NASDA-TMR-950012T

NASDA Technical Memorandum

Research on Reusable Transportation System :
Study of VTOL (Vertical Takeoff and Landing) Experiment

October 1996

NASDA

NASDA-TMR-950012T

NASDA Technical Memorandum

Research on Reusable Transportation System : Study of VTOL (Vertical Takeoff and Landing) Experiment

Hiroshi Aoki[*1], Yutaka Takano[*1], Akira Ginoza[*1],
Toshiyuki Yoshida[*2], Shigeyasu Iihara[*3].

[*1] Future Space System Laboratory, Systems Engineering Department,
Office of Research and Development,
National Space Development Agency of Japan.
[*2] Kawasaki Heavy Industries,
[*3] Ishikawajima-Harima Heavy Industries.

National Space Development Agency of Japan

List of Contents

1. Objective

As the first step to establish the "Vertical Landing Technology Using Rocket Engines," one of the common technological tasks necessary for realizing "Reusable Rockets" and "Lunar and Planetary Landing Vehicle," the "Takeoff and Landing Flight Experiment" on the ground are planned.

2. Guidelines for Examining the Concept of the Experiments

2.1. Objectives of the Experiment

The "Winged" and "Vertical Soft Landing" modes are currently considered possible as returning modes for "Reusable Rockets"; with the examination of the former having far advanced in the course of the "HOPE Experiment," the present research focuses on the "Latter" and is demonstrating its superiority in mass freight transportation. In the United States also, the operation modes for "Vertical Takeoff and Landing" are being verified as flight testing of the "Delta Clipper" makes progress.

On the other hand, the "Vertical Soft Landing Technology" is a landing means in common use for the "Lunar Landing and Probing" which will provide an important step toward space development. It is necessary, therefore, for Japan to demonstrate and acquire this technology at an early stage.

Although conditions to be simulated differ, depending on whether or not the atmosphere is present and how heavy the experimental machine is, it is considered possible to demonstrate through flight tests in the earth environment basic technological problems including deceleration, hovering, descent, and soft landing; therefore, it is not always necessary to plan direct simulations.

Based on the factors mentioned above, the objectives of the present experiment have been decided as described below. Although the "Engine Thrust Control Technology" and some other items contain portions that can be verified through engine unit tests, the present experiment aims ultimately at actually verifying the system consistency by combining all systems and simultaneously at acquiring the ground data for reviewing the reasonable allocation of the technological risk margin for all systems.

(1) Acquisition of the engine thrust control technology
  Confirmation of design techniques for thrust controlled engines
  Confirmation of operating methods for thrust controlled engines
  Confirmation of operating limits of thrust controlled engines and grounds for the limits
  Confirmation of responsiveness of thrust controlled engines and its dominant factors
  Abstraction of other problems with thrust controlled engines
(2) Acquisition of the engine cluster technology
  Confirmation of the interference prevention design method
  Confirmation of the method for compensating for individual differences and alignment
(3) Acquisition of the soft landing guidance technology
  Confirmation of the design technique for vertical takeoff and landing guidance and control principles
  Evaluation of the vertical takeoff and landing guidance and control technology using