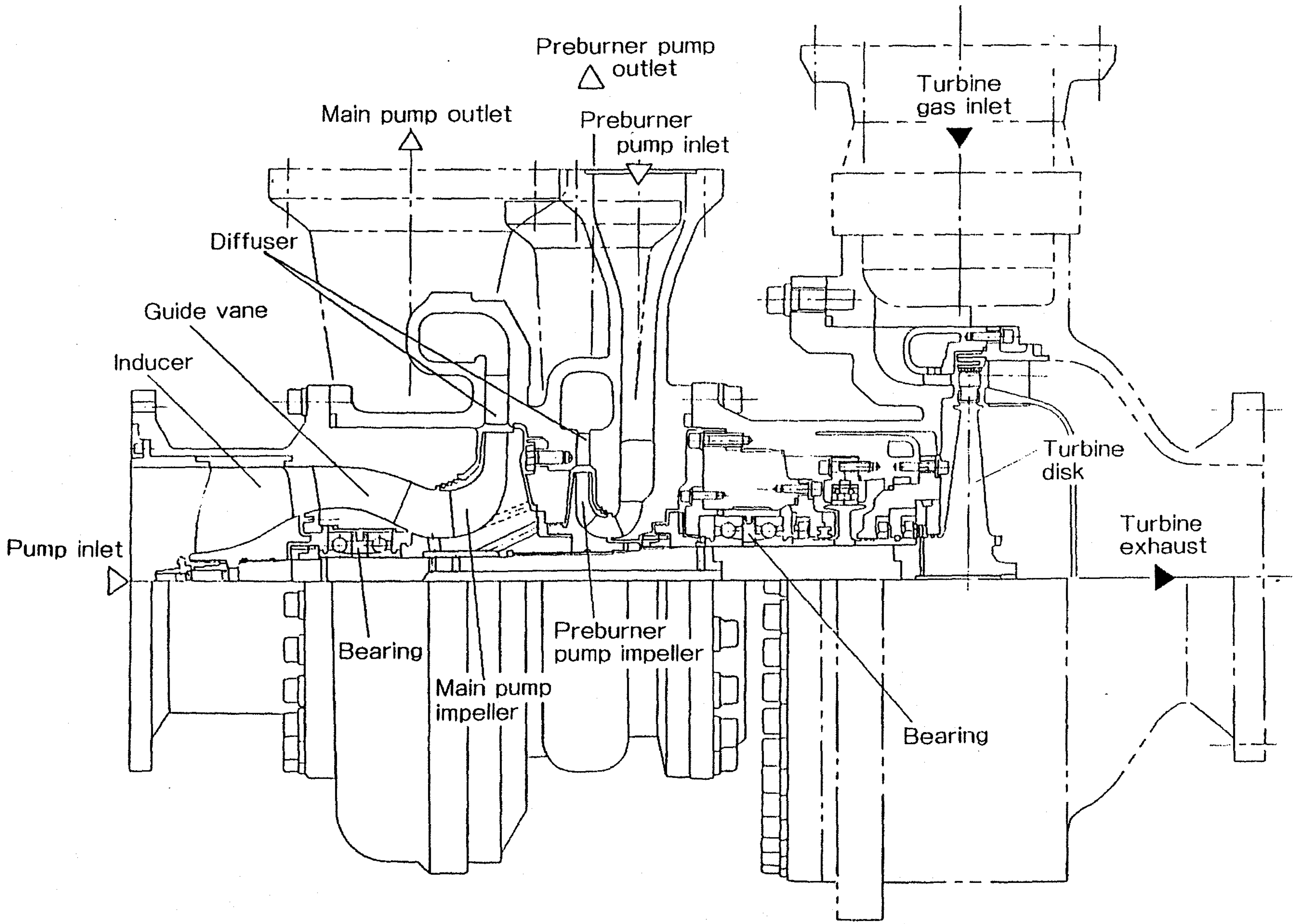

Figure 4.24. LE-7 LOX Turbopump

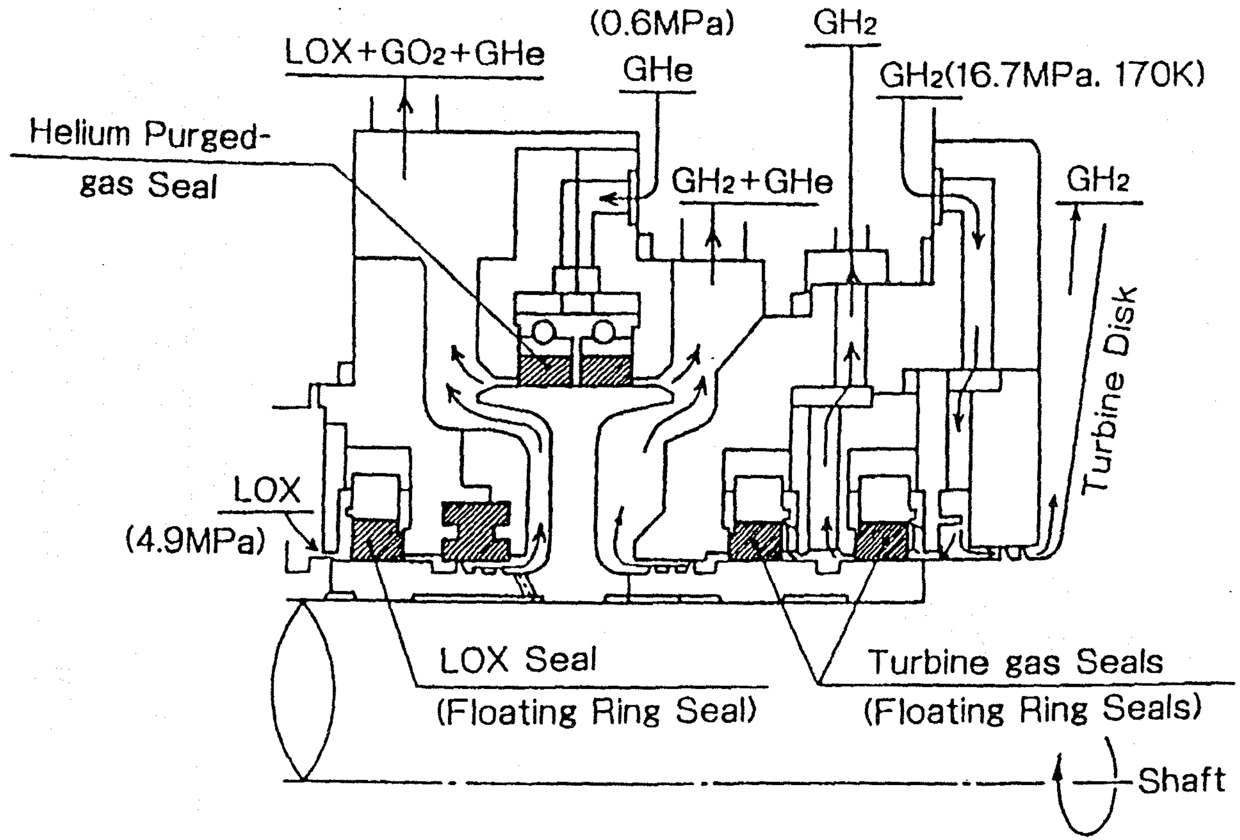

Figure 4.25. Shaft Seal System of LOX Turbopump

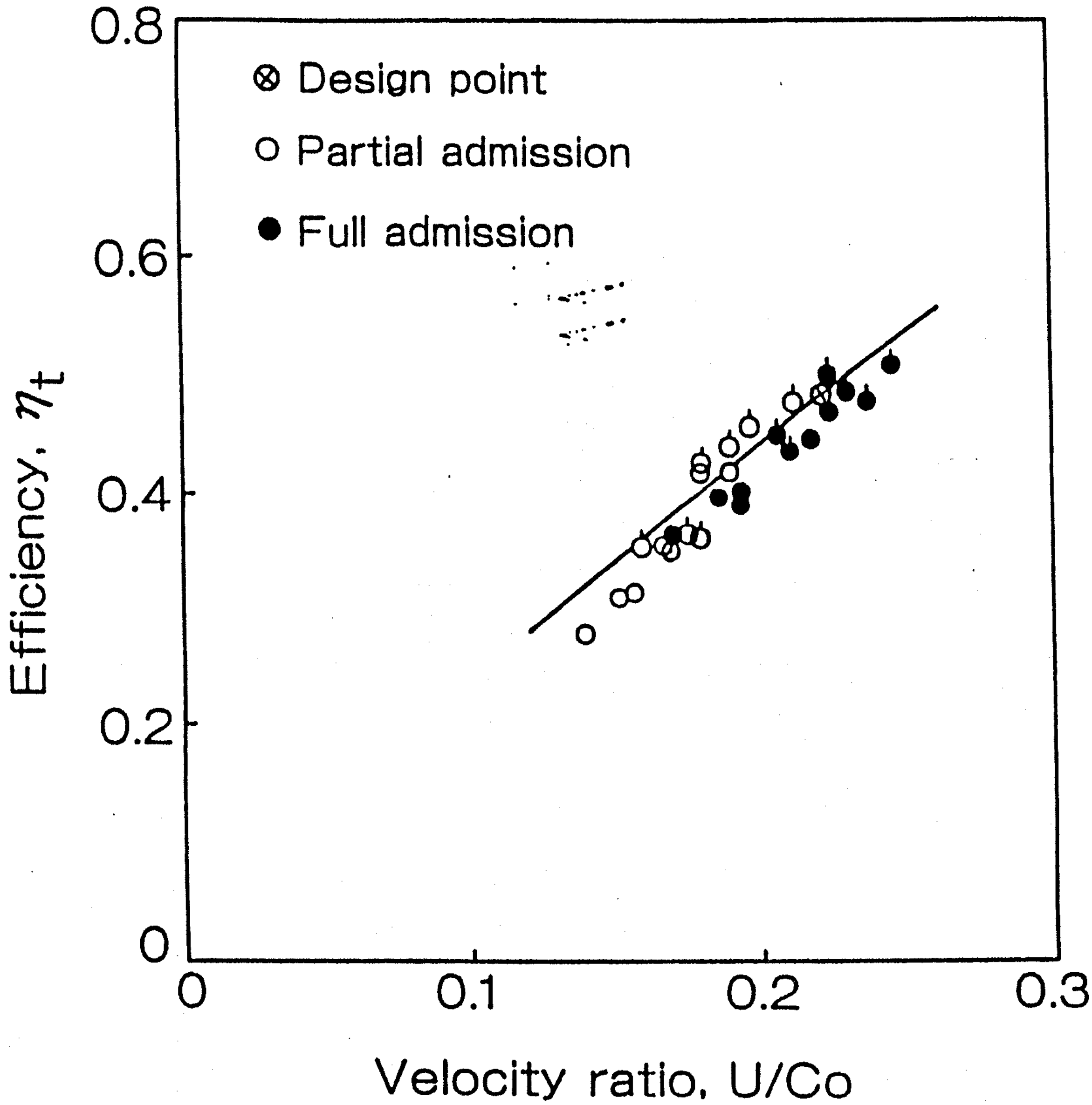

Figure 4.26. Overall Performance of Turbine

HIPEX Turbines

The HIPEX (high-pressure expander cycle) engine has an added heat exchanger in the combustion chamber to increase the turbine inlet temperature to approximately 700R. A schematic of the HIPEX engine cycle is shown in Figure 4.27 with the turbines arranged in series similar to the U.S. OTV expander cycle engine. Operating parameters for the two-stage fuel turbine and single stage LOX turbine are summarized in Table 4.9. The turbine efficiencies were not listed in referenced reports, but were determined from the state point pressures and temperatures listed in Table 4.9 using gaseous hydrogen properties. The turbine diameters and admissions were not listed in the referenced reports so the turbine blade speeds were estimated and the mean diameters were then calculated. Flow rates for the HIPEX turbines are relatively high and the pressure ratios are relatively low, resulting in high velocity ratios and higher efficiencies. No turbine test performance verification results were found in the referenced reports for the HIPEX turbines.

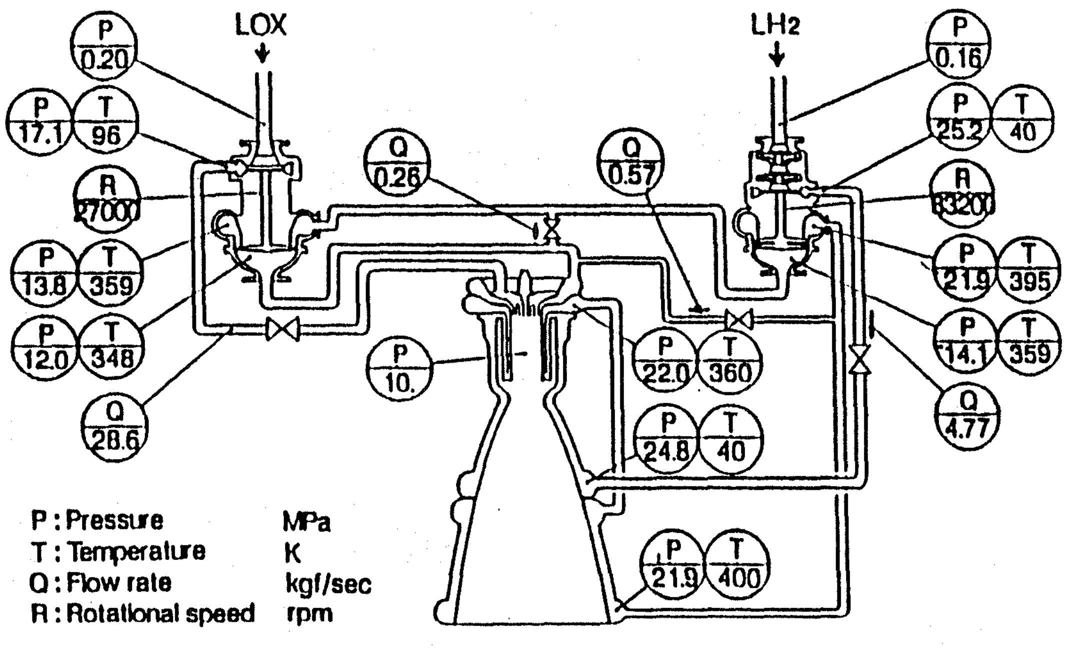

Figure 4.27. System Map of HIPEX Engine

Table 4.9
HIPEX Engine Turbines

TURBINE		FUEL	OXIDIZER
INLET TEMPERATURE	R	711	646
INLET PRESSURE	PSIA	3078	1939
OUTLET PRESSURE	PSIA	1982	1686
OUTLET TEMPERATURE	R	646	626
FLOWRATE	LBM/SEC	9.26	8.69
SHAFT SPEED	RPM	83,200	
	27,000		
PRESSURE RATIO	-	1.55	1.15
STAGING	-	2 STAGE	1
STAGE			
RESULTS FROM "DGNPROP" PROGRAM:			
ISENTROPIC ENTHALPY	BTU/LBM	307.5	93.11
ISENTROPIC VELOCITY	FT/SEC	3924	2160
EFFICIENCY (1)	PERCENT	81 (1)	80 (1)
SHAFT POWER	HP	3263	916
FOR ESTIMATE OF MEAN BLADE SPEED:			
MEAN BLADE SPEED	FT/SEC	1500 (3)	1100
(3)			
ISENTROPIC VELOCITY RATIO	-	0.38 (2)	0.51
MEAN DIAMETER	INCH	4.1	9.3

NOTES: TURBINES BYPASS FLOW = 12 PERCENT OF FUEL FLOW
OXIDIZER TURBINE BYPASS FLOW = 6 PERCENT OF FUEL TURBINE FLOW

(1) DETERMINED FROM TEMPERATURE DROP
(2) STAGE VELOCITY RATIO = 0.54
(3) ESTIMATE

Other Turbine Configurations

Turbine experience prior to the LE-5 and LE-7 turbopumps included a counter-rotating, dual shaft turbine in a single housing with a single nozzle and no stator between the rotors, as reported in Reference 4.36. Figure 4.28 shows the cross-section of the counter-rotating oxygen-hydrogen turbopump (TP-1002) for the 20,000 lb. thrust engine.

Another unique configuration is a three-stage, partial admission tip turbine for the ATREX (Air Turbo Rocket Expander Cycle) engine reported in Reference 4.9. The turbine is mounted on the outside diameter of the compressor fan with carbon-carbon material blades to withstand the 2700R turbine inlet temperature. The tip speed is 1640 feet-per-second, and the efficiency is listed as 50%. A cross-section is shown in Figures 4.29 and 4.30.

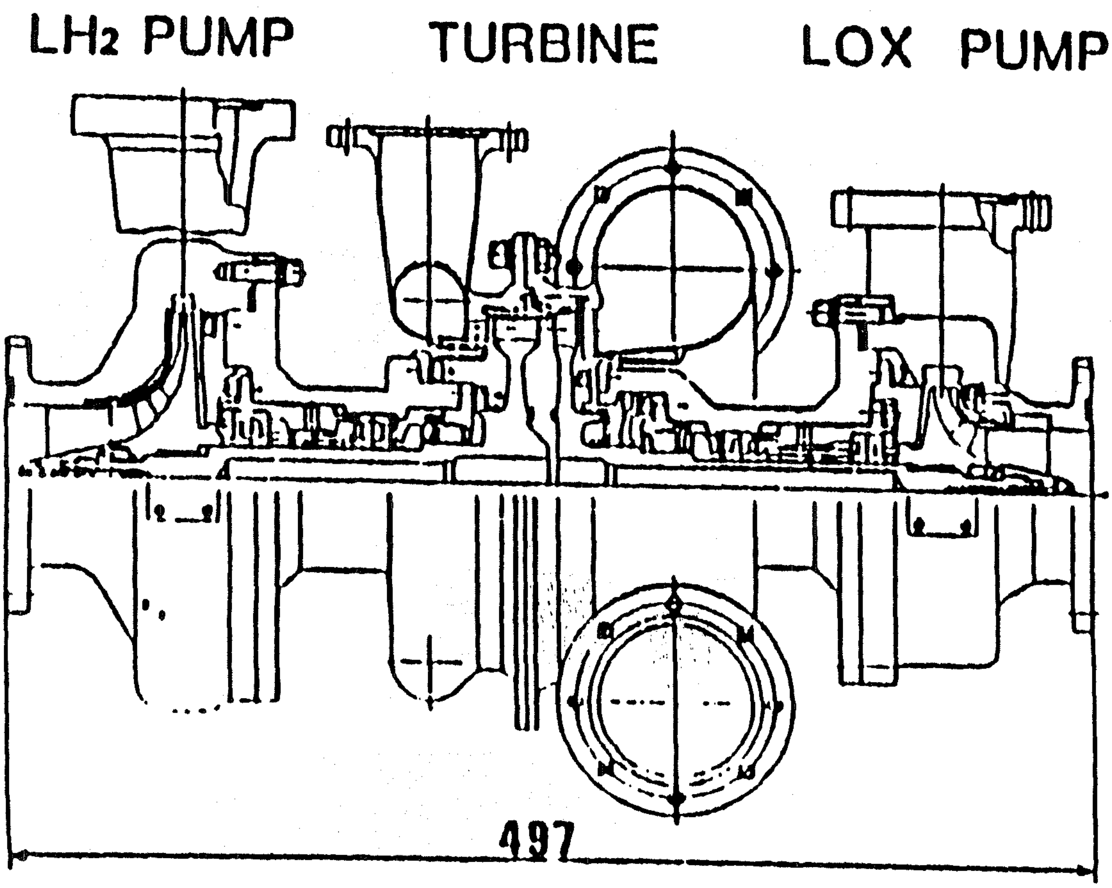

Figure 4.28. TP-1002 Turbopump

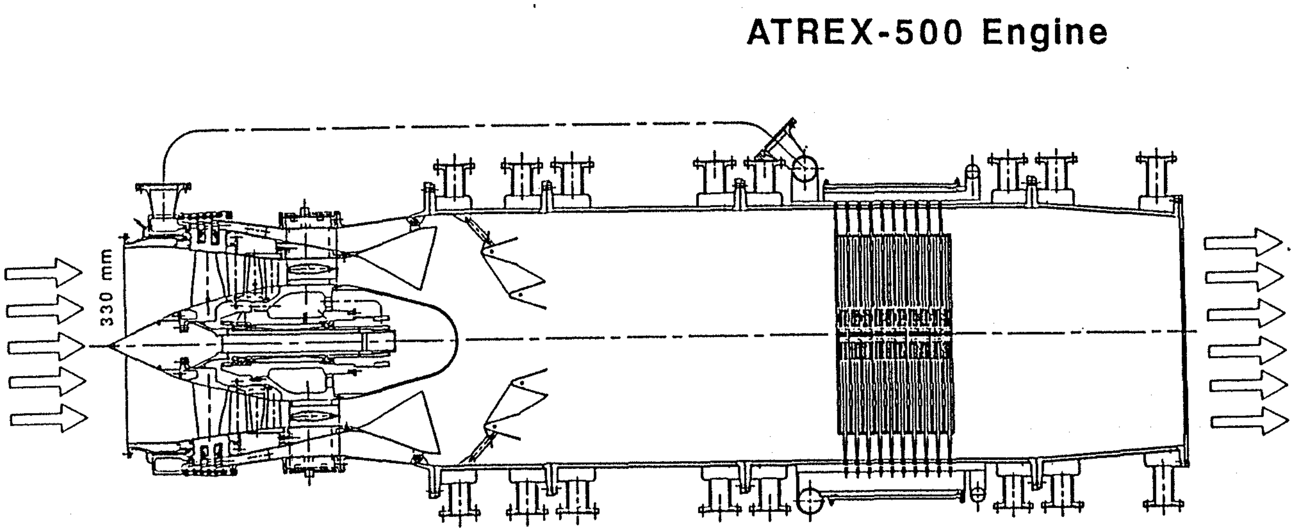

Figure 4.29. Configuration of ATREX-500 Engine

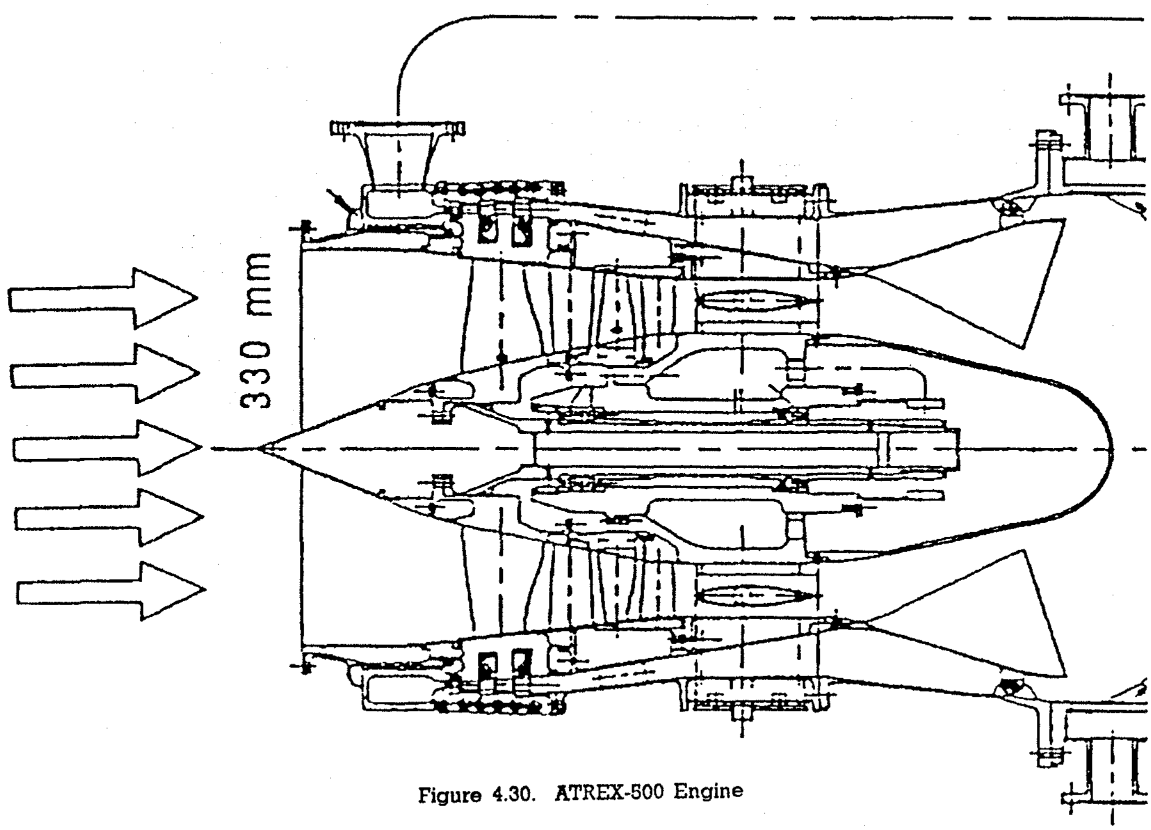

Figure 4.30. ATREX-500 Engine

Turbine Design Methodology

Japanese turbine design and analysis methodology includes three-dimensional flow analysis and performance calculations programs as reported in References 4.22 and 4.23. Two reports, References 4.24 and 4.25, verified the Denton code including analysis and cascade test cases. A research Navier-Stokes code was also reported in Reference 4.26. The LE-7 fuel turbine blading is a three-dimensional design with reaction varying from hub to tip as reported in Reference 4.27. The Japanese turbine design practices are in part based on design reports of the M-1 fuel and LOX turbines, referenced for both the LE-5 turbines and the LE-7 LOX turbines (Refs. 4.3 and 4.5).

The Japanese use extensive testing to verify the turbine designs and analyses. Verification programs include linear and annular cascade tests, cold flow stage tests, and full-size, full-power, and speed tests. References 4.28 and 4.29 reported on an extensive test and analysis study of supersonic impulse two-dimensional blade cascades. Experimental results for radial inflow turbines were reported in References 4.30, 4.31, and 4.32. Unsteady aerodynamics were studied experimentally using shock tubes (Ref. 4.33). Experimental results were reported for air-breathing cooled turbine heat transfer in References 4.34 and 4.35.

Turbine Conclusions

Japanese turbine design methodology and the efficiencies obtained are at a similar level to that of the United States. However, their experience and data base are limited, relying heavily on U.S. published reports, data, and experiences.

The technology for long life rocket engine turbines has not been required or demonstrated which is an area where they are lagging the U.S.

In the area of technological advancements and their application to future turbine designs, the Japanese are on a par with the U.S.

BEARINGS

Rolling Element Bearings

Comparison of Japanese rocket engine turbopump bearing operating conditions and U.S. applications is presented in Table 4.10. Published information defining Japanese rocket engine turbopump rolling element bearing technology indicates that it is based upon, and is currently on a par with, that of the United States in

ORIGIN	VEHICLE	ENGINE	THRUST	PROPELLANT PUMP	DISCHARGE PRESSURE	MASS FLOW	SPEED	BEARING BORE	BEARING dN
			Klbf		psia	lb/sec	rpm	mm	MILLIONS
-	-	-	-	-	-	-	-		-
USA	SATURN	J-2	208	OXYGEN	930	453	8750	60	.5
		2nd, 3rdSTAGE		HYDROGEN	1020	84	27130	60	1.6
USA	SHUTTLE	SSME	426	OXYGEN	4641(7658)	1067(86)	29830	57	1.7
		ORBITER		HYDROGEN	6179	141	36600	45	1.64
JAPAN	LE-5	H-I	21	OXYGEN	745	44	16500	30	.5
				HYDROGEN	800	8	50000	25	1.25
JAPAN	LE-7	H-II	240	OXYGEN	3090(4743)	504(97)	20000	45	.9
				HYDROGEN		4583	46000	40	1.84
JAPAN	UNK	HIPEX	31.4	OXYGEN	2480	63	27000	40	.7
				HYDROGEN	3655	10.5	83200		
JAPAN	UNK	HIPEX-X01	5	OXYGEN	305	UNK	12000	UNK	UNK
				HYDROGEN		UNK	44000	UNK	UNK
JAPAN	SPACE PLANE	ATREX-500	1012	AIR	14.66	15.75	17800	40	.71
				HYDROGEN	UNK	.55			
EUROPE	ARIANE 5	HM-60	210	OXYGEN	1856	444	13230	83	1.1
		1ST STAGE		HYDROGEN	2306	88	35480	45	1.6

NOTES: UNK=UNKNOWN VALUE
( ) DESIGNATES PREBURNER PRESSURE

Table 4.10
Operating Conditions Compared

the field of liquid hydrogen turbopumps in terms of speeds and loads. The Japanese LOX service bearing operating requirements, however, are significantly less severe than those of the SSME and other U.S. LOX applications. Reported test life for the Japanese applications is approximately two hours, roughly equivalent to that achieved in engine service in the United States.

Materials cited in the literature are similar to standard practice for U.S. applications, with 440-C corrosion resistant rings and rolling elements combined with glass fabric/PTFE (Armalon) cages.

Hydrofluoric acid etching of Armalon cages to eliminate the glass fiber abrasion was reported to reduce bearing wear (Ref. 4.37). This technique was evaluated in the United States, but was found to reduce the tensile strength of the Armalon material significantly. Because cage loadings are high in the SSME turbopumps, HF etching has not been adopted.

Gold plating of rings and balls has been utilized in Japan for corrosion resistance. Although not intended as a corrosion inhibitor, gold plating has been tested in the United States as a wear modifier for LOX bearings. Bearings plated in both Japan and the United States were tested at 1.8 million DN with no significant difference in wear life noted between plated and bare 440-C bearings.

Ceramic materials are being considered for both the rings and balls of a bearing for the Japanese spaceplane project (see Table 4.10 and Fig. 4.30). The bearing will operate at approximately 0.7 million DN, with no lubrication, and will be cooled by the combustion products of air and hydrogen. To date, evaluation of ceramic materials for U.S. rocket engine turbopump bearings has consisted of testing bearings with silicon nitride balls and 440C rings in LOX. The use of ceramic rings is considered high-risk due to the low fracture toughness of the material and the serious consequences of a race fracture.

Hydrostatic Bearings

Applications. Most of the Japanese literature dealing with hydrostatic bearings is concerned with the optimum design of gas journal and thrust bearings for machine tools and precision measurement instrumentation. Recently, Mori et al. (Ref. 4.43) investigated hydrostatic bearings for application to cryogenic expansion turbines. However, most works are limited to industrial applications utilizing air, oil, or water as the working fluid. Several of the papers allude to the possibility of applying the technology to cryogenic turbopumps, and IHI has recently reported successfully testing a hydrostatic bearing to 3.4 million DN in hydrogen.

Design. Since most of the Japanese literature concerns the application of hydrostatic bearings to gas machine tools and measuring instruments, the bulk of the design focus is on the minimization of gas consumption and increasing bearing stiffness. With this goal in mind, the Japanese have analyzed and tested many concepts which require very small radial clearances and would, therefore, not be applicable to cryogenic turbopump design. The most recent emphasis (Ref. 4.45) has been on bearings with circumferential grooves, slot-fed bearings, and groove-compensated bearings.

More recently, the Japanese have attempted to apply hydrostatic journal bearing technology to cryogenic expansion turbines (Mori et al., Ref. 4.43). The high rotor speeds have led to whirl instabilities which, in turn, has shifted the design focus to shaft stability. Research has focused on a bearing with circular slot restrictors (Refs. 4.43, 4.46, and 4.47).

Analytical capabilities. The analysis of hydrostatic journal and thrust bearings in Japan is generally accomplished using Reynolds' equations. There is an abundance of papers dealing with corrections and additions to those equations for turbulence and inertia effects and the application of sophisticated solution techniques. For example, Lin et al. (Ref. 4.40) applied the finite element method to solve the time dependent Reynolds' equations for air hydrostatic thrust bearings. However, there are few papers which attempt to depart from the Reynolds' equation technique and apply either film-averaged or CFD formulations of the Navier-Stokes equations. One exception is Mori et al. (Refs. 4.41 and 4.42) which presented a solution of the film-averaged Navier-Stokes equations.

Experimental capabilities. There are many hydrostatic bearing test apparatuses operating in Japan. The bulk of the testers measure only static quantities such as leakage, torque, temperature rise, and load versus deflection. MHI (Taniguchi et al., 1987) has developed a tester capable of extracting rotordynamic coefficients for journal bearings and has demonstrated it utilizing water bearings. This same test rig has been used successfully for testing load bearing seals. No work has been reported for testing hydrostatic bearings (or seals) in cryogenic fluids.

Magnetic Bearings

Magnetic bearing technology has been vigorously developed in Japan for the past ten years. Most of the earlier papers were published in Japan's journals, some in Japanese and some in English. In recent years, the situation has changed, with an increased number of papers appearing in American or English journals. Applications of magnetic bearings for aerospace have been reported indicating that the magnetic bearing technology in Japan is leading in the following areas:

Digital control technology. The most recent development work indicates that flexible shaft (with rotor gyroscopic effects) dynamics have successfully been incorporated into the bearing control algorithm, rendering more predictable and stable rotordynamic characteristics. The control technology development will help resolve some of the problems with the high power supply requirements since an advanced control system design would require lower voltage or current supply. This effort should lead to the design of a compact controller and power supply cabinet suitable for space applications.

Power amplified technology. Japan has several manufacturers producing state-of-the-art compact solid state power amplifiers and is probably the world leader in this field. Aggressive development work could be underway in Japan, gearing for space applications.

Magnetic material technology. Japan has a solid material technology base that can be applied to the development of high performance magnetic materials. Literature data base in this area is lacking.

DYNAMIC SEALS

A comparison of Japanese rocket engine dynamic seal technology to U.S. turbopumps indicates that most of the published Japanese technology is based on the seal designs utilized on the J-2, ASE and SSME programs. The seal systems for separating liquid oxygen and hydrogen rich hot gas are very similar. The face contact seals are similar to the J-2 welded bellows but incorporate improvements in carbon retention and vibration damping. The floating ring seals are similar to the ASE straight bore seals but with the more advanced tapered bore design incorporated. The segmented carbon seals are similar to experimental Rayleigh Step concepts which were successfully tested on a NASA technology program (Ref. 4.1). The static lift-off seal is similar to the SSME fuel turbopump seal. A comparison of the Japanese LE-5 and LE-7 seal operating conditions to the U.S. J-2 and ASE turbopump seals is given in Table 4.11. Seal arrangements for the LE-5 LOX turbopump and the LE-7 LOX turbopump are shown in Figures 4.31 and 4.32, respectively.

Table 4.11
Comparison of LE-5 to J-2 and ASE Turbopumps

TP	FLUID	SEAL TYPE	DIA.IN.	RPM	FPS	PRESSURE PSIA
LE-5	LOX	FACE CONTACT	1,500	16500	108	NA
LE-5	GH_E	PURGED SEGMENTED CARBON RAYLEIGH STEP PADS	1,500	16500	108	NA
LE-5	HOT GAS H_2+H_2O	SEGMENTED CARBON RAYLEIGH STEP PADS	2,400	16500	172	NA
LE-5	LH_2	FACE CONTACT	1,700	50000	370	200
LE-7	LOX	FLOATING RING	1,968	25000	214	710
LE-7	GH_E	PURGED SEGMENTED CARBON RAYLEIGH STEP PADS	3,937	25000	427	87
LE-7	COLD GH_2 -153F	DOUBLE FLOATING RING WITH DRAIN	1,968	25000	214	2420
LE-7	LH_2	LH_2 PURGED DOUBLE FLOATING RING	NA	46130	NA	NA
J2-0	LOX	FACE CONTACT	2,974	8650	112	200
J2-F	LH_2	FACE CONTACT	2,950	28000	360	200
ASE	LOX	FLOATING RING	1,200	77000	402	50
ASE	GH_E	PURGED FLOATING RING	1,200	77000	402	50
ASE	HOT GAS H_2+H_2O	DOUBLE FLOATING RING WITH DRAIN	1,200	77000	402	2470

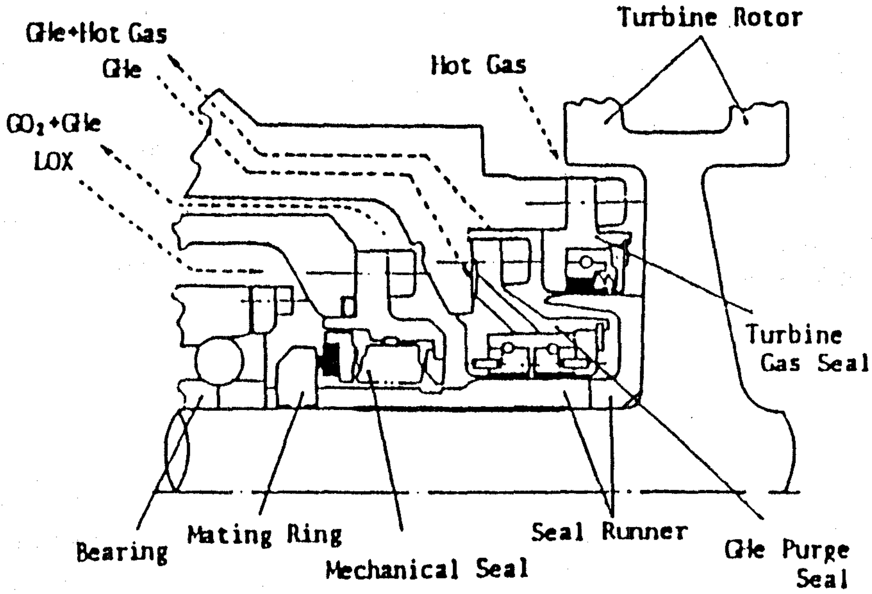

Figure 4.31. Japanese LE-5 LOX Turbopump Shaft Seals

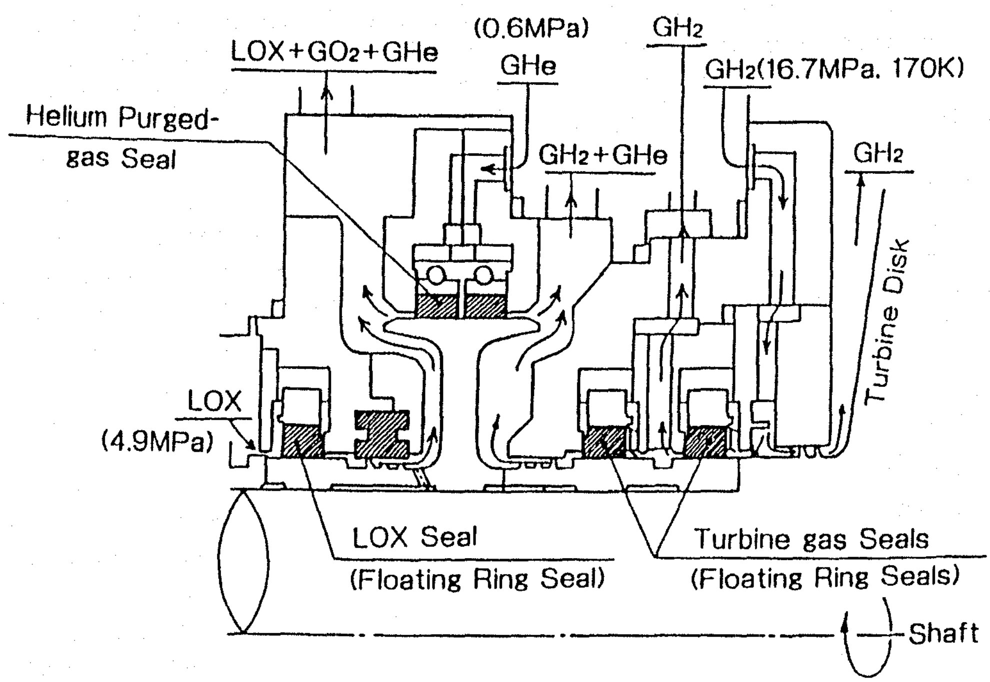

Figure 4.32. Japanese LE-7 LOX Turbopump Shaft seals

Technology Status

The Japanese have developed the segmented carbon Rayleigh Step seal to production status and are experimentally evaluating reverse Rayleigh step face seals to reduce leakage at higher speeds. Japanese publications and applications indicate their dynamic seal analytical and experimental capability is approaching that of the United States.

ROTORDYNAMICS

Experience

Japanese turbopump designs imply a state-of-the-art knowledge of rotordynamics. The LE-7 LH_2 turbopump operates at 46,000 rpm which is above its rotor third critical speed. Early in the development program subsynchronous whirl and synchronous vibration prompted design changes to (1) decrease internal friction by eliminating splines, (2) incorporate soft bearing supports, and (3) add Coulomb (friction) damping to stabilize the rotor.

The LE-7 LO_2 turbopump operates above its first critical speed at 20,000 rpm. Dynamic data from both GN_2 and hot fire tests indicate low-level stable subsynchronous vibration at its first critical frequency that is not detrimental to safe operation.

Although not documented in the available literature, the small high-speed LH_2 turbopump for the HIPEX engine operates at 80,000 rpm, which is probably between its second and third critical speeds. A thorough understanding of rotordynamics, including destabilizing forces and stabilizing damping, is required for successful operation.

Technology

The Japanese rotordynamic technical capability appears to be commensurate with that of the United States and Europe. Active involvement of the Japanese at major rotordynamic conferences around the world ensures a state-of-the-art awareness. Papers presented by Japanese authors include both linear and nonlinear modeling capability. They have also performed analyses and published results on determining the impeller-diffuser coupled influenced coefficients similar to the work performed at Cal Tech in the United States.

MATERIALS

LE-5 Turbopump Materials

LOX turbopump materials. Materials for major components of the LE-5 LOX turbopump include a 304 stainless steel inducer, a carbon filled Teflon inducer liner, and a volute casing made of an aluminum alloy (AC4C). The impeller is a brazed assembly made from an aluminum alloy (6061).

Inconel 718 is used for the turbine stator vanes, manifold and rotor. The rotor is an integrally bladed disk (blisk) fabricated from Inconel 718 by a combination of conventional machining and electrical discharge machining (EDM). Seals are carbon with both tungsten carbide and chrome plated Inconel 718 rub rings. The bearings are 440C stainless steel with a ball separator made from glass fiber filler teflon.

Fuel turbopump materials. The liquid hydrogen pump inducer is machined from A-286 and the volute casing is an A-356 aluminum casting. The impeller is made in two pieces from Ti-5Al-1.5Sn with the front shroud diffusion bonded to the vanes, which are integrally machined with the back shroud. The turbine end is similar to the LOX turbopump with manifold, stator vanes, rotor, nozzle, and shaft made from Inconel 718. Seals and bearings are also similar to the LOX turbopump components.

Materials used in these two turbopumps, in general, are the same materials used in the United States at the time these turbopumps were designed. Serious consideration would have been given in the United States to investment casting the integrally blocked disk (blisk) which supports the perception that investment casting technology in Japan lags that in the United States. The Japanese were, however, successful in developing a diffusion bonding process for the titanium impeller at approximately the same time a similar American program was unsuccessful. Inspectability of the finished part was a major concern in the United States.

LE-7 Engine Turbopumps

LOX turbopump materials. The turbine rotor was initially a machined blisk, but the design was changed to a firtree bladed disk because of thermal stress cracking at the blade/disk intersection during engine cutoff. Directionally solidified MAR-M 247 cast blades were incorporated with the design change.

Fuel turbopump materials. The LH_2 turbine blades were also changed to directionally solidified MAR-M 247 castings to eliminate disk rim thermal cracking.

The impellers are machined from Ti-5Al-2.5Sn with the front shroud diffusion bonded to the vanes which are integrally machined with the back shroud. Materials selected for the LE-7 engine are very similar to materials used in the SSME.

ATREX Turbopump Materials

Integral carbon/carbon blades and disks were selected for the ATREX turbopump. Refractory materials are being considered as a backup in the intermediate development version. All of the materials in the ATREX engine are very advanced; either fiber-reinforced metal matrix composites or carbon/carbon composites including ceramic bearings.

Material Technology Application

Overall, the superalloy and titanium technology in Japan is behind that in the United States in the precision casting area. Although the Japanese are now using directionally solidified blades in the LE-7 turbopumps, this technology has been available commercially for nearly 20 years in the United States. The materials used in LE-5 and LE-7 engines are essentially the same as those used in the United States rocket engines earlier in the same decade.

In the area of advanced materials such as titanium aluminides, monolithic ceramics, ceramic matrix composites, diamond films and high-temperature superconductors, the Japanese are at the forefront of material technology. Application of this technology to the next generation designs such as the ATREX engine will place them ahead of the United States.

STRUCTURES

Analysis Methods

Standard finite element analysis codes and related consulting for the use of these codes are available in Japan. NASTRAN, ANSYS, MARC, and other codes, currently used to structurally analyze rocket engine components in the United States, are all represented and have users in Japan. A literature search in the area of fracture mechanics and probabalistic analysis methodology indicated that the Japanese are doing extensive work in both areas and their technology appears to be equivalent to the United States.

Design Criteria

The turbomachinery for the expendable LE-5 and LE-7 engines was designed based on a safety factor of 1.2 on yield and a factor of 2 for fatigue life. A 5% variation about the design point is applied for engine-to-engine variations to predict maximum operating conditions. No safety factor is applied to the ultimate strength because the yield strength governs, according to information obtained from MHI. The yield criteria, although more conservative than U.S. criteria for ductile materials, would result in less margin for low ductility castings and hydrogen embrittled materials.

The flange criteria dictated by the Japanese government (SSME type flanges with stretch bolts are not acceptable) combined with the 1.2 yield criteria typically result in heavier weight designs than U.S. turbopumps.

FLOW DIAGNOSTICS

Component and Model Testing

Turbomachinery diagnostics concerns measurements made on rotating turbomachinery primarily using advanced laser-based instrumentation. The measurements are not intended to define the overall turbomachine performance per se but rather to measure the flowfield of discrete turbomachinery components as the basis for anchoring CFD codes. Two principal technologies exist for non-intrusive, laser-based measurements of turbopump flowfields, laser Doppler velocimetry (LDV) and laser transit velocimetry (LTV or L2F).

The United States has conducted several test programs using L2F velocimetry to map inlet, discharge, and interblade flowfields in several rocket engine inducers (Refs. 4.53 - 4.55). KHI, which has an L2F velocimeter, has recently begun to make discharge surveys on inducers (Ref. 4.56) after some consultation with U.S. companies. It is quite possible that other Japanese firms have conducted measurements but the results are kept in-house.

The open literature is more forthcoming in the area of impeller testing. The impellers tested are unshrouded, allowing relatively easy optical access. Hagami et al. (Ref. 4.57) used a L2F velocimeter to make interblade measurements in a high-pressure ratio centrifugal impeller. Inui et al. (Ref. 4.58) used a commercially available 2-channel LDV to measure the axial and circumferential velocities near the inside leading edge of an impeller. Hitachi Ltd. uses an L2F velocimeter to study flowfields on high-speed miniature turbocharger compressors (Ref. 4.59). The overall picture in this area is that the Japanese are utilizing state-of-the-art laser-based instrumentation to probe the flowfields of impellers. The research they have reported is on a par with similar work in the United States.

In the area of turbine nozzle and rotor flowfield measurements the literature is again scarce. Senoo (Ref. 4.60) used a single channel LDV to measure the distribution of tangential velocity components in a small Francis-type turbine rotor. In contrast, the United States has numerous turbine research activities utilizing laser velocimetry. These include efforts by NASA (Refs. 4.61 - 4.63) as well as commercial jet engine manufacturers (Refs. 4.64, 4.65).

In summary, the literature suggests that the Japanese are active and on equal terms with the United States in their impeller turbomachinery laser velocimetry research. Published research relating to laser velocimetry measurements of inducers and turbine components is scarce. As mentioned earlier, however, a lack of published data does not necessarily mean no data has been acquired by private firms wishing to retain the knowledge internally.

Trends and Advancements of the State of the Art

The art of laser velocimetry has undergone advances and improvements in the technology on a continuing basis, both in the United States and in Europe and Japan. Fiber optics has made an extensive inroad in the field with commercially available laser velocimeter systems now routinely using fibers in the design of their optical heads. The result has generally been a significant decrease in the volume and mass of these optical heads. As a result, previously unattainable regions of interest in all sorts of internal flow fields can be measured. Access to multiple planes in a flowfield are also possible with relative ease. Fiber optic versions of 1-, 2- and 3-channel LDVs and 2- and 3-channel L2F optical heads are commercially available. These products are not Japanese, however, being primarily developed and sold by American and European firms.

Laser Doppler velocimetry has also seen the advent of new signal processing systems based upon hard-wired fast Fourier transform techniques. The primary advance promoted is the units' capability of extracting useful information from very noisy signals, as well as complete interfacing to and control via PC-class computers. Again, these new technologies were primarily developed and sold by American and European firms with the Japanese role simply that of customer. LDV and L2F technologies have both extensively turned to PC-class computers for interfacing, control, and processing of the velocimeter instruments and the data they acquire.

The Japanese have primarily been consumers of the new technologies rather than the innovators. Although this appears the case for turn-key commercial systems, the Japanese have developed single and multi-point fiber-optic based probe heads

(Refs. 4.66 - 4.69); a beam-scan-type of LDV for simultaneous and continuous measurement of velocity profiles (Ref. 4.70); new systems based upon speckle velocimetry (Refs. 4.71 - 4.74) and systems employing small diode lasers (Refs. 4.75, 4.76). Additional information presented in Chapter 6 suggests Japan may be starting commercial development of advanced laser-based diagnostics.

In conclusion, the Japanese are on equal footing with the United States relative to using laser velocimetry technology for acquiring data in centrifugal impellers. The lack of published literature regarding the use of laser velocimeters on inducers and turbines makes it difficult to conclusively state the same for those turbomachinery components. Other published papers indicate they are keen adaptors and modifiers and inventors of significant related laser velocimetry equipment and systems but not for commercial purposes.

OVERALL TURBOPUMP COMPONENT TECHNOLOGY ASSESSMENT

Japanese state-of-the-art in turbomachinery is rapidly approaching that of the United States. They have accomplished this level in a relatively short period of time by basing the turbopumps designed for the LE-5, LE-7, and HIPEX engines on technology extracted from the J-2, RL-10, M-1 and SSME engine programs. The Japanese approach is to: (1) determine the technology advancement level (2) apply an adequate margin to increase the probability of success, and (3) conduct components testing to verify and anchor the technology before proceeding with the flight design. It is a cooperative effort between universities, ISAS, NAL, NASDA and private industries such as MHI and IHI to minimize duplication of effort and maximize the rate of advancement.

Japan is surpassing the United States in terms of applying technology advancements to turbomachinery for hypersonic propulsion by progressing to the detail design and hardware phase of the ATREX engine.

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Chapter 5

AEROSPACE PLANE RESEARCH ACTIVITIES IN JAPAN

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INTRODUCTION

The present chapter summarizes the ongoing efforts in Japan relative to aerospace plane research and related advanced air-breathing concepts. In terms of propulsion, Japan, like the United States, is pushing technology from its rocket engine experience and from its air-breathing engine experience to give it a foothold in this relatively new area. The primary focus of Japan's efforts is directed towards the low-speed ranges (Mach numbers from 0 to 5), but it is also conducting work in the hypersonic Mach number regimes. Much of its advanced air-breathing propulsion research has potential application to either a spaceplane or to an advanced air-breathing strap-on booster for the H-II. We begin by reviewing Japanese spaceplane activities, then discuss the various propulsion cycles they are considering, and finally describe the component and engine testing in progress.

SPACEPLANE OVERVIEW

The activities relative to the spaceplane are conducted primarily in conjunction with the Institute of Space and Astronautical Science (ISAS) and the National Aerospace Laboratory (NAL). As mentioned previously, both national universities and public and private universities are also involved to some extent.

The current activities at the National Aerospace Laboratory for Hypersonic Flight involve systems studies of a spaceplane for manned space transportation, conceptual study of a hypersonic experimental aircraft and propulsion system, related research activities, and construction of test facilities. A current concept of a single-stage-to-orbit spaceplane was presented by Yamanaka at the AIAA/NASP conference. The aircraft configuration utilizes forebody compression, multipropulsion modules, and employs turbomachinery, ramjet, scramjet, and rocket propulsion. The take-off gross weight is 350 tons. The general appearance of the aircraft is quite similar to the NASP configuration.

Related research activities include the aerodynamic configuration, computational fluid dynamics with real gas effects, fuel studies including slush hydrogen, structures, advanced propulsion concepts such as LACE and SCRAM, and system development scenarios. A typical example of aero configuration research is illustrated in Figure 5.1 for a vehicle at a 50 degree angle of attack and a flight speed of Mach 7. The related shock structure and surface pressure distribution for this configuration were computed using advanced CFD techniques. The TVD (total variational diminishing) Navier-Stokes code illustrates the state of computational technology in Japan. The capability of treating real gas effects also exists in Japan. Figure 5.2 shows constant pressure lines around a re-entry vehicle at Mach 15 with chemically reacting viscous flow. The upper part shows the results for a perfect gas, the lower shows the effects of ionization or dissociation.

Propulsion systems under investigation for the spaceplane include LACE, Turbo/Ram, Air-Turboramjet (ATR), and Supersonic Combustion Ramjet (SCRAM). Current approaches involve use of turbo engines from take-off to Mach 3. A ramjet cycle would then be employed to Mach 6, including such candidates as an air-turboramjet. Supersonic ramjets would be employed from Mach 6 to Mach 12 with the possibility of a LACE cycle or a rocket system at an appropriate point.

ENGINE CYCLES

A number of engine cycles for advanced air-breathing engines are being studied in Japan in analytical investigations and in several experimental programs involving component testing and demonstration engines. These cycles can be used for a vehicle which cruises at velocities in the supersonic flight regime, i.e., for flight Mach numbers up to 5 or 6, or as air-breathing boosters for large launch systems such as the H-II vehicle.

In this section, some of the engine cycles which the Japanese are considering are first described and then the Japanese efforts to develop these engines are discussed in more detail. From this information, we then attempt to assess their intentions.

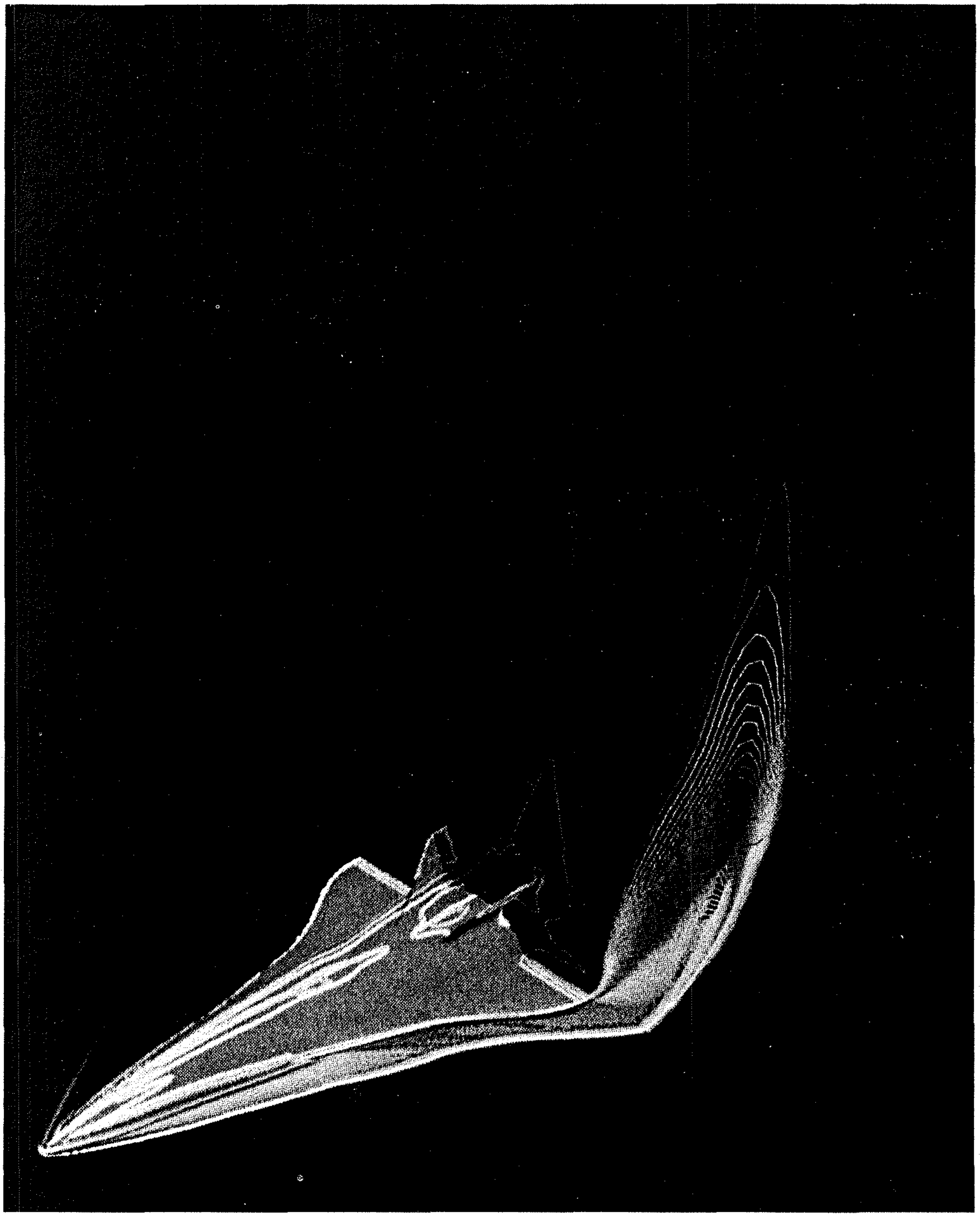


Figure 5.1. Spaceplane

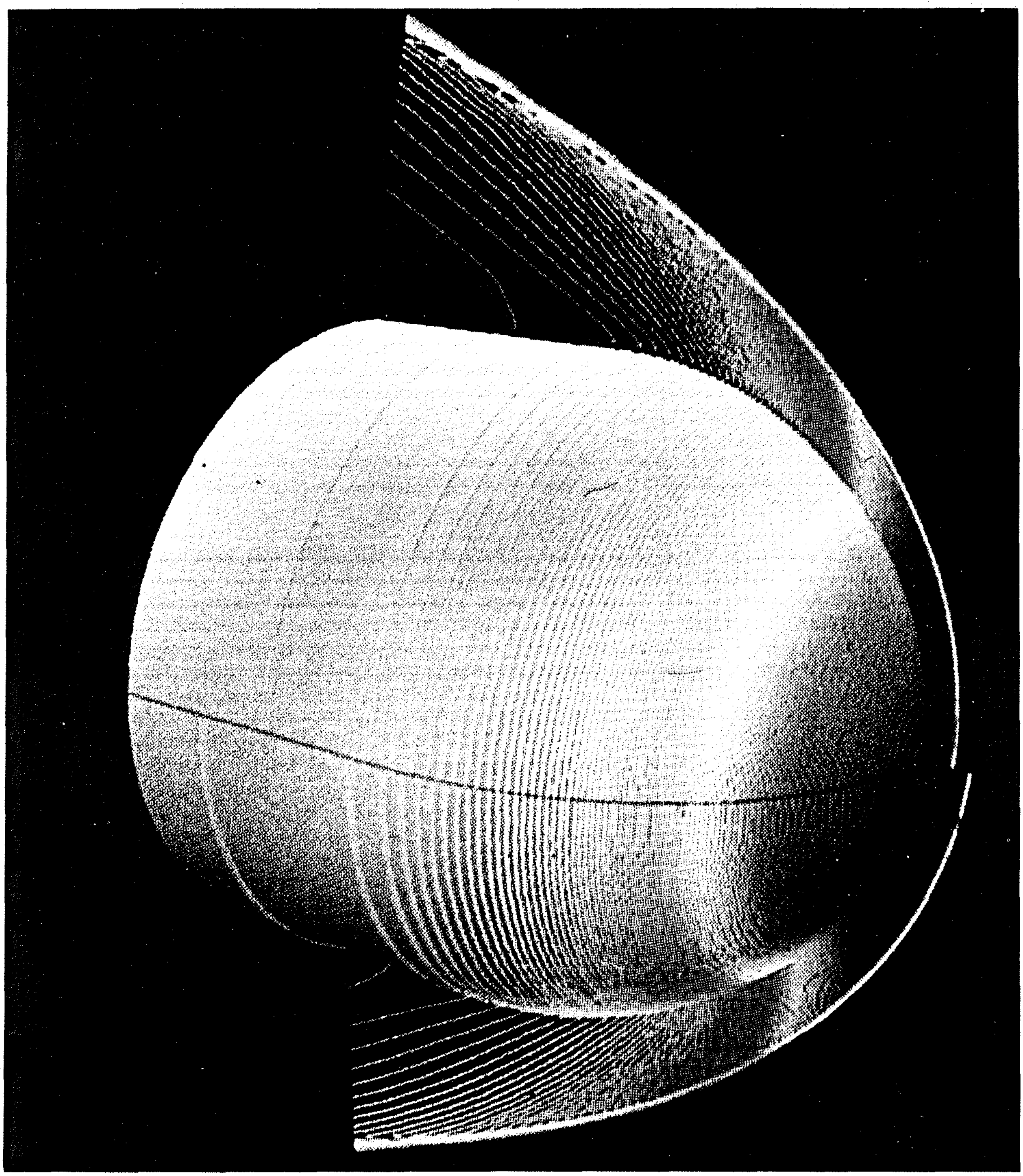


Figure 5.2.

Supersonic Engine Cycles

The wide range of propulsion cycles being considered in Japan is illustrated in Figure 5.3 (Ref. 5.1). These cycles are not unlike those being considered in the United States or Europe. The cycles are first discussed in general, and then the status of hardware buildup and analytic studies in Japan are described.

Turbojet cycle. The standard gas turbine engine and the bypass fan with supersonic through-flow compressor are shown in Figures 5.3 (a) and (b). In these cycles, air enters the engine and is compressed to high pressure in an axial flow compressor which is driven by an axial flow turbine. Fuel is burned in the primary burner located between the compressor and turbine. The hot products of combustion then pass through the turbine to extract the power required to drive the compressor, and are ejected through a nozzle to produce thrust. In the fan engine, some of the air is compressed and then bypasses the core engine. The bypassed air can greatly increase the mass flow through the engine and produce more thrust with a smaller fuel consumption rate.

A primary limitation on the amount of heat which can be added in the primary burner, and hence on the magnitude of the thrust which can be produced per mass flow rate of air, is a gas temperature limitation set by the materials which are used in the turbine blades. Even when relatively cool compressed air drawn from upstream of the primary burner is used to cool the turbine blades, the maximum turbine inlet temperature which the structure allows is about 1800 K or 3300 R. At Mach numbers above 3, this turbine inlet temperature limit leaves little room for burning in the primary burner following the significant temperature rise accompanying compression in the inlet and compressor. Consequently, the ratio of the thrust produced by the engine to the fuel flow rate, called the fuel specific impulse or Isp, will drop off rapidly for flight Mach numbers above 2.5 to 3.0.

This temperature limit also prevents the complete combustion of the oxygen in the airstream in the primary burner, and in some engines this oxygen is burned in an afterburner located downstream of the turbine. The chevrons shown in Figure 5.3 (a), (c), and (d) show schematically the flame stabilization devices used in this afterburner system. Afterburning can be used to increase the thrust of the engine by a factor of two, but greatly increases the thrust-per-mass flow rate of the fuel. Turbojet engines with afterburning are now limited to flight Mach numbers less than about 3.0, due to the temperature limitations discussed above.

Finally, even if the turbine inlet temperature problem could be avoided, the same type of temperature limitation would be reached for the compressor blades themselves at high enough speeds, at which point no cool air would be available to use as a coolant. This limit is reached at a flight Mach numbers in the 4 to 6 range, and air-breathing turbomachines can no longer be used.

ORGANIZATION: NATIONAL AEROSPACE LABORATORY	CONCEPTUAL STUDY OF HYPERSONIC PROPULSION	NAME: T.YAMANAKA
CHART NO: 19-2		DATE: SEPTEMBER 1988

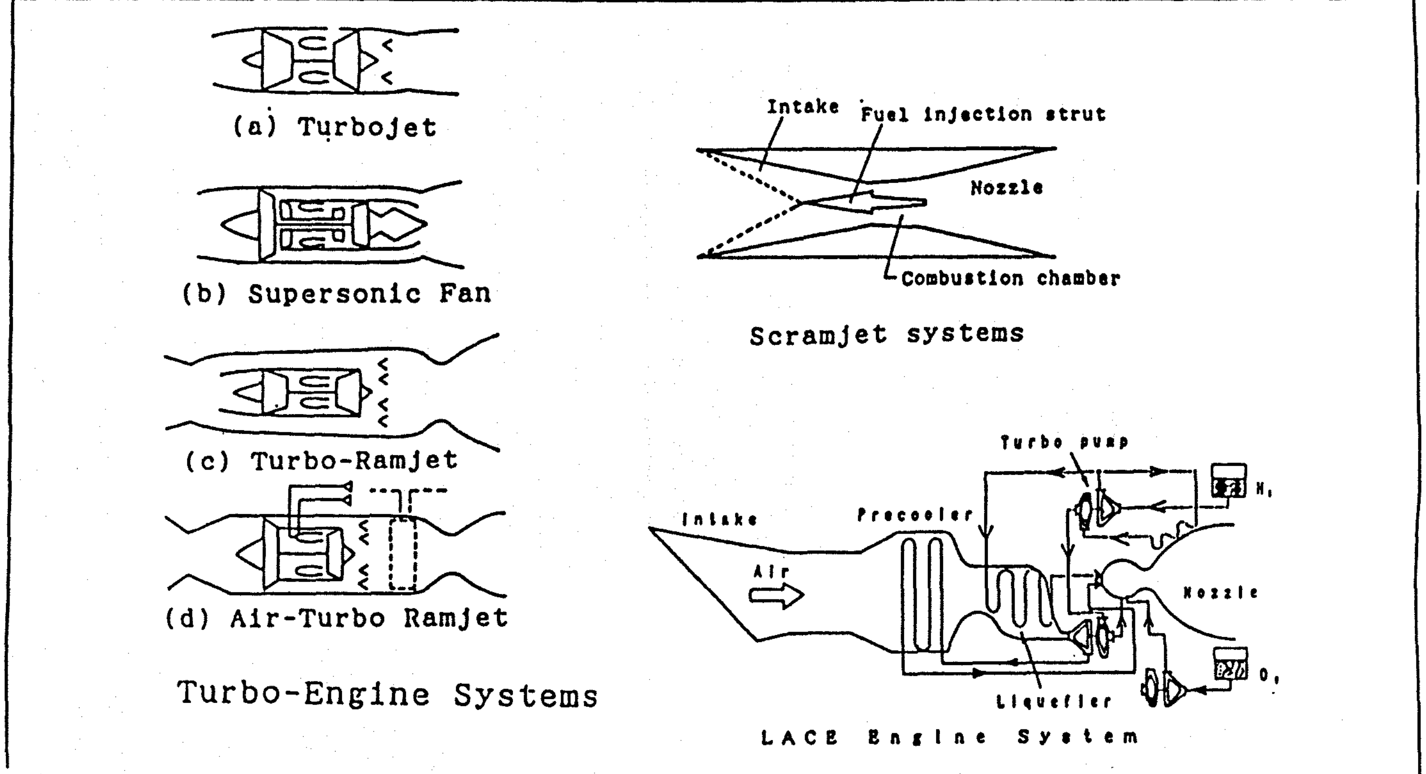


Figure 5.3. Hypersonic Air-breathing Engine Systems

Ramjet engine. A second engine cycle that avoids the temperature limitations discussed above, by avoiding the use of a turbo-compressor and using the ram compression of an inlet to compress the gas, is the ramjet cycle. In this cycle, the air entering the engine is compressed as it is slowed to subsonic speed in the inlet, heat is added by combustion in the burner, and the hot gas is ejected through the nozzle. Although use of the ram compressor limits the compression which can be achieved, the performance of this cycle is useful above Mach 2, and no temperature limitations are placed on the heat addition in the burner, except those fixed by the combustion processes.

Unfortunately, the compression which can be achieved with ram inlet compression is too small to make this system useful at Mach numbers below 2; consequently, a ramjet-powered vehicle must be boosted up to Mach numbers around 2 before the engine can be started. For Mach numbers above 4, the performance of the system begins to decrease due to the inefficiency of the ram compression process, and at Mach numbers above 6, the temperature of the gas entering the combustion chamber is so high that dissociation of products of combustion begins to seriously reduce the performance of the system. Both of these processes cause the performance of the ramjet cycle to drop off above Mach 4.

Turboramjet or combined cycle engines. The turboramjet engine shown in Figure 5.3 (c) is an attempt to combine the best features of the ramjet and turbojet cycles. At low Mach numbers, this engine operates as a conventional bypass fan engine with an afterburner. For Mach numbers above 2.5 to 3.0, most of the air bypasses the rotating machinery of the turbojet and burns in the afterburner. In this high Mach number range, the engine acts as a ramjet engine. An engine of this type will be able to supply useful performance over the range Mach zero to 6 if a lightweight and efficient bypass system can be developed.

The Japanese analysis of this cycle has been carried out for conditions which are appropriate for a transport vehicle in the Mach 3 to 5 range. The cycle is certainly the least complex and least risky cycle to be developed for this flight regime; however, we heard no discussion and saw no evidence that Japan is developing an experimental engine of this type, other than several comments which indicated that Kawasaki Heavy Industries (KHI) was responsible for its development.

Air-turboramjet. This cycle, shown schematically in Figure 5.3 (d) avoids the temperature limitations set by the turbine by using an auxiliary gas flow to drive the turbine that is required to drive the compressor. In early forms of this cycle that were developed at Aerojet in the early 1960s, the auxiliary power was supplied by using a fuel which had a positive heat of decomposition. In current Japanese versions, the compressor power is furnished by liquid hydrogen expansion cycles or a gas generator cycle which will be described later.

In the air-turboramjet cycle, air is compressed in an axial flow fan with a pressure ratio as small as 2:1, is mixed with hydrogen, and then burned. The products of this combustion process are cooled as they pass through a heat exchanger before flowing through the nozzle. In the hydrogen expansion cycle, liquid hydrogen is first pumped to high pressure and is then vaporized in the heat exchanger. The hot hydrogen then passes through a turbine, which drives the air compressor, and is then injected into the combustion chamber where it burns with the air. Because the products of combustion do not pass through the turbine, this cycle avoids the temperature limitation set by turbine blade materials. However, the compression ratio for the air is usually low because it is limited by the power that can be extracted from the fuel flow, and this limits the performance at low flight Mach numbers. Analytical studies discussed later show that this cycle can produce a good performance from Mach zero to values as high as 6.

LACE cycle. The liquid air cycle engine, called the LACE cycle (lower right sketch of Figure 5.3), is a cycle studied in the United States in the 1960s at the Marquardt Company. The cycle involves using the heat sink capability of liquid hydrogen to liquify a flow of air. Liquid hydrogen is pumped to a high pressure in a turbopump, and its heat sink capacity is used in a heat exchanger to liquify a flow of air and heat the hydrogen stream.

In the expander cycle, high-pressure, high-temperature gaseous hydrogen is used to drive turbopumps for both liquid hydrogen and air. The high-pressure hydrogen gas and air are burned in a conventional rocket combustion chamber and then expand through a nozzle to produce thrust. Several variations of this cycle are possible. One uses a gas generator in which air and hydrogen are burned to drive the turbopumps. Analytical studies show that this cycle can produce good performance at Mach numbers from zero up to 8.

Scramjet. The supersonic burning ramjet, or scramjet, system (upper right sketch of Figure 5.3) is a ramjet cycle in which the inlet losses are reduced by allowing the gas to enter the combustion chamber at supersonic speeds. Because the reduction to subsonic speeds can be avoided, the compression losses are greatly reduced. Also, because the gas is still at supersonic speeds, the temperature of the gas entering the combustion chamber will be much lower than in the corresponding subsonic burning ramjet cycle, so that some of the high-temperature combustion problems can be avoided. However, these problems are traded for severe problems involving fuel-air mixing and combustion because residence times of gas within the combustion chamber are in the order of milliseconds. This cycle shows promise of producing useful performance up to Mach numbers of 12 and perhaps much higher.

JAPANESE ENGINE DEVELOPMENT WORK

Our impression is that the Japanese are actively considering two classes of air-breathing engines: their most serious interest appears to be in engines for a Mach 3 to 5 range for a supersonic transport vehicle; their secondary interest is in air-breathing systems which would be used as booster engines to augment rocket engines in large launch systems, or as engines for the low Mach number flight regimes of hypersonic cruise or single-stage-to-orbit vehicles. The Japanese have made a number of studies of air-breathing boost systems which indicate that these systems do allow substantial increases in payloads. They are developing prototypes of two and perhaps three of the systems discussed above. The turboramjet system is apparently being developed by KHI, and the LACE and ATR systems are being developed by MHI and IHI, respectively. However, we were told that experimental development programs related to these systems are currently stopped due to the lack of availability of liquid hydrogen test facilities, which are now completely occupied by the development program for the LE-7 rocket engine.

LACE Cycle

The current Japanese version of this engine has been assigned for development to Mitsubishi Heavy Industries (MHI). Its studies indicate that this system performs well up to Mach 6, and may be useful up to Mach 8.

A sketch of the demonstration engine under development is shown in Figure 5.4 (described in some detail in Ref. 5.2). Air enters the engine through a conventional internal contraction inlet in which the Mach number is reduced to subsonic speeds. The air is then condensed in a liquefier, pumped as a liquid to high pressure, and injected into a rocket-motor-like combustion chamber, where it is burned with gaseous hydrogen fuel. Liquid hydrogen is pumped to high pressure in a turbopump, is then used to liquify the air in the liquefier or heat exchanger, then the gaseous hydrogen is injected into the combustion chamber.

The turbopump is driven by hot gas produced in a gas generator rather than by the hydrogen fuel itself in this first-generation experimental version of the system. The hydrogen pump and the combustion chamber nozzle are taken from the LE-5 rocket engine. This component interchange demonstrates the cross-fertilization between the two programs which appears to be a typical feature of Japanese development programs. A new liquid air pump and the air liquefier are under development.

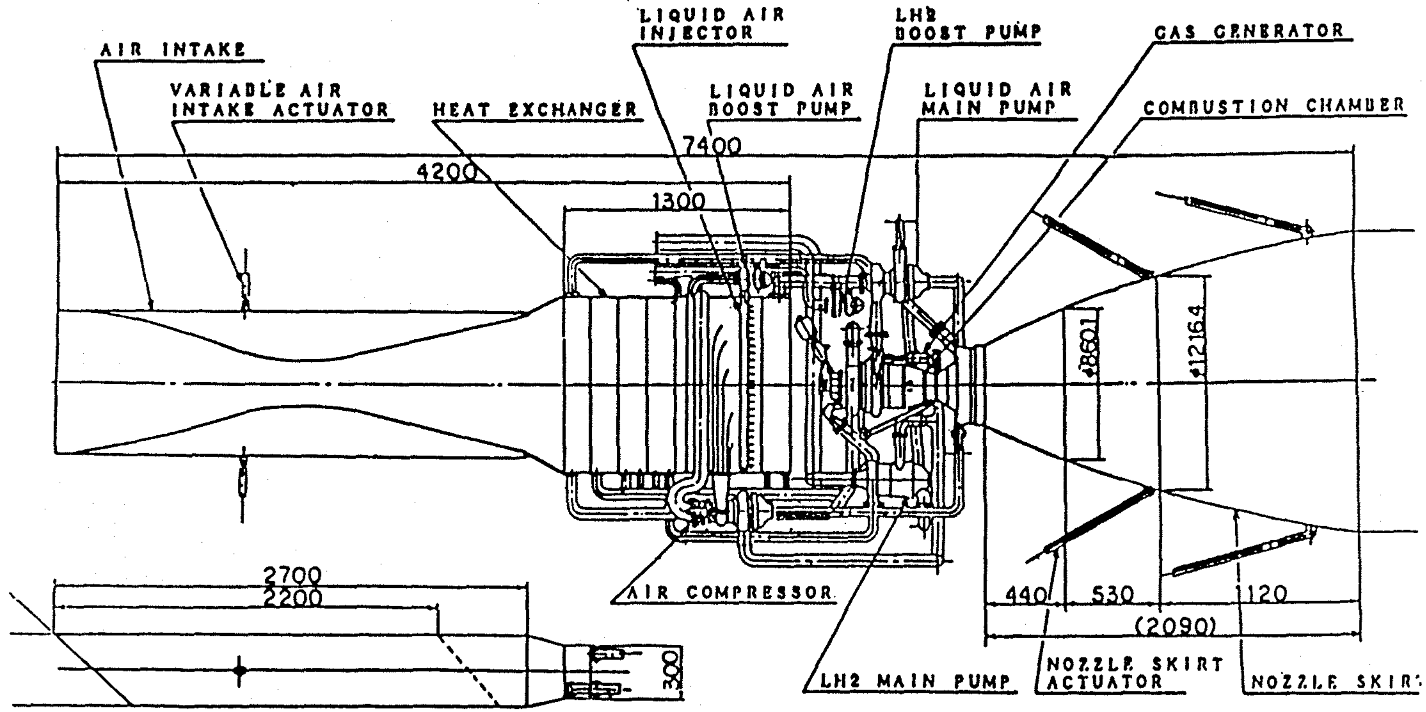

AIR INTAKE
VARIABLE AIR INTAKE ACTUATOR
HEAT EXCHANGER
LIQUID AIR INJECTOR
LIQUID AIR BOOST PUMP
LH2 BOOST PUMP
LIQUID AIR MAIN PUMP
GAS GENERATOR
COMBUSTION CHAMBER
7400
4200
1300
ϕ8601
ϕ12164
2700
2200
300
AIR COMPRESSOR
440
530
1120
(2090)
LH2 MAIN PUMP
NOZZLE SKIRT ACTUATOR
NOZZLE SKIRT

Figure 5.4.

Because the liquid hydrogen gas is used to condense the air, the ratio of the mass flow rates of air to hydrogen will be limited by the thermal properties of the two fluids and by the efficiency of the heat transfer process and cycle details.

This mass flow ratio will fix the performance parameters, such as the Isp, of the rocket engine. In the Japanese system under development, the planned ratio of air to fuel flow rate is about 3.0 for the first tests; the final target is 3.7.

Hardware tests of critical components have been started. In particular, the air liquefier "has been tested twice," and video tapes of the tests have been circulated in Japan--at least to NAL-KRC. As noted earlier, tests have been stopped due to use of liquid hydrogen facilities for LE-7 engine development work.

The liquefier is a critical component in this engine because of the possibility that the heat transfer surfaces can become clogged with water and carbon dioxide which solidify at the temperatures required to liquify air. We were not able to get an explanation of how this problem was solved in the MHI tests, other than the statement that "staged cooling was used." Engineers at NAL-KRC were equally mystified, but suggested that "vibration of the heat exchanger tubes may have prevented icing." (If this is the solution, metal fatigue may be a problem.)

System performance is shown in Figure 5.5 for an engine designed to operate with a flight dynamic pressure of one atmosphere. The curves show the dependence of the fuel specific impulse (Isp) on the ratio of thrust to the weight flow rate of hydrogen fuel, as a function of the flight Mach number. The performance shown here is a function of the state of the liquid hydrogen. Performance at low Mach numbers is considerably enhanced when partially frozen (i.e., slush hydrogen) is used, and is further enhanced when hydrogen is recirculated through the fuel tank. The fuel specific impulse values shown here are about 60% of the fuel specific impulse that could be attained with a good turbojet engine at low Mach numbers, but are higher for the regime above Mach 2.5.

In the earlier work at Marquardt, the possibility that the oxygen could be removed from the air and burned alone with the hydrogen was investigated and was found to increase the performance of a system based on this engine. This modification was not mentioned in connection with the MHI demonstration engine, but has been considered in analytical studies.

Air-turboramjet or Atrex Cycle

The second cycle based on the heat capacity of liquid hydrogen is the air-turborocket system which has been assigned to Ishikawajima-Harima Heavy Industries (IHI) for development. An early version of this engine was developed at Aerojet in the late 1950s. That cycle was based on the decomposition of an acetylenic fuel rather than the evaporation of liquid hydrogen.

In the cycle being developed at IHI, liquid hydrogen is pumped to high pressure and is then heated as it passes through a heat exchanger placed in the combustion chamber. The hot gas produced in this process is then used to drive a turbofan which pressurizes the airstream. A tip turbine attached to the tips of the compressor blades is used to drive the two-stage fan. The hot hydrogen gas is then mixed with the pressurized air and burned in a conventional afterburner system (Ref. 5.3).

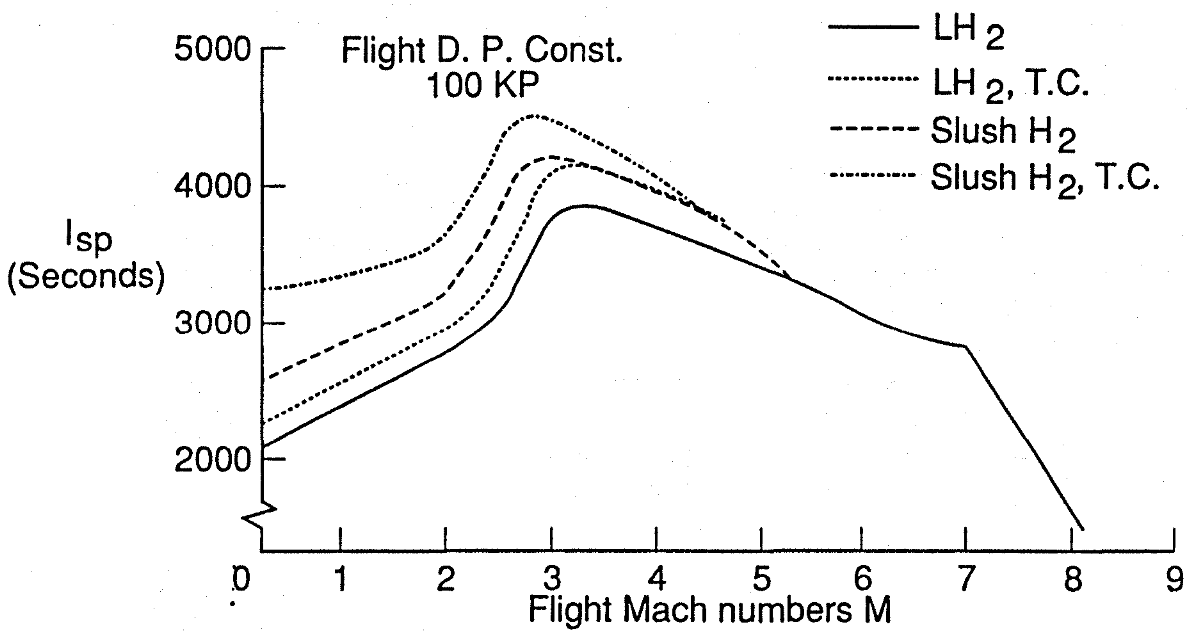


Figure 5.5. Isp vs. Flight Mach Numbers

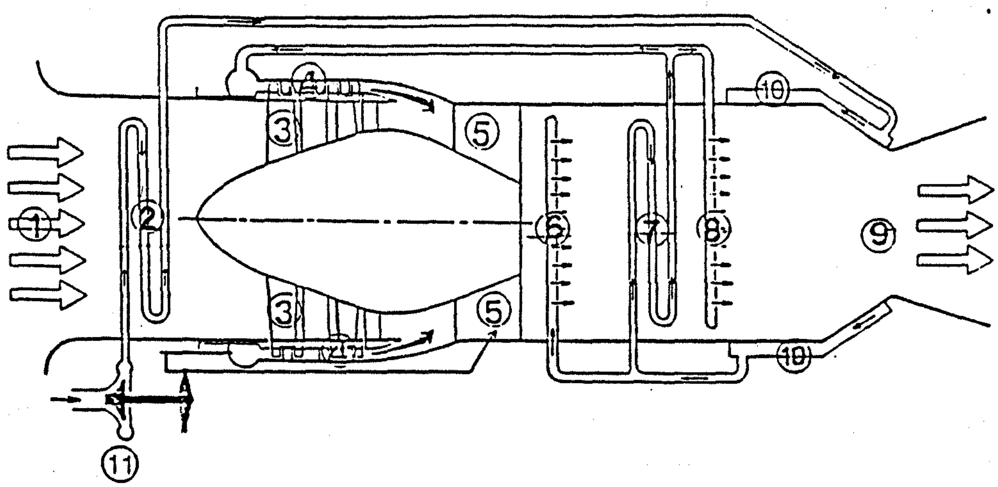
One major advantage of this system is that the power to drive the fan is not dependent on the airflow; thus, the turbine inlet temperature is not directly coupled to the temperature of the products formed by the combustion of the airstream. Because the fan power, and hence the compression ratio, can be controlled by partitioning the fraction of hydrogen which flows through the tip turbine, the work done by the fan can be reduced to zero as the flight Mach number increases, and the engine can make a smooth transition from a turbojet-like cycle to the ramjet cycle.

Analysis at IHI indicates that this system will be competitive with the LACE system or turboramjet up to Mach 5, and will be useful up to Mach 7 or 8. Typical parameters include a fan pressure ratio in the range between 1.7 and 2.5, and a thrust-per-unit area in the range of 15 tons/m^2, with a thrust-to-weight ratio of about 10. The tip turbine development is being used to explore the use of new high-temperature materials.

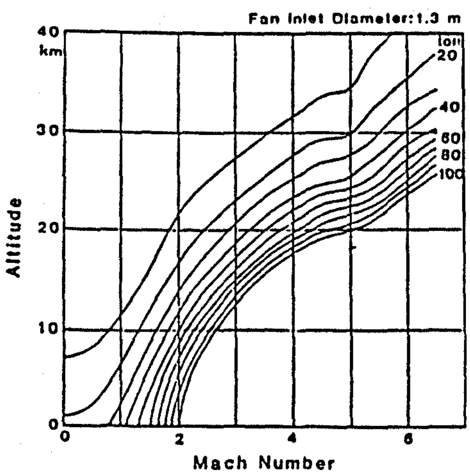
A detailed flow path diagram is given in Figure 5.6. In the cycle shown here, an additional heat exchanger is placed in front of the air compressor, [2] on the sketch. Cooling the inlet air allows the engine to operate at higher Mach numbers before the high air temperatures at the fan exit prevents its use. Also, the reduction in the work required to operate the fan with a given pressure rise (which results from the lower inlet air temperature) may partially offset the pressure losses produced in the heat exchanger. An additional fuel injection port [8] is located downstream of the primary engine heat exchanger [7] to allow control of the temperature of the gas which enters the primary hydrogen heat exchanger.

The performance calculated for a system of this type is also shown on Figure 5.6. Here constant thrust and constant fuel specific impulse contours are given as a function of altitude and Mach number for an engine with an inlet diameter of 1.3m. Thrust levels can be high for low altitudes over the whole Mach number range, and fuel specific impulse values are also high. However, estimates of pressure losses in the heat exchangers used in these calculations were rather low.

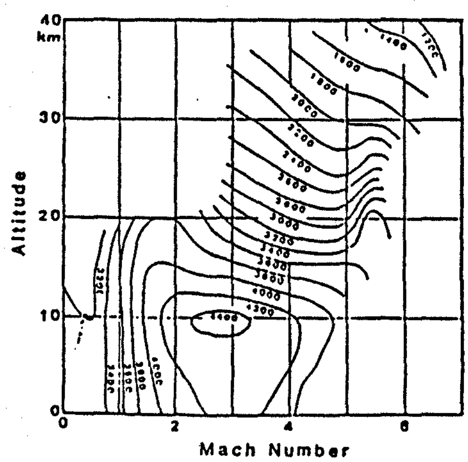
A sketch of the development engine, the ATREX-500, is shown in Figure 5.7, and some details of the properties of this engine are given in Table 5.1. Although the engine case is water cooled, the hydrogen heat exchanger, the compressor-turbine, and the mixer-combustion chamber are operational systems. We were told that hardware has been developed to the point that tests of a complete engine could be carried out when liquid hydrogen test facilities become available.



1: Air Intake 2: Pre Cooler 3: Fan 4: Tip Turbine
5: Mixer 6: Injector 7: Heat Exchanger 8: Injector
9: Nozzle 10: Regenerative Cooling Wall 11: LH2 Pump



Constant Thrust Contours of ATREX



Constant Isp Contours of ATREX

Figure 5.6. ATREX Engine Flow Diagram, Constant Thrust and Isp Contours

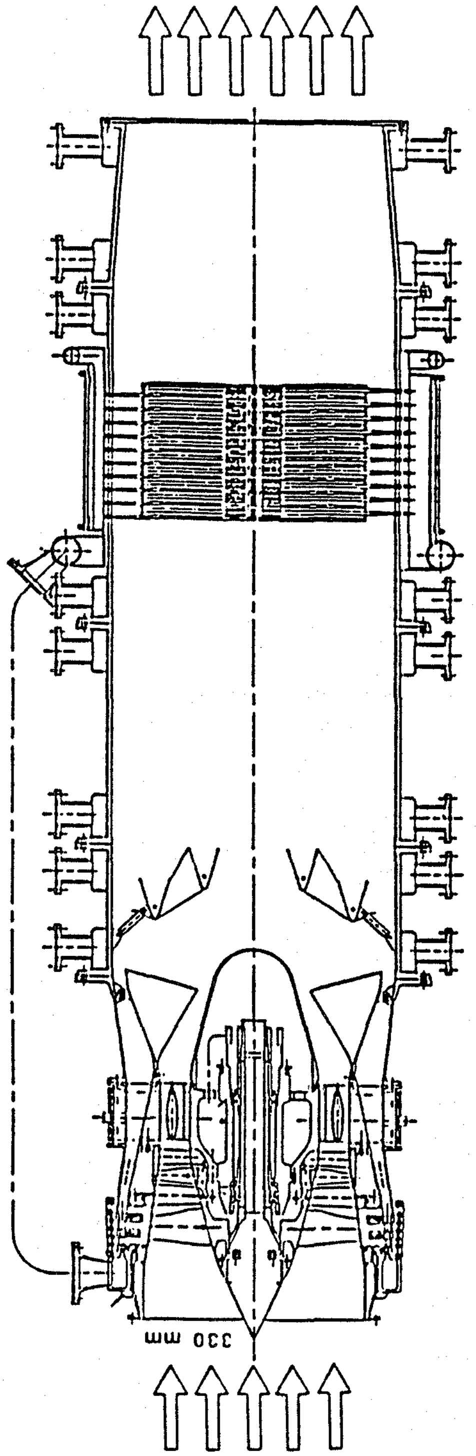


Figure 5.7. Configuration of ATREX-500 Engine

Table 5.1
Design Performance of ATREX-500 Engine

Fuel	LH_2	
Thrust	460	kgf
Specific Impulse	1800	sec
Air Flow Rate	7.16	kg/sec
Fuel Flow Rate	0.25	kg/sec
Equivalence Ratio	1.21	
(Fan)		
Type	Axial Flow	
Stage	2	
Inlet Diameter	0.3	m
Rotational Speed	17800.	RPM
Air Flow Rate	7.16	kg/sec
Pressure Ratio	1.56	
Inlet Total Temp.	288.	K
Inlet Total Press.	1.033	kg/cm^2
Adiabatic Efficiency	83.	%
(Turbine)		
Type	Axial Flow, Tip	
Stage	3	
Rotational Speed	17800.	RPM
Average peripheral Speed	350.	m/sec
Hydrogen Flow Rate	0.25	kg/sec
Pressure ratio	5.0	
Inlet Total Temp.	650.	K
Inlet Total Press.	7.9	kg/cm^2
Adiabatic Efficiency	39.	%
Shaft Power	460.	PS

(Heat Exchanger)		Cold Side	Hot Side
Inlet Total Temp.	K	25.	2380.
Outlet Total Temp.	K	650.	2244.
Flow Rate	kg/s	0.25	7.41
Inlet Total Press.	kg/cm^2	20.	1.394
Outlet Total Press.	kg/cm^2	8.94	1.348

Table 5.2
Pressure Loss in Key Components

Air Intake	-----	%
Fan Strut	0.5	%
Diffuser	1.5	%
Mixer	3.0	%
Flame Holder	4.0	%
Heating by Burn	5.0	%
Heat Exchanger(Hot Side)	3.0	%

The development plans for the engine include an 8-10 year cycle in which the turbine inlet temperature and fan pressure ratio will be gradually increased, from 600 K to 1600 K, and from 1.5 to 2.5, respectively, and the overall cycle performance will be greatly improved.

The engine group at IHI has extensive experience with the production of advanced turbojet and turbofan engines such as the V2500 and the F100, and has designed and is producing the F3 turbojet engine. Hence, its experience in designing turbomachines is considerable.

Supersonic Combustion Ramjet (Scramjet) Engine Cycles

Several individuals indicated that work on hypersonic propulsion systems such as the supersonic combustion ramjet (scramjet) would be delayed until later. At MHI, we were told that the liquid air cycle or the air-turborocket cycle systems are accessible now, whereas the scramjet cycle is not. However, basic technology experiments are being carried out, and this work will be discussed in the following sections.

CYCLE COMPARISONS

Comparisons of the fuel specific impulse produced by the engine cycles discussed here are shown in Figure 5.8, where thrust divided by the weight flow rate of the fuel, Isp in seconds, is plotted versus flight Mach number for a specified trajectory and for three versions of the Air-turboramjet or ATR engine cycles (a LACE engine cycle, a turboramjet cycle, and a scramjet cycle). The performance of the ATR, TRJ, and LACE cycles below Mach 5 are very similar; clearly performance parameters other than the specific impulse must be used to determine the best cycle for this area. Above Mach 6, the fuel specific impulse of all of these drops off rapidly. However, excellent performance can be obtained up to Mach 5 or 6. This result emphasizes again the Japanese concentration on this speed range, rather than on the hypersonic regime in the Mach number range from Mach 6 to 12 or higher in which the scramjet cycles may be useful.

ENGINE TEST FACILITIES

Plans exist for the construction of a supersonic engine development facility at NAL Kakuda. This facility, which is described in detail in a later section, is expected to be operational in 1992 or 1993 and will allow testing of components and complete engine systems at supersonic Mach numbers.

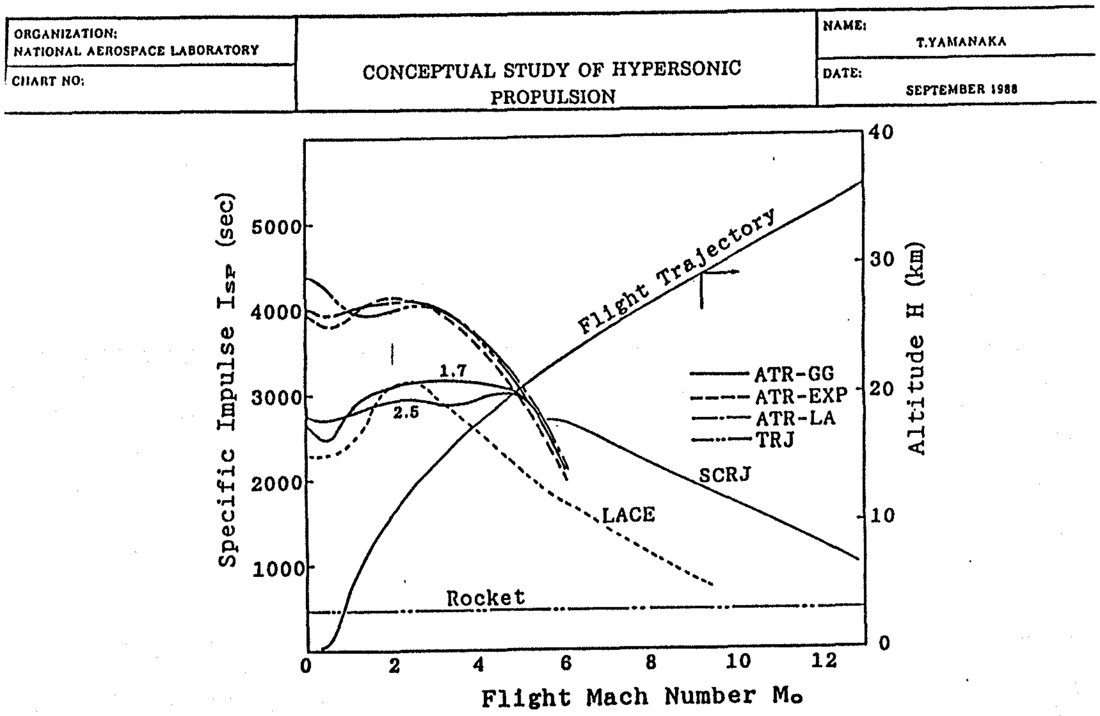


Figure 5.8. Comparison of Isp on the Flight Trajectory

In addition, the Japanese have plans to build a small experimental spaceplane vehicle which will be used as a flying test bed for engine work. The vehicle will be the size of a T-4 trainer and will operate in the range up to Mach 6 and beyond. According to plans described to us, the program would be funded through NAL at a rate of about $357 million over a ten-year period. The vehicle is being designed as a test bed for all of the engines discussed here and could also be used for the low Mach number stages of a scramjet system.

JAPANESE PLANS FOR SUPERSONIC TRANSPORT ENGINE DEVELOPMENT

During our visit, we saw in newspaper reports that the Ministry of International Trade and Industry (MITI) had announced the start of a program to develop an engine for a supersonic transport which will operate in the Mach 0 to 5 range.

The total proposed expenditure was about 28 billion Yen to be spent over eight years (about $25 million/year).

Following the collapse of the Battelle Columbus Center for High-Speed Commercial Flight (Ref. 5.4), MITI announced its intention to seek $5 billion for large-scale supersonic transport test facilities. MITI has also formed a new organization, the New Energy and Industrial Technology Development Organization (NEDO) which will be responsible for a seven-year research program to support engine development for a high-speed civil transport. This vehicle will cruise in the Mach 2.5 to 5.0 range. According to Reference 5.5 this project has been budgeted at about $194 million. The connection between this new organization and the other initiatives ascribed to MITI was not made clear in any of the articles.

In addition, the Society of Japanese Aerospace Companies (SJAC) and MITI support work on combined cycles air-breathing engines and the high-temperature materials required for these engines. The general impression received during our visit, from a review of the technical literature, and from press reports is that the Japanese are primarily interested now in developing engines for the Mach 3 to 6 range, but are accumulating the data base required for the development of higher Mach number engines. They appear now to be more interested in taking a substantial part in the development of hypersonic vehicles as part of a consortium than in carrying out the development alone.

SCRAMJET TECHNOLOGY

Basic Technology programs for scramjet technology are underway at several locations in Japan. The work includes experimental and CFD efforts in inlet configurations and in mixing and combustion technology. An overview of this effort is given below.

Inlets

Scramjet inlet testing is underway at the NAL Kakuda Research Center utilizing a small Mach 4 tunnel. The sidewall compression inlet is mounted on the tunnel sidewall. Boundary layer development along the sidewall simulates forebody boundary layer flow entering the inlet. Because of the relatively small scale of the test hardware, extremely small size instrumentation is required. Questions of scale effects are present. The test program at Kakuda involves a number of parametric variations, including inlet contraction ratio effect, sweep angle of leading edge, and cowl lip angles. This effort appears to be well coordinated with the NAL Chofu computations. It is anticipated that as this work progresses, different configurations will evolve with emphasis on both high-speed and low-speed configurations. Tests at different Mach numbers are planned in the Hypersonic Wind Tunnel at the Chofu test area.

Mixing and Combustion

Optimization Studies. In addition to considerable CFD analyses of scramjets, a number of cycle codes are being developed to predict the performance of the scramjet as a function of Mach number, combustor geometry, area ratio, and mixing schedule. Many of these studies are being used to estimate the engine cooling load and the relationship between fuel cooling requirements and the fuel requirements for combustion (Ref. 5.6). Also, different pump energy sources are being considered (Ref. 5.7). In many cases in the Japanese literature, the mixing schedule from the Langley Hypersonic Propulsion Branch is being used as the technique to model the unknown fuel/air mixing profiles throughout the combustor. (Ref. 5.8) This model does not include all the variables necessary to predict the mixing in a combustor that is to operate over a wide Mach number range, but until a better data base is developed both through experiment and CFD techniques, the Langley model may be a good starting point.

The scramjet optimization code being developed by Y. Tsujikawa is unique in the open literature (Ref. 5.9). He has started the development of an optimization procedure for scramjets based on previous work on the optimization of various gas turbine cycles (Ref. 5.10). Figure 5.9 shows an example of the calculated variation

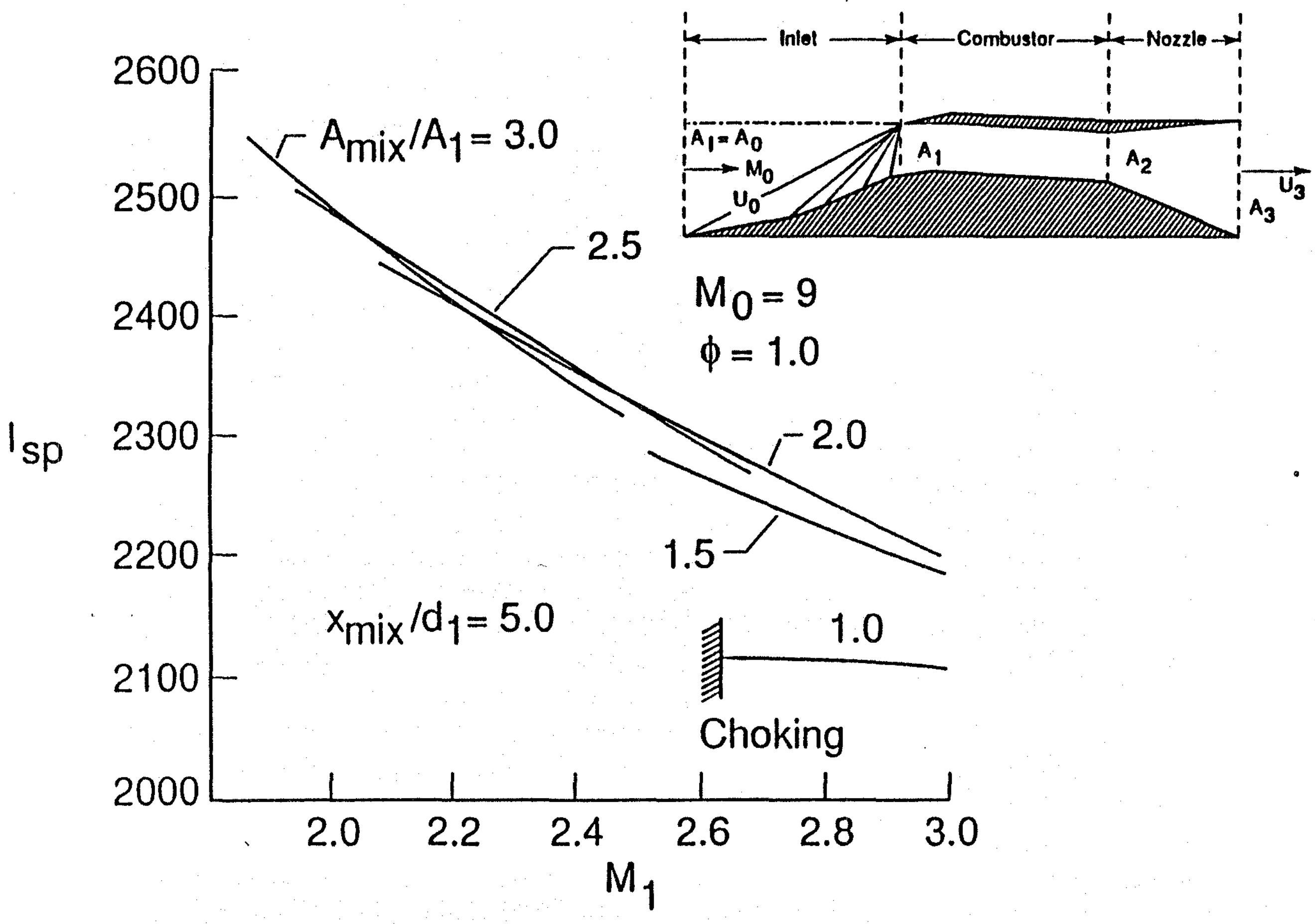


Figure 5.9. Variation of Specific Impulse with Combustor-Inlet Temperature

in scramjet specific impulse for a freestream Mach number of 9 from one of these studies. The inset figure of the modeled engine defines the nomenclature and illustrates the various engine parameters being considered, i.e., inlet, combustor, and nozzle. This work, although conducted at the university, appears to be supported by NAL and will probably continue to mature as the scramjet work in Japan progresses.

The scramjet cycle is complex. Attempts to develop optimization codes to include freestream Mach number, inlet performance, combustor area ratio, fuel air mixing, the coupling between combustor area ratio and finite rate chemical kinetics, and nozzle performance are indeed a challenge. Just the optimization of scarfed nozzles with variable inflow and external flow conditions is the subject of numerous optimization studies. Techniques to prevent base drag resulting from nozzle separation at transonic speeds is still an unresolved issue.

Combustion Research. Supersonic mixing and combustion studies in Japan are, and have been, conducted at NAL Chofu, NAL Kakuda, and the University of Tokyo. Four of the major universities in Japan have aeronautical departments with research related to propulsion and combustion: Kyushu (aeroengine); Kyoto (propulsion); Nagoya (aeroengine, propulsion); Tokyo (space propulsion, rockets, jet propulsion, aeroengine). These universities work with both the national laboratories and the major propulsion industries on research projects of mutual interest. Supersonic combustion research has been conducted at the Komaba campus of the University of Tokyo since 1974 when the Mach 2 pebble bed heater facility was constructed at the Research Center for Advanced Sciences and Technology (RCAST) (originally a part of the ISAS complex at the University).

Studies of supersonic combustion of hydrogen in a vitiated airstream using transverse injection were reported in 1977 by A. Yoshida and H. Tsuji of the University of Tokyo (Ref. 5.11) These tests were conducted in a Mach 1.81 supersonic combustion tunnel fired with city gas-air combustion. The stagnation temperature was varied from 1000 to 1700 K. Autoignition data from these early experiments are shown in Figure 5.10. In Figure 5.10 the static temperature at ignition is plotted as a function of the injected to free jet pressure ratio. These data indicate the usually observed trend of autoignition at about 1000 K static temperature. Autoignition characteristics of a supersonic combustor are also affected by the local boundary layer thickness in the vicinity of the fuel injectors.

The University of Tokyo is also currently conducting Mach 2 direct connect tests to study both perpendicular and parallel mixing and combustion. The parameters to be investigated include molecular weight, wall interference, interaction among 2-D slots and 3-D holes, ignition/autoignition, dual mode combustion, and fuel type. Some of these data are to be used to validate CFD codes. In addition to total

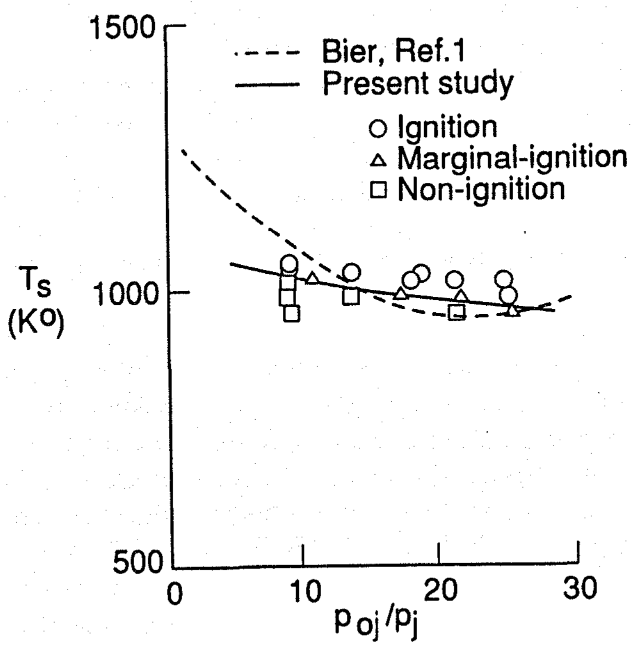


Figure 5.10. Ignition Temperature

pressure, wall pressures, and species concentrations by probe techniques, advanced diagnostics techniques are being considered for current and future experiments. Professor Fujiwara at the University of Nagoya is also coordinating a study of "hypersonic reactive flows in scramjet engines." This project (FY 1989-1991) involves about 20 professors from the aerodynamics group. A joint institute of the University of Tokyo and the National Aerospace Laboratory was formed in 1989 to conduct joint research on "high-speed air-breathing engine technology." The institute has eight professors and about twenty NAL staff members participating. The group will concentrate its research on (1) performance of air-breathing engines, (2) fundamental component technology for turbo-engines, (3) fundamental component technology for ram/scramjet engines, and (4) measurement technology of internal flow of engines.

Combustion research has also been reported recently by the NAL Kakuda group in cooperation with IHI and the University of Tokai (Ref. 5.12). The vitiated heater at NAL was used to study autoignition of hydrogen from different wall injection stations downstream of a rearward facing step. A rectangular, uncooled combustor was used and fuel was injected from both walls, from one wall, and from single and multiple jets. The duct divergence was 3.4 degrees on the two side walls. Multiple orifices appeared to enhance autoignition. Ignition limit curves were derived from the data. This same combustion apparatus has been used to evaluate the utility of using a plasma torch for ignition and flame holding in scramjets (Ref. 5.13). This work which reported combustion augmentation due to the plasma was based on photographs at the exit of the combustor. Pressure rise in a longer duct would have been a better technique to assess any combustion augmentation. The plasma does appear to be an effective ignitor and pilot at conditions where flameholding is difficult. Plans to attempt plasma torch flameholding for base burning and other difficult areas were not discussed. Earlier work reported in Combustion and Flame in 1981 (Ref. 5.14) indicated that the idea of plasma ignition and plasma-stabilized combustion have been considered for some time.

The NAL Kaduda laboratories have been conducting mixing and combustion studies for several years using their 100-mm diameter vitiated test facility shown in Figure 5.11. The scramjet combustor has a rearward facing step with ports for injection both up and downstream of the step. Fuel can also be injected from a number of ports parallel or normal to the combustor walls. In this configuration, autoignition of hydrogen occurred at about 1500 degrees Kelvin total temperature. The step provided the required flame holding. Tests have been conducted at simulated flight Mach numbers 4.4 and 6.7 (Refs. 5.15, 5.16). (CFD results from the parallel injection tests are presented elsewhere in this report.) Basic studies of supersonic turbulent mixing layers have also been investigated at NAL Kakuda. The thrust of this research was to determine the effects of compressibility on the mean properties of supersonic mixing layers. Tests were conducted with two streams at Mach 2.3 and up to Mach 1.35, or with a velocity ratio up to 0.74. The results showed higher compressibility effect for larger velocity difference. Pitot-static probes and high-quality Schlieren were used to study the effects (Ref. 5.17, 5.18).

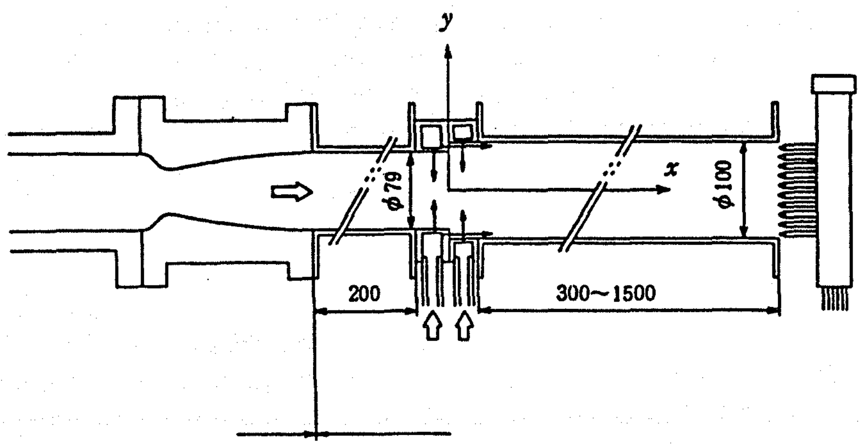


Figure 5.11. NAL - Combination Facility

Recently, basic mixing studies in Mach 2 flow have been conducted at NAL Chofu with the aid of nonintrusive diagnostics (Schlieren and CARS--Coherent Antistokes Raman Spectroscopy--to be discussed later). The objective of this research is to develop a basic understanding of the mixing process with transverse injection into a supersonic stream, and to establish a data base that can be used to compare with the CFD codes that have been developed for compressible flow. This understanding will lead to design methods for stable and efficient combustors. Also, the validated numerical codes can then be used to predict performance beyond the limits of test facilities (Mach 8 for nonpulse facilities) (Ref. 5.19).

Nagoya University has an active combustion research program and in recent times the resources of this group have been applied to high-speed combustion issues. The university has an assortment of shock tube facilities for both flow physics and chemical kinetics studies. The university has a program to understand the use of silane as an ignition and combustion aid for hydrogen-fueled scramjets. As part of this program, researchers at the university have constructed a reaction kinetics mechanism, evaluated the Langley kinetic mechanism, and conducted tests in their kinetics shock tunnel to derive a new mechanism. OH emission diagnostics at 3064 Angstroms were used to measure ignition delay times. This work was motivated by the need to shorten the ignition delay and reaction length, and thus the resultant combustor length (Ref. 5.20).

The effects of conducting scramjet tests in the exhaust of air-hydrogen rockets was addressed in response to questions posed by the JTEC panel. NAL is aware of the possible effects of testing in vitiated air and has designed its new scramjet test facility to accommodate studying vitiation effects. The storage heater that is to be used as the preheater to the vitiation heater at Mach 8 conditions can be operated alone at about Mach 6 conditions. These results will be compared with Mach 6 tests with just the vitiated heater operating. This is an ideal arrangement in that the flow quality, turbulence levels, and exact combustor configuration will be used for both vitiated and nonvitiated tests. These tests will show the effects of water and the resultant OH on supersonic diffusion flame combustion, but the effects of NO generated at the 2% level in arc heaters will not be addressed in these tests. Both NO and OH have been shown to affect ignition delay times in premixed combustion studies and in opposed jet burner studies of diffusion flames. The question remaining to be resolved is, "What is the order of the effect of vitiation in supersonic mixing/kinetically controlled combustion?"

Japanese researchers have published several articles on the effects of electric fields on combustion. In a 1984 paper, researchers at Nagoya University explored the use of electrons and ions on the ignition delay and ignition limits of oxygen/hydrogen combustion systems (Ref. 5.21). This research was developed for the LO_2/LH_2 rocket interest but could be equally applied to hydrogen-fueled scramjet combustion. The orders of magnitude reduction in induction time with increasing concentration of electrons is shown in Figure 5.12. Although there might be some ambiguity in the kinetics model used, the drastic reduction in induction time is impressive. These calculations may explain part of the effectiveness of the plasma torch as a high-speed ignitor. The observed effects with the plasma torch far exceed just the thermal energy injected into the flow.

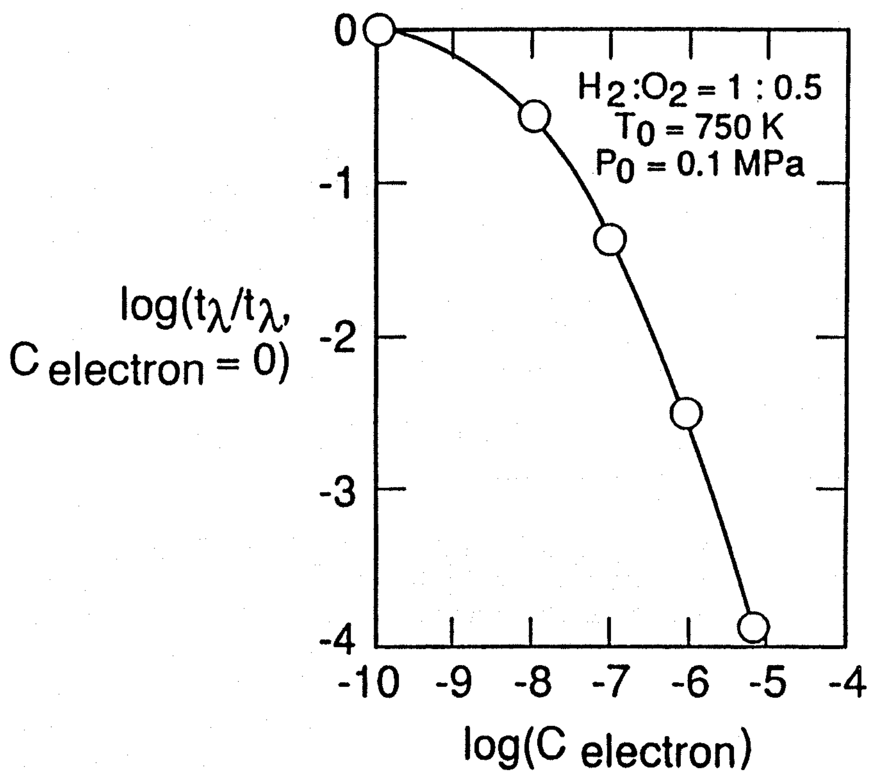


Figure 5.12. Logarithmic Fraction of Induction Time with Electron Participation Normalized by That Without Electron

Shock tube studies of the elementary reaction rate constants of H_2/O_2 were recently reported by researchers at the Nagaoka University of Technology (Ref. 5.22). Shock tube tests were conducted over a temperature range from 1700 to 2600 degrees Kelvin. Rate constant for $H + O_2 = O + OH$ was monitored using 130 nm atomic oxygen absorption. The rate constant for the initiation reaction ($H_2 + O_2 = H + HO_2$) was also evaluated from the shock tube induction period. This work extended the temperature range of these data and illustrates the Japanese commitment to understanding hydrogen combustion from the most basic research through the design and development of rocket engines and scramjet combustors.

Facilities

The major scramjet combustion test facilities are located at NAL Kakuda Research, NAL Chofu and the Center University of Tokyo Komaba Campus. The current facilities are being used to conduct direct-connect tests to understand supersonic combustion and mixing. The injection Mach number is on the order of 2 for most of the tests (Ref. 5.23). Enthalpy levels approaching 7 have been simulated in the facilities. Most of the ignition, combustion, mixing, and plasma torch studies conducted in these facilities have been to generate a data base for the design of efficient combustors and to validate CFD mixing and combustion codes. NAL has recently awarded a contract to Kobe Steel and Fluidyne Engineering of Minneapolis for the construction of a new and larger scramjet test facility to be built next to the rocket test facility at Kakuda. The facility is scheduled for completion in 1992. The major characteristics of the facility are shown in Figure 5.13, and a general layout of the facility is shown in Figure 5.14. In addition to these test facilities, there are a number of aerodynamic and combustion shock tunnels in Japan. In fact, a national shock tunnel conference was being held in Tokyo while the JTEC team was visiting. There is also interest in building a Stalker Tube, i.e., a free-piston-driven shock tube for combustion tests. A pilot tunnel is being constructed at Tohoku University to develop free piston driver technology. Some of the larger parts for the RHYFL pulse facility at Rocketdyne were manufactured in Japan due to the size of the casting required.

FUELS DEVELOPMENT

The following article, entitled "Rocket Fuel Helps Launch UK Satellite," was copied from a Japanese newspaper during the on-site tour:

> A Japanese improved liquid fuel was used in successfully launching a British satellite with an American rocket in August (1989), Japan's state run space agency said Monday. Nippon Oil Co. developed the rocket fuel, called RJ-1 on commission by the National Space Development Agency (NASDA). The company improved the prototype of the RJ-1 fuel originally developed in the United States to use in launching Japan's N-1 and N-2 rockets. The Japanese fuel was used in the successful Aug. 27 launch of a British broadcasting satellite from Cape Canaveral Space Center in Florida. The U.S. suspended production of its own rocket fuel because oil fields ran out of crude oil suitable for the fuel after 1980.--NASDA, finding its use limited to commercial and peaceful purposes, permitted Nippon Oil to export the fuel in late 1987.

This article is included to indicate that Japan has the resources to develop advanced fuels for rockets and will soon be moving into hydrogen production for the new series of hydrogen-fueled rockets and aerospace plane research.

NAL CONSTRUCTING MACH 4 TO 8 SCRAMJET TEST FACILITY WITH 51 X 51 CM NOZZLE

COMBINATION STORAGE AND / OR VITIATED HEATER

MACH #	8	6 *	4
ALT. km	35	25	20
TEMP. K	2835	1651	884
MDOT kg/s	10	30	46

* TEST CONDITIONS WITH EITHER THE VITIATED HEATER OR THE STORAGE HEATER

EJECTOR SYSTEM IS PART OF THE ROCKET TEST FACILITY

HYDROGEN STORAGE AREA AVAILABLE

CONTRACT WITH FLUIDYNE AND KOBE STEEL

Figure 5.13. Facilities

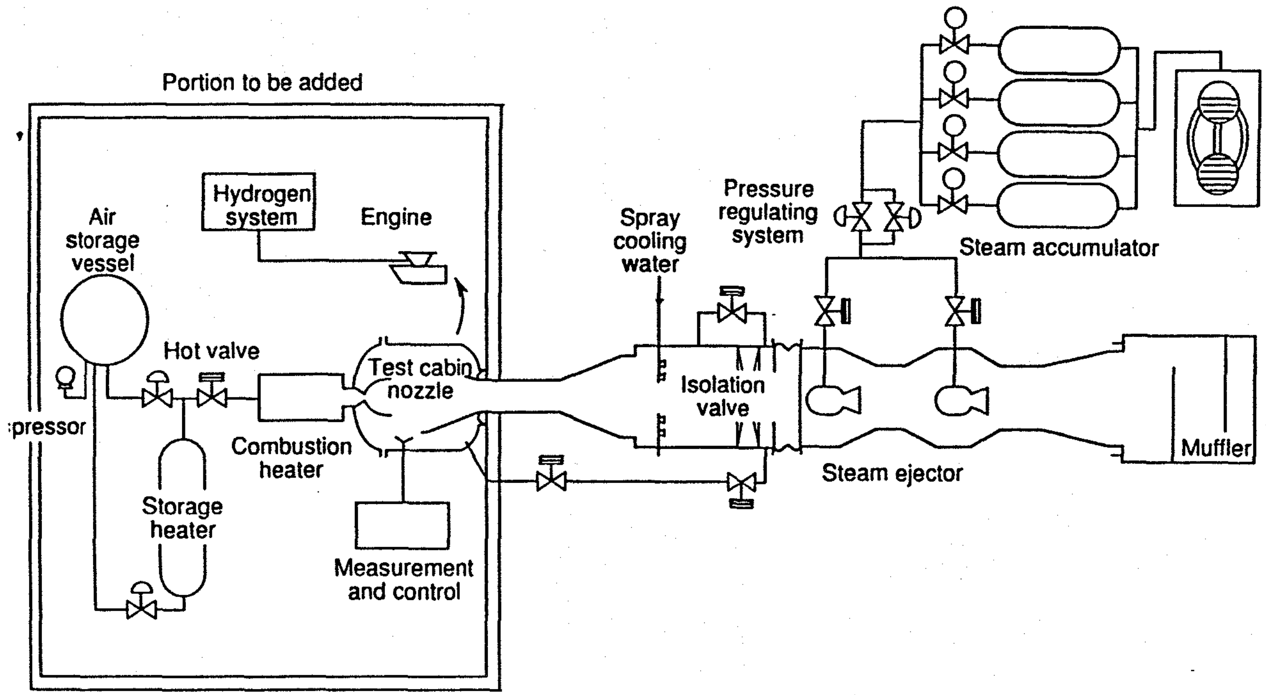


Figure 5.14. General Concept of the Scramjet Engine Test Facility

In 1987 the general chemical manufacturer Showa Denko established a joint venture, Pacific Liquid Hydrogen, with the French (Relikido) for further development and promotion of liquid hydrogen business (Ref. 5.24). The initial plant has a capacity of 7400 kiloliters/year and uses hydrogen surplus from the company's ethylene production. The company expects the use of liquid hydrogen to increase as the H-II rocket development continues, as work on advanced propulsion is initiated for supersonic aircraft, and in the long term, as technology matures to allow hydrogen-fueled automobiles. (The high purity of liquid hydrogen also has many applications in the semiconductor business.)

In supporting research, a group at Waseda University under professor E. Kikuchi has developed a palladium/porous glass composite film that has outstanding hydrogen selective permeability characteristics (Ref. 5.25). This film will allow hydrogen to be removed from ongoing reactions, and offers another technique for production of high-purity hydrogen for rocket or semiconductor applications. The technique has been used to produce hydrogen from water and carbon using the water gas process at low temperatures.

Work has also been reported on the development of a coal gasification plant for the production of both gaseous and liquid hydrogen. In this article the manufacture, storage, transportation, and end use of hydrogen was evaluated. Storage and transportation issues considered tank trucks, pipelines, liquid hydrogen storage, and hybrid storage in metals. The end uses considered were fuel cells, industrial power, fuel for homes, fuel for the chemical industry, fuel for rockets and aircraft, and fuel for automobiles.

A pilot plant that can process 20 tons per day of coal is being built. The schedule indicated research operation of the pilot plant is to begin by 1990. The pilot plant is to be used to generate design data for a demonstration plant that could convert about 500 tons of coal per day to hydrogen (Ref. 5.26).

The Nippon Oil and Fats Co. and Nippon Petroleum Co. have developed two new hydrocarbon liquid rocket fuels which have high density and thus reduced tank size for the same stored energy. The density and heating values of the new fuels, HDF-1 and HDF-3, are shown in Figure 5.15 (Ref. 5.27). Note that the heating value of HDF-1 is about equal to that of JP-10, but the density is higher. These fuels are considered economical replacements of RJ1J currently used in the H-I rocket. The new fuels have undergone combustion tests at NAL and are reported to have acceptable combustor performance. High-density hydrocarbon fuels have advantages over hydrogen fuels for volume limited vehicles that require storable fuels.

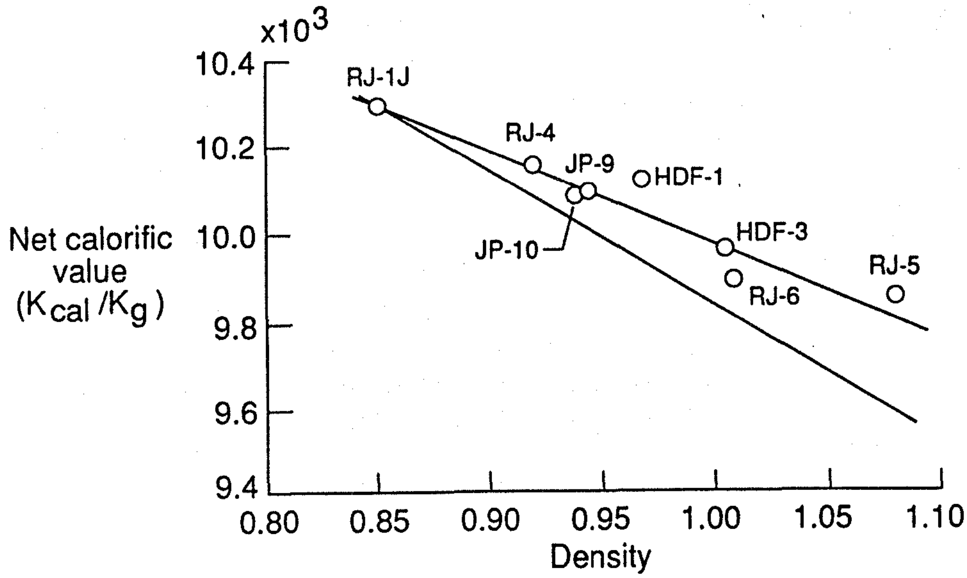


Figure 5.15. Net Calorific Value and Density

Methane has also been considered as a fuel for the scramjet in the Mach 6 to 12 flight regime. Although hydrogen is the most promising fuel for scramjets, methane was examined due to its higher density. The specific impulse of the methane fuel scramjet was about 1000, compared to 2500 sec. for hydrogen at Mach 6. The differences in the thrust coefficients of the two fuels were found to decrease as the Mach 12 flight speed was approached. Some type of dual fuel system that uses methane or other hydrocarbons at the lower speed and hydrogen at higher speeds, where not only the energy but also the cooling of the hydrogen is required, may turn out to be the optimum system. This work was the result of a joint program between NAL and Nissan (Ref. 5.28).

Combustion Diagnostics

From a brief review of the Japanese literature dealing with diagnostics, it is obvious that there is a lot of expertise in the country related to lasers, optics, electro-optics, and the interfacing of computers with these systems to yield high-quality data displayed in a usable format. The emphasis on the post processing analysis and display of flow physics data is very impressive. Many of the post processing techniques that are being applied to the display of CFD data are also being used for experimental data. Also, a number of different data reduction techniques and optical arrangements are being used to increase the yield of LDV data. NAL has conducted simultaneous CARS and LDV measurements in the combustion facility using TiO_2 seed particles and common optics for the CARS and LDV lasers to insure common sample volume. Simultaneous temperature and velocity histograms from 1000 realizations were compared as a function of equivalence ratio at various locations behind a bluff body flame holder (Refs. 5.29, 5.30).

The CARS system is unique in that it uses a low-resolution monochrometer and two photomultiplier tubes to record different portions of the CARS spectra. The system allows temperature measurements to be recorded and displayed in near real time. The accuracy of the fast system is as good as the multichannel analyzer system. (See Figure 5.16, and Refs. 5.31, 5.32.) The other advantage of the photomultiplier system is the lower cost of the detector system and the computer required to invert the data. NAL has done an outstanding job of constructing the system and applying the system in the harsh environment associated with combustion testing. The system is now being used to study supersonic mixing and the validation of CFD codes for supersonic mixing studies.

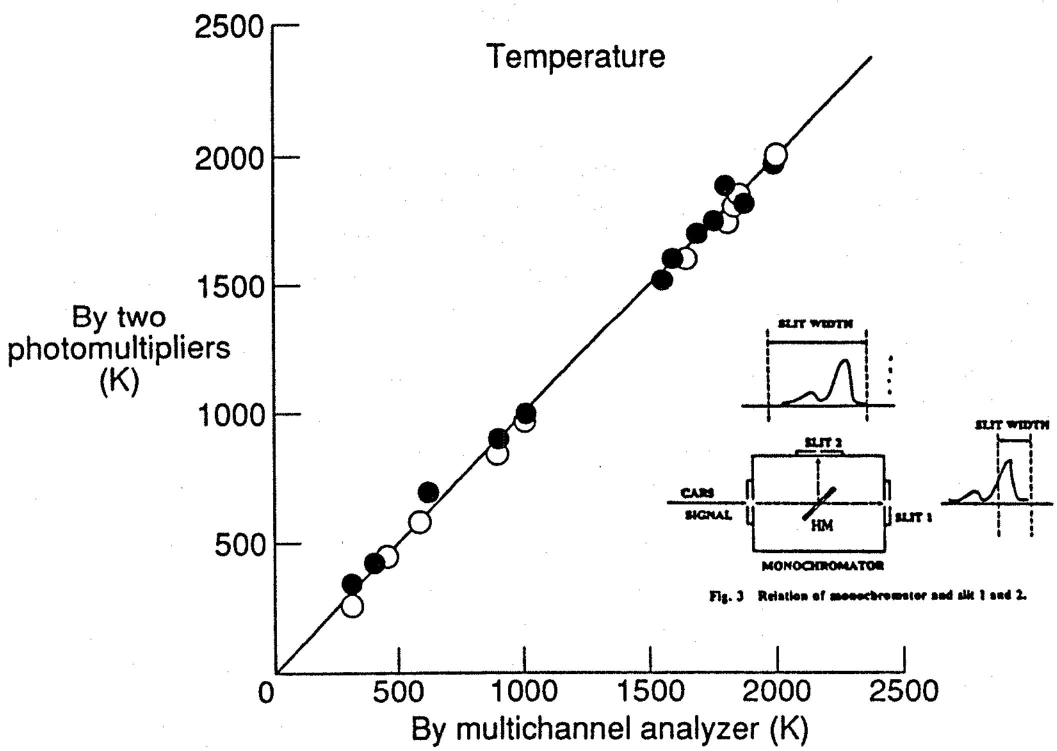


Figure 5.16. Comparison of Two Temperature Measurements

In addition to the combustion-related diagnostics cited here as examples of the fine work being done in the diagnostics area, the electronic industry in Japan is developing tunable diode lasers that have application for "line-of-sight" measurements using oxygen and water vapor and possibly OH. Both NASA and the Air Force are developing diagnostics systems to measure density, inlet mass-capture, and combustion efficiency using Japanese-produced diode lasers. The present lasers are temperature- and current-tuned to the proper wavelength and can be scanned in wavelength by current modulation.

In a recent report the Central Research Institute of Mitsubishi Electric announced the development of a two-wavelength diode laser (Refs. 5.33, 5.34). The wavelength can be switched from one color to the other just by changing the injection current. Although the laser is being developed as a light source for optical communication, optical information, etc., the two colors would be advantageous in a diagnostics mode for tuning off and on an absorption line at high rates to measure absorption strength and thus concentration. Mitsubishi Electric has also developed a surface emitting diode laser that has less beam divergence than conventional lasers and thus would allow more energy to be directed in a tighter path for diagnostics applications (Ref. 5.35). This laser is also easily modulated in wavelength and operates at room temperature.

Mitres Mining and Smelting has joined the Institute of Chemical and Physical Research to develop single crystal growth technology. They have developed a laser tunable from 0.7 to 1.0 microns (Ref. 5.36). This wavelength range can be used for the detection of several species important to combustion research (Ref. 5.35). Also, Hitachi, Ltd. has developed a semiconductor laser that has a natural frequency of 30 gigahertz, five times the current frequency (Ref. 5.37). This high-frequency modulation of the diode laser would increase the time resolution for diagnostics measurements as well as allow signals to be transmitted at about 10 gigabits per second. Asukaru has developed a new doubling crystal with transmission down to 0.2 microns (Ref. 5.38). The new crystal will allow doubling down to lower wavelengths with higher efficiency because the new crystal contains no water to absorb the short wavelengths. The crystal is potassium titanate phosphate (KTP) and will probably replace potassium deuterium phosphate (KDP) as a doubling crystal for YAG lasers.

These last several examples of Japanese technology are cited to illustrate the depth that Japanese diagnostics has as support when attempting to apply advanced diagnostics techniques to combustion and scramjet research. (Our diode lasers come from Japan.)

APPLICATIONS OF COMPUTATIONAL FLUID DYNAMICS TO PROPULSION

The present section presents the Computational Fluid Dynamics (CFD) activity related to aerospace plane research and advanced air-breathing propulsion. The growth in computational capability in Japan has been very impressive. The national laboratories possess the computing power to perform state-of-the-art computations on aircraft and propulsion systems. Further enhancements are already underway to improve their performance and storage capability. A wide variety of problems have been worked using three-dimensional viscous solution techniques. CFD propulsion research is being conducted at the National Aerospace Laboratories (NAL) and at several universities. The unique facilities of the Institute for Computational Fluid Dynamics are worthy of special note. In this chapter, we shall review the computer facilities and then present examples of computations for the Hope vehicle as well as for scramjet inlet/combustors.

CFD Capability at NAL

The development of supercomputers has provided the stimulus for Japanese CFD activities. Emphasis is placed on the analysis of physical phenomena for both aerodynamics and propulsion. The computations provide confidence and reinforce the reliability of experimental results as well as providing results where no test data exists (Ref. 5.39).

The National Aerospace Laboratory (NAL) has been active in numerical simulation research for at least 25 years. Figure 5.17 illustrates the growth in NAL's computer performance. The computer facility is referred to as the Numerical Simulator System (Ref. 5.40). This system is configured as shown in Figure 5.18. With the installation of its first supercomputer in 1977, NAL has achieved a number of excellent results in the field of CFD.

In 1987, a loosely coupled multi-processor system with two supercomputers serving as local processors was installed. It incorporates a FACOM VP-400, a FACOM M-780, and a FACOM VP-200. The speed and storages are respectively 1.14 GFLOPS and 1 GB; 45 MIPS and 128 MB; and 570 MFLOPS and 128 MB. The three computers are linked together into a large-scale external storage system having both magnetic discs (100 GB) and magneto-optic discs (180 GB).

The FACOM M-780 multi-processor is linked through a high-speed network to various research areas. The networking consists of an Ethernet type LAN interface (10 Mbit/s), a channel interface (3 MByte/s), and a CCP interface (64 Kbit/s). Connections currently are to a central I/O station; a structures research subsystem having a NEC ACOS 610; a flight simulator having an NDG Eclipse MV-2000; a space science/AI research subsystem having a DEC VAX-8200; the Chofu remote

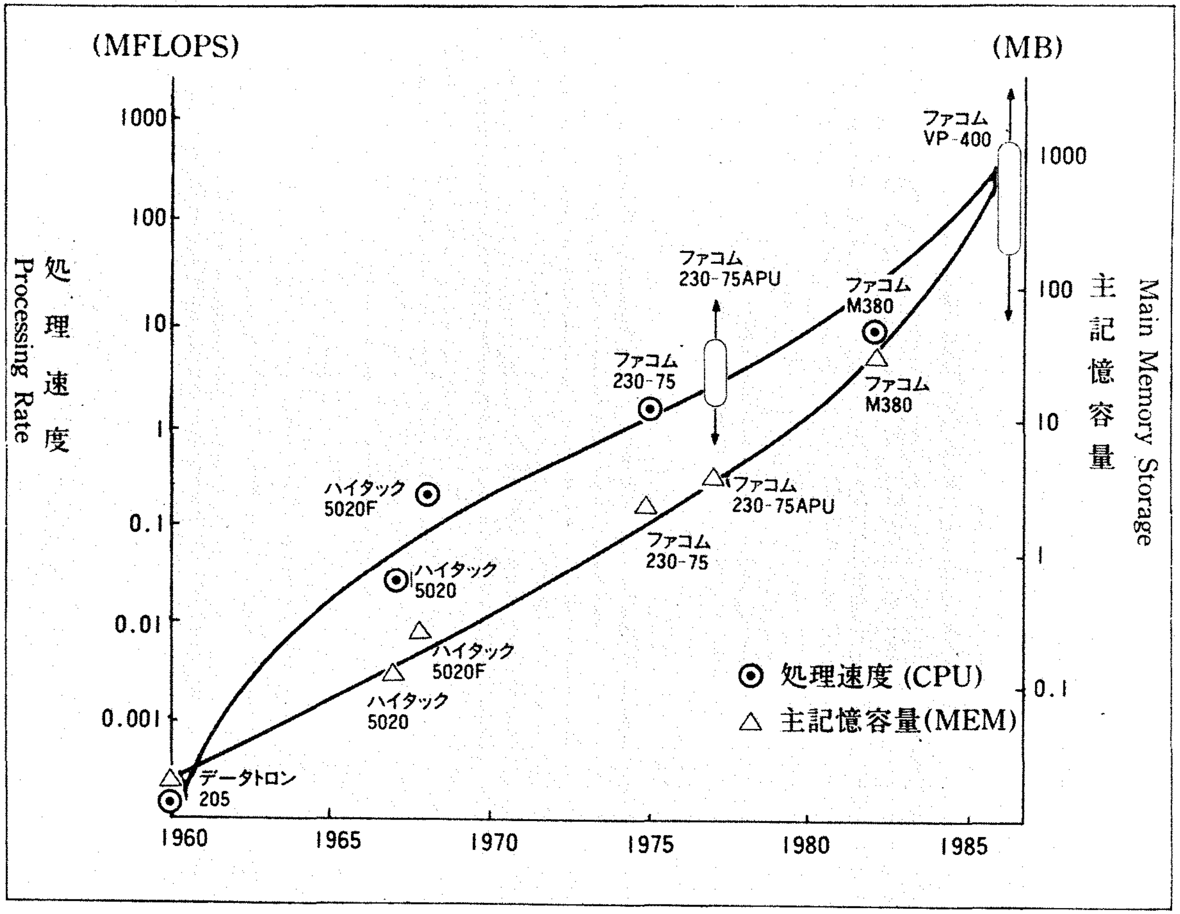


Figure 5.17. Evolution of NAL Computer's Performance

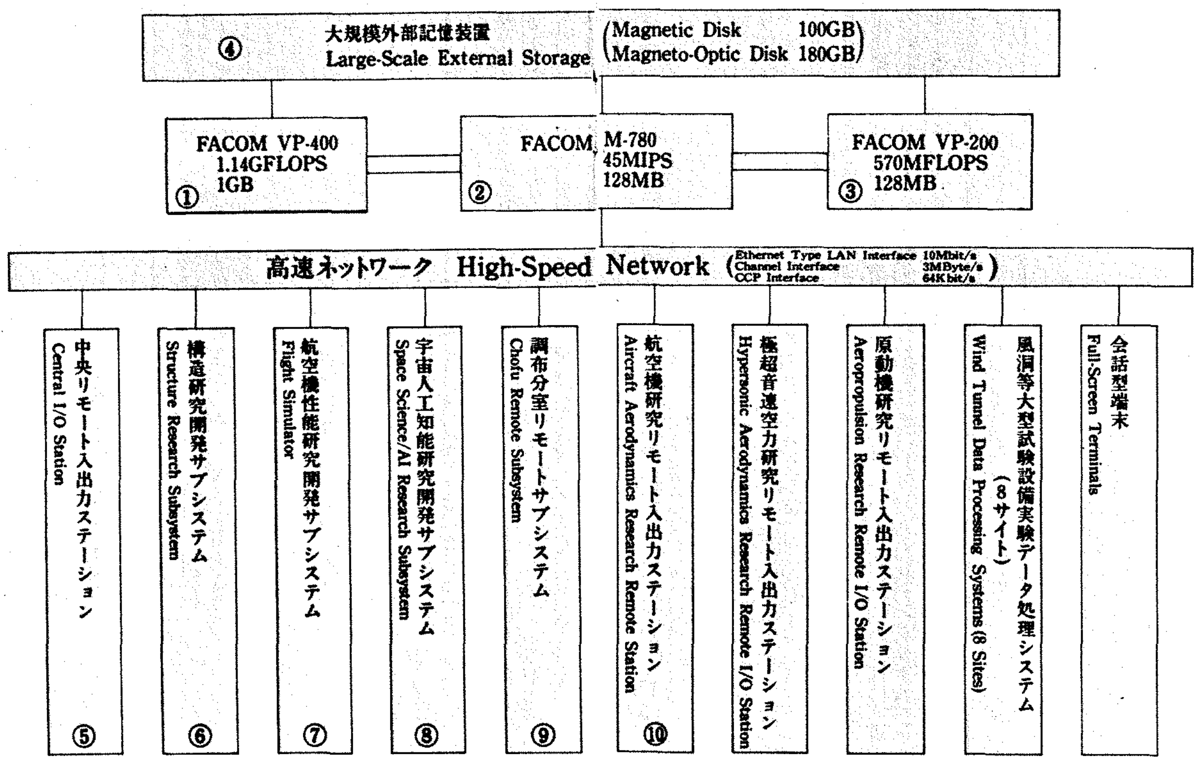


Figure 5.18. Numerical Simulator System at NAL

subsystem with a FACOM 330-FX; the Aircraft Aerodynamics research remote station with a COMTEC DS351B and a FACOM F6513C; a hypersonic aerodynamics research remote I/O station; an aeropropulsion research remote I/O station; wind tunnel data processing systems (8 locations); and full-screen terminals.

Some Examples of NAL Computations.

The Numerical Simulator System at NAL has been used to investigate both the flow field around a spaceplane, and the flow through a spaceplane propulsion system. Figure 5.1 shows the computation of the hypersonic flow around a spaceplane. Surface pressure contours and shock waves are shown for a Mach number of 7 and a 50 degree angle of attack. This example, which is based on a TVD (Total Variational Diminishing) Navier-Stokes code, was mentioned previously as being illustrative of the advanced state of computational technology in Japan.

Figure 5.19 illustrates the numerical simulation of the supersonic flow around the HOPE and H-II Rocket combination at Mach 1.8 and 3 degree angle of attack. This particular example was computed using a grid embedding scheme to solve the entire flow field, where three computational grids are overlapped with each other and solutions combined by interpolation at the boundaries.

Figure 5.2 shows constant pressure lines around a re-entry vehicle at Mach 15 with chemically reacting viscous flow, using a TVD scheme. The upper part shows the results for a perfect gas, the lower shows the effects of ionization or dissociation. This example illustrates the Japanese capability to compute real gas effects.

CFD for high-speed propulsion systems has also received a great deal of attention at NAL. Figure 5.20 depicts the numerical simulation of a reacting flow field in a cylindrical scramjet combustor. The density distribution is shown for an experimental configuration where hydrogen is injected from 18 injectors into a Mach 2.5 freestream. In this computation, a TVD code with 9 chemical species is used.

Figure 5.21 depicts a Navier-Stokes computation for chemically reacting flow. Isobaric contours and iso-contours of H_20 mass ratio are shown for the case of four transverse sonic injections of hydrogen. The geometry is a hypersonic inlet with sidewall compression and a centerbody or strut which serves as a fuel injector. The five examples shown above are indicative of the current CFD technology in Japan. More detailed resolution of the flow fields, such as the injection location in Figure 5.21, reveal that the proper physical and chemical processes are being simulated.

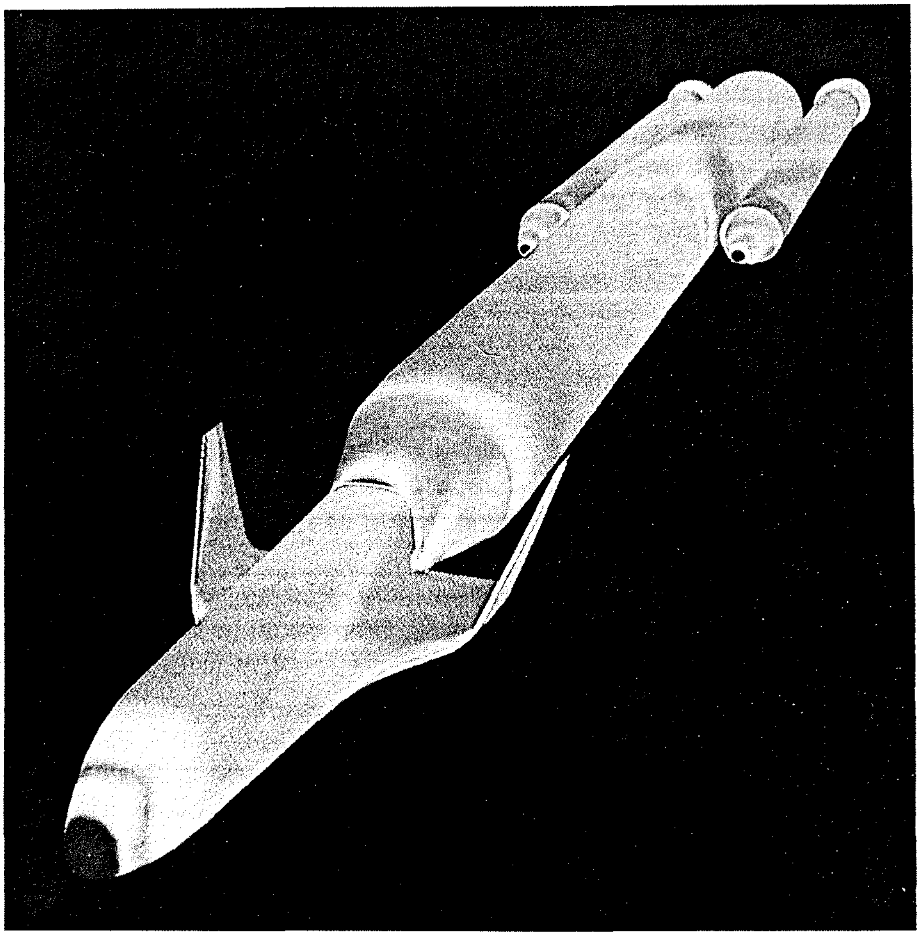


Figure 5.19.
Numerical Simulation of the Hope/H-II System at Mach 1.8

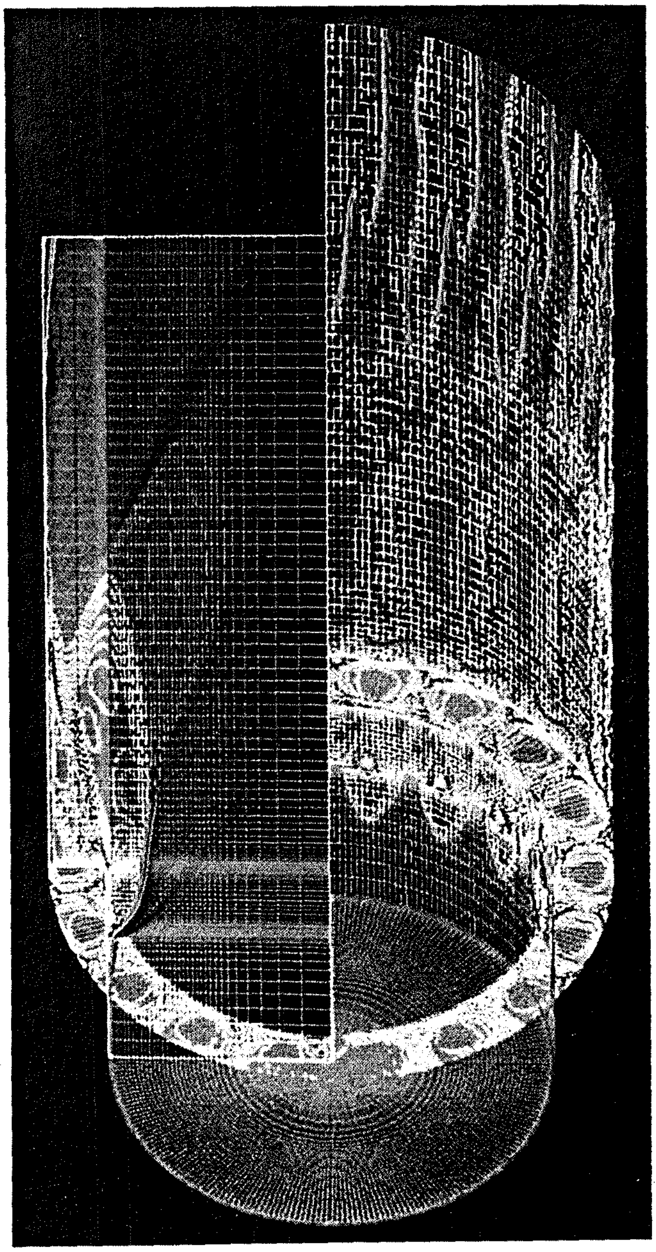


Figure 5.20.
Computations of Supersonic Injection and Combustion in a Scramjet Chamber

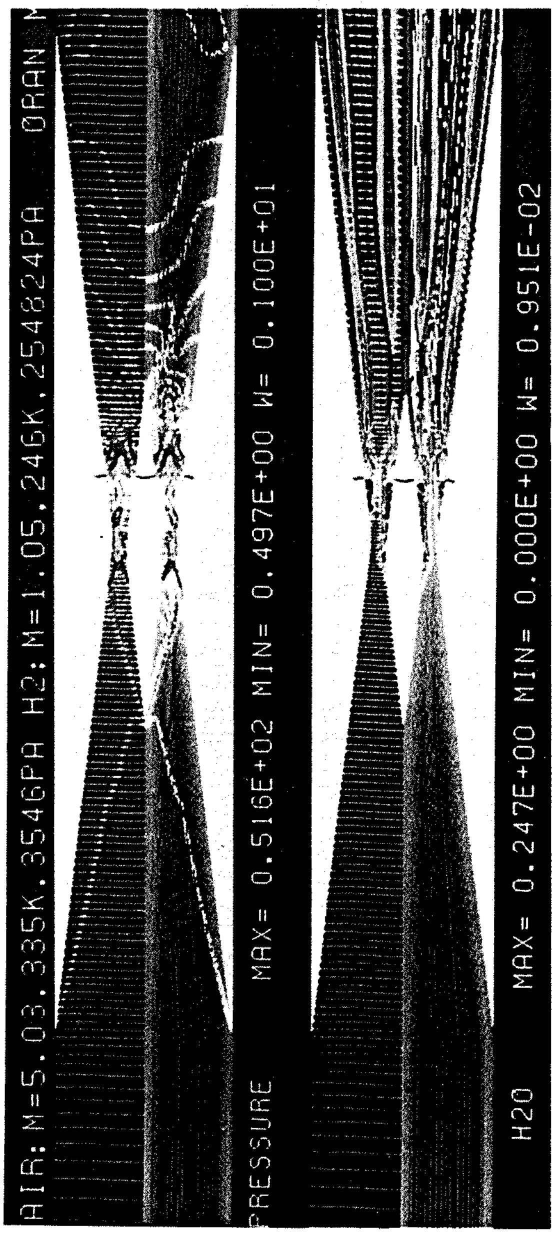


Figure 5.21. Isobaric Contours

CFD Capabilities at ICFD

The Institute for Computational Fluid Dynamics (ICFD) is operated by Professor Kuwahara and employs 27 people, including 16 engineers. Numerous Navier-Stokes computations at low Reynolds number have been carried out for analyzing the flow around automobiles. Direct Numerical Simulation (DNS) is employed for these solutions. The computational hardware available at the Institute is shown in Table 5.3 and includes a Hitachi machine with a 2G FLOPS processing rate and 512 M Bytes of memory. As computational capability increases, application to higher speeds will follow.

Table 5.3
Institute for Computational Fluid Dynamics

Professor Kuwahara
27 People/16 Engineers

Year	Machine	Speed	Memory
1986	VP 200 Fujitsu	570 MFLOPS	256 MBYTES
1989	VP 400e	1.3 GFLOPS	512 MBYTES
1987	SX2 NEC	1.3 GFLOPS	256 MBYTES
1990	S8-20/80 Hitachi	2 GFLOPS	512 MBYTES

CFD Capabilities in Universities

Inlets. The University of Osaka Prefecture is working with NAL to develop CFD techniques for scramjet applications. The work of K. Nakahashi on using adaptive grid techniques to calculate inlet flows in a two-strut inlet illustrates the use of the advanced CFD capability for scramjet research. In References 5.41 and 5.42 he demonstrates the use of an automatic grid generator to produce the grid required for CFD calculations around two sharp struts imbedded in a scramjet inlet. The technique is then used with adaptive grid to capture the shocks using an unstructured upwind differencing technique (Ref. 5.41).

Additional work is underway at Nagoya University using a Mach 8 shock tunnel facility. Numerous studies have been carried out by Professor Yasuhara on the behavior of a Mach 8 boundary layer on a sharp leading edge flat plate. In addition, research on the numerical simulation of compressible flows has been performed using a variety of methods (Ref. 5.42). A great deal of attention has focused on shock wave capturing, using a variety of techniques: Euler, thin layer Navier-Stokes, and parabolized Navier-Stokes. Two schemes have been employed: the MacCormack explicit finite volume method, and the Beam-Warming implicit method. Since the MacCormack scheme has time step limitations, the researchers at Nagoya turned to the Beam-Warming scheme. Recently some work with the TVD method was carried out to suppress spurious oscillations at discontinuities, such as shock waves. Comparisons of the numerical simulations to experimental data from the Mach 8 tunnel have been carried out. It is claimed that good agreement with experimental data is obtained for the bow shock wave location and the pressure distribution along the body.

Computational research on a Mach 15 inlet has been carried out and is reported in Reference 5.43. The purpose of the research was to investigate chemical equilibrium flow fields around a high-speed inlet and demonstrate the differences from a perfect gas. The concept of an equivalent γ was used to simplify the numerical calculations. The numerical calculations of the flow properties behind the inlet shock were compared with the values of the flow properties estimated from a shock jump condition for one-dimensional flow.

Supersonic Combustion and Shear Layers. Numerical solutions of supersonic chemically reacting flows have been performed using 2-D spatially elliptic Navier-Stokes equations in general coordinates (Ref. 5.44). A global two-step finite rate chemistry model with five species was employed to represent hydrogen-air combustion. A subgrid scale turbulence model and the k-ε two-equation model were adopted for the turbulent flow calculations. The explicit unsplit finite-difference technique of MacCormack was used to advance the governing equations, whereas the chemical source terms were treated implicitly in order to avoid the stiffness of the equations. The method was applied to the supersonic flow over a backward facing step with hydrogen injection. Comparisons of the flow field results, as well as species concentration predictions, with experimental data are reasonable. The combination of a two-step, five-species chemistry model and the subgrid scale turbulence model yielded the most effective results.

A Lomax-Bailey implicit scheme has also been applied to reacting flow calculations (Ref. 5.45). Special attention was focused on the ignition phenomena behind reflected shock waves.

Numerical analysis of two-dimensional subsonic and supersonic shear layers have been performed in order to understand the mixing process (Ref. 5.46). Such phenomena are important for fuel-air mixing and reaction.

Finally, a standing oblique detonation wave has been studied by Fujiwara et al. (Ref. 5.47) at Mach 6 and Mach 10 upstream of a semisphere and cylinder.

Prognosis

The future trend in the operating speed of numerical simulators foreseen by NAL is illustrated in Figure 5.22. Whereas, the current capability is approximately 10 GFLOPS, it is projected that processing rate will be an order of magnitude higher within five years. Corresponding increases are also projected in main memory storage, with exponential growth starting by 1995.

Japanese expectations are optimistic, and hopefully, realistic. They, like we in the United States, envision the application of CFD to multi-mode engines for hypersonic aircraft as well as for hypersonic spaceplane configurations (Figure 5.23). These applications provide a unique challenge to numerical simulation, due to the lack of high Mach number flight data. In theory, CFD provides us with the means of realistic flight simulation. The Japanese are doing their part to ensure that it will occur.

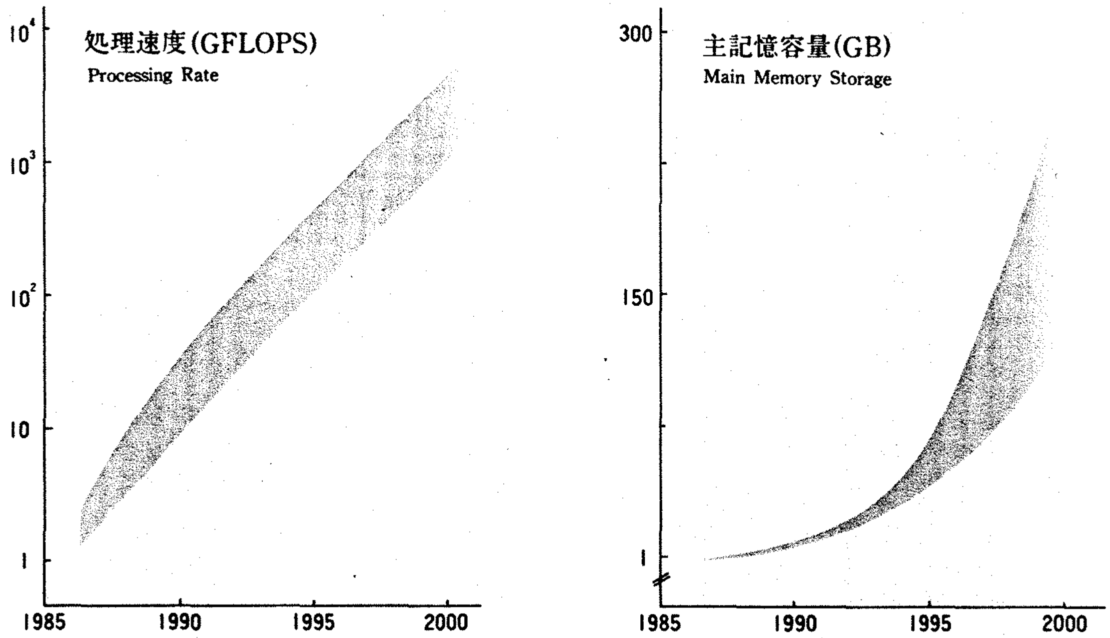


Figure 5.22. Future Trends in Numerical Simulators

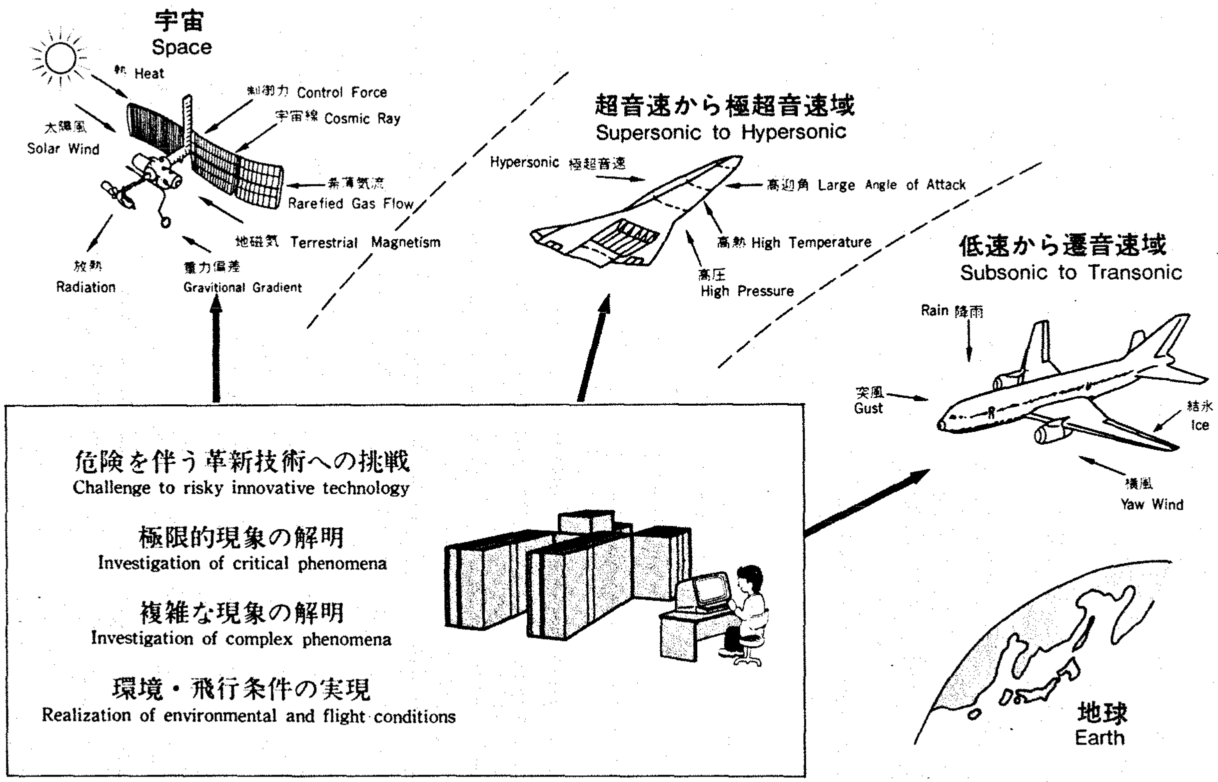


Figure 5.23. Future of Numerical Simulation

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APPENDICES

A. PROFESSIONAL EXPERIENCE OF PANEL MEMBERS

Professor Charles L. Merkle (Chairman)

The Pennsylvania State University
Department of Mechanical Engineering
104 Research Building E - 28 Bigler Road
University Park, Pennsylvania 16802 USA
Telephone: (814) 863-1501

1962 B.S.	Engineering Science, Case Institute of Technology
1966 M.S.	Mechanical Engineering, Rensselaer Polytechnic Institute
1968 M.A.	Aeronautical Engineering, Princeton University
1969 Ph.D.	Aeronautical Engineering, Princeton University

1962-1966	Senior Engineer, Pratt & Whitney
1969-1974	Senior Engineer, General Electric Co.
1974-1978	Division Manager and Senior Research Scientist, Flow Research, Inc.
1977-1979	Staff Scientist, TRW Systems, Inc.
1979-Now	Professor of Mechanical Engineering, The Pennsylvania State University
1988-Now	Director, Center for Space Propulsion, The Pennsylvania State University
1988-Now	Distinguished Alumni Professor of Mechanical Engineering, The Pennsylvania State University

James R. Brown, Manager, Upper Stage Programs, Pratt & Whitney, Government Engine Business, West Palm Beach, Florida

1964	B.S.	Chemistry, Tri-State University, Angola, Indiana

PRATT & WHITNEY

1964-1968	Analytical Engineer, Analysis of RL10 development engine and component test data.
1968-1972	Senior Analytical Engineer, Responsible for advanced RL10 engine test planning and data evaluation. P&W site manager at NASA-LeRC for RL10/Centaur system testing.
1972-1976	Senior Analytical Engineer, Responsible for cycle analysis of all RL10/RL10 derivative engines and data reduction of RL10 production engines. P&W site manager at AEDC for Space Tug Performance Evaluation Program.
1976-1978	Assistant Project Engineer, Responsible for all rocket systems performance and cycle analysis.
1979-1982	Advanced Rocket Studies Manager, Responsible for all OTV Engine study programs.
1982-1986	Rocket Systems Requirements and Technology Manager, Responsible for determination of upper stage rocket systems requirements and managing rocket technology programs.
1986-Present	Manager, Upper State Programs, Responsible for defining tasks, obtaining commitments and tracking performance of the functional groups necessary to the successful operation of all liquid upper stage programs at Pratt & Whitney.

John P. McCarty, Director, Propulsion Laboratory NASA, Marshall Space Flight Center

1958	B.S.	Mechanical Engineering, Massachusetts Institute of Technology
1972	M.S.	Engineering, University of Alabama in Huntsville
1978	M.S.	Business, Stanford University

1958-1960	Research Engineer, Rocketdyne Div., Rockwell International Corporation

1960-1963	Supervisor, Advanced Development Unit, Missile Division, Chrysler Corporation
1963-1967	Propulsion Systems Engineer, Marshall Space Flight Center
1967-1969	Chief, Propulsion Application Section, Marshall Space Flight Center
1969-1974	Deputy Chief, Propulsion and Power Branch, Marshall Space Flight Center
1974-1977	Senior Project Engineer, Marshal Space Flight Center
1977-1978	Alfred P. Sloan Fellow, Graduate School of Business, Stanford University
1978-1980	Technical Assistant, Director of Science and Engineering, Marshal Space Flight Center
1980	Chief Engineer, Cool Gasification Task Team, Marshall Space Flight Center
1981-1982	Technical Assistant, Marshall Space Flight Center
1982-1985	Chief, Propulsion Division, Marshall Space Flight Center
1985-1986	Deputy Director, Structure & Propulsion Laboratory, Marshall Space Flight Center
1986-Present	Director, Propulsion Laboratory, Marshall Space Flight Center

G. Burton Northam, Head, Propulsion Fundamentals Group,
Hypersonic Propulsion Branch, NASA Langley Research Center

1960 A.S.	Bluefield College
1962 B.S.	Mechanical Engineering, Virginia Polytechnic Institute
1965 M.S.	Mechanical Engineering, Virginia Polytechnic Institute
1969 Ph.D.	Mechanical Engineering, Virginia Polytechnic Institute

1962	Entered NASA Langley Research Center, Rocket Propulsion Branch, Solid Rocket Group
1970-1977	Atmospheric Sciences Division
1977-Now	Head, Hypersonic Propulsion Branch

Member	JANNAF Combustion Subcommittee Technical Steering Committee

Louis A. Povinelli, Deputy Chief, Internal Fluid Mechanics Division,
NASA Lewis Research Center

1954 B.S.	Mechanical Engineering, University of Detroit
1956 M.S.	Mechanical Engineering, University of Kentucky
1959 Ph.D.	Mechanical Engineering, Northwestern University
1951-1956	Engineer and Co-op Student, Bell Aircraft Corp.
1957	Research Engineer, Armour Research Foundation Illinois Institute of Technology
1959-1960	Fulbright-FIAT Postdoctoral Fellow, Polytechnic Institute of Turin, Turin, Italy
1960	Entered NASA Lewis Research Center
1960-1967	Aeronautical Research Engineer, Rocket Combustion Section, Chemistry and Energy Conversion Division
1967-1973	Aerospace Research Scientist, Hypersonics Propulsion Section, Air-Breathing Engines Division
1973-1981	Aerospace Research Scientist, Aerodynamic Analysis Section, Wind Tunnel and Flight Division
1981-1984	Head, Turbine Aerodynamics Section Aerothermochemistry and Fuels Division
1984-1985	Chief, Computational Applications Branch Internal Fluid Mechanics Division
1985-Now	Deputy Chief, Internal Fluid Mechanics Division

Additional Responsibilities:

1982-1986	Manager, Aerothermodynamics Loads Activity Space Shuttle Main Engine, Shuttle Technology Program
1984	Team Leader, Altitude Wind Tunnel Circuit Aerothermodynamics Group
1984-Now	Technical Monitor, Institute for Computational Mechanics in Propulsion

Maynard L. (Joe) Stangeland, Director, Combustion Devices and
Rotating Machinery

Rocketdyne Division
Rockwell International Corporation
6633 Canoga Park, California 91304 USA

1958 B.S.	Mechanical Engineering, South Dakota School of Mines and Technology, Rapid City, South Dakota

1957	Entered Rocketdyne as Engineer
1964-1970	Manager, Turbomachinery Stress Analysis Group
1970-1971	Manager, Component Structures Group
1971-1973	Manager, Turbomachinery Analysis and Technology
1973-1975	Program Manager, Marine Systems
1975-1979	Manager, SSME Turbomachinery
1979-1989	Director, Rotating Machinery
1989-Now	Director, Combustion Devices & Rotating Machinery

Edward E. Zukoski, Professor, Jet Propulsion & Mechanical Engineering, California Institute of Technology

1950 B.S.	Engineering Science, Harvard University
1951 M.S.	Aeronautical Engineering & Physics, Caltech.
1954 Ph.D.	Aeronautical Engineering & Physics, Caltech.

1966	Professor, Caltech.
1960-1966	Associate Professor, Caltech.
1957-1960	Assistant Professor, Caltech.
1954-1957	Engineer, Jet Propulsion Laboratory

B. JAPANESE ORGANIZATIONS AND FACILITIES VISITED

Space Planning Section
Science and Technology Agency
Tokyo

Planning Section
National Aerospace Laboratory
Science and Technology Agency
Tokyo

Space Technology Research Group
National Aerospace Laboratory
Science and Technology Agency
Tokyo

Aeroengine Division
National Aerospace Laboratory
Science and Technology Agency
Tokyo

National Aerospace Laboratory
Chofu Branch

National Aerospace Laboratory
Kakuda Research Center

Space Development Division
Ishikawajima-Harima Heavy Industries
Tokyo

Institute for Space and Astronautical Science (ISAS)
Sagamihara Campus

Space Sytems Department
Mitsubishi Heavy Industries
Tokyo

Nagoya University
Nagoya

Tanegashima Space Center
National Space Development Agency
Tanegashima

Kakuda Rocket Propulsion Center
National Space Development Agency
Kakuda

Department of Aeronautical Engineering
University of Osaka Prefecture
Sakai

C. GLOSSARY OF ACRONYMS AND TERMS

AAS -	American Astronautical Society
Afterburner -	A device used for increasing the thrust of a jet engine by burning additional fuel in the uncombined oxygen present in the turbine exhaust gases
AIAA -	American Institute of Aeronautics and Astronautics
Airbreather -	An aerodynamic vehicle propelled by the combustion of fuel that is oxidized by air taken in from the atmosphere
ALS -	American Lubrication Society
ASE -	Advanced Space Engine
ASLE -	American Society of Lubricational Engineers
ASME -	American Society of Mechanical Engineers
ATR -	Air Turbo Ramjet
Autoignition -	Self-ignition or spontaneous combustion of propellant
ANSYS NASTRAN } - ARC	U.S.-developed computer codes currently used analyze structural components
ANSYS -	Engineering Analysis System, developed by Swanson Analysis Corp. of Pittsburgh, PA
NASTRAN -	NASA Structural Analysis, developed by contractors and released by NASA
MARC -	MARC Structural Analysis, developed by Pedro Marcal
BLISK -	Turbine disk with integral blades
CARS -	Coherent Antistokes Raman Spectroscopy
CFD -	Computational Fluid Dynamics

Denton Code -	Compressible flow Euler analysis for centrifugal compressors and turbines
DNS -	Direct Numerical Simulation
DN -	The product of the bearing diameter (in mm) and the shaft speed (rpm)
EDM -	Electrical Discharge Machining
ELV -	Expendable Launch Vehicle
Enthalpy -	The sum of the internal energy of a system plus the product of its pressure multiplied by its volume
ERS -	Earth Resources Satellite
ESV -	Engine Start Valve
ETS -	Engineering Test Satellite
FCV -	Fuel Chill Valve
Flowfield -	Aerodynamic and thermodynamic states of the gas flow in a chamber or nozzle
FMPT -	First Material Processing Test
FTP -	Fuel Turbopump
Fourier Transform -	An analytical transformation of a function f(x) obtained by multiplying the function by e^{-iux} and integrating all over x
GFV -	Gas Fuel Valve
GH_2 -	Gaseous Hydrogen
Gimbal -	A device on which an engine or other object may be mounted that has two mutually perpendicular and intersecting axis of rotation and thus gives free angular movement in two directions

GIV -	Gas Intake Valve
GLV -	Gas Oxidizer Valve
GN2 -	Gaseous Nitrogen
HDTV -	High Definition Television
HF -	Hydrogen Flouride
HIMES -	Highly Maneuverable Experimental Space Vehicle
HST -	Hypersonic Transport
ICFD -	Institute for Computational Fluid Dynamics
IHI -	Ishikawajima-Harima Heavy Industries
Inconel -	(Inco), an alloy of nickel and about 16% chromium and 7% iron, highly resistant to heat and corrosion
ISAS -	Institute for Space and Astronautical Science
I_{SP} -	Fuel specific impulse, the ratio of the thrust produced by the engine to the fuel flow rate
JDA -	Japanese Defense Agency
JEM -	Japanese Experimental Module (to be attached to the U.S. Space Station)
JPRS -	Joint Publications Research Service - a U.S. Government publication service distributing international political, military, economic, environmental, and sociological news, commentary, and other information, as well as scientific and technical data and reports
JRS -	Japanese Rocket Society
JSLE -	Japanese Society of Lubrication Engineers
JSME -	Japanese Society of Mechanical Engineers

Kaufman Engine -	An electric propulsion device that generates thrust by accelerating ionized gases throught a magnetic field
Keidanren -	A Japanese federation of economic organizations
KHI -	Kawaski Heavy Industries
KSC -	Kennedy Space Center
LACE -	Liquid Air Cycle Engine
LAN -	Local Area Network
LEO -	Low Earth Orbit
LCV -	Liquid Oxidizer Chill Valve
LDV -	Laser Doppler Velocimetry
LH2 -	Liquid Hydrogen
LMATO -	Linear Motor Assisted Take Off
LN2 -	Liquid Nitrogen
LOX -	Liquid Oxygen
LTP -	Oxidizer Pump
LTV -	Laser Transit Velocimetry, also known as L2F, a non-intrusive device for measuring the velocity of a gas at a point.
MC -	Combustion Chamber
MFV -	Main Fuel Valve
MHI -	Mitsubishi Heavy Industries
Microgravity -	Condition of very low gravity, approaching weightlessness
MITI -	Ministry of International Trade and Industry

MIV -	Main Igniter Valve
MLV -	Main LOX Valve
Monbusho -	Ministry of Education
Mono-chrometer -	A device to isolate a single color or spectral wavelength from a spectrum
MOV -	Main Oxidizer Valve
Multi-Propellant -	A rocket propellant consisting of two or more substances fed separately to the combustion chamber
NAL -	National Aerospace Laboratory
NASDA -	National Space Development Agency of Japan
NASP -	National Aerospace Plane
Navier-Stokes Equations -	The equations of motion for a viscous fluid
NE -	Nozzle Extension
NEDO -	New Energy and Industrial Technology Development Organization
NPSH -	Net Positive Suction Head
NSF -	National Science Foundation
NSV -	Nozzle Skirt Valve
OTV -	Orbital Transfer Vehicle
Photo-Multiplier Tube -	A vacuum tube having a series of supplementary electrodes between the photocathode and the anode

Pitot Static Tube -	A device combining a Pitot tube and a static tube, used to determine the speed of air or other fluid by measuring the difference in pressure between moving and still fluid
Pitot Tube -	A tube with an open end pointed against the flow of a gas or liquid used for determining the total (stagnation) pressure of fluids
Pogo-Suppressor -	An accumulator device designed and placed so as to absorb low frequencies, thus reducing the tendency toward feed line or structural vibration-induced combustion instability.
POV -	Preburner Oxidizer Valve
Preburner -	An auxiliary combustor used to provide hot gases to drive fuel and/or oxidizer turbopumps. To prevent turbine damage, preburners are generally operated fuel-rich at temperatures of about 1000 degrees Kelvin.
PSIA -	Pounds Per Square Inch Absolute
PTFE -	Polytetraflouroethylene
Ramjet Engine -	A reaction propulsion (jet) engine that produces thrust by combusting fuel in an air stream that has been compressed solely by "ram" compression obtained from the vehicle's forward motion without the use of turbomachinery.
RCAST -	Research Center for Advanced Sciences and Technology
Reynold's Equations -	A subset of the Navier-Stokes equations that describe the motion of a very slow moving fluid, for example that of a lubrication fluid through a bearing.

Rotor-dynamics -	The branch of physics dealing with the motion of rotating parts of an apparatus and the action of forces on the apparatus
RVC -	Row Velocity Compounded
SAC -	Space Activities Committee
SAE -	Society of American Engineers
Schlieren -	A special optical technique that renders regions of different density in a moving or stationary fluid visible as light or dark streaks
Scramjet -	Supersonic Combustion Ramjet
SFU -	Space Free Flyer
SHSC -	Small High Speed Centrifugal Pump
Silane -	A molecular compound composed of silicon and carbon (SiH_4) that is analogous to methane (CH_4)
SJAC -	Society of Japanese Aerospace Companies
Splines -	A series of projections and slots used instead of a key to prevent relative rotation of cylindrically fitted machine parts
SRB -	Solid Rocket Booster
SSME -	Space Shuttle Main Engine
SST -	Supersonic Transport
SSTO -	Single Stage To Orbit
STA -	Science and Technology Agency
Stator -	Part of a turbopump assembly that remains fixed or stationary relative to a rotating or moving part of the assembly

STOL -	Short Take Off and Landing
TRJ -	Turbo Ram Jet
Turbine -	A machine consisting of one or more bladed disks (rotor or turbine wheel) and one or more sets of fixed vanes (stator) inside a casing, so designed that the wheel is turned by incoming fluid (hot gas in aerospace applications) striking the blades
Turbopump -	An assembly consisting of one or more pumps driven by a hot gas turbine
TVD -	Total Variation Diminishing
Viscosity -	Fluid resistance to flow caused by internal molecular attraction

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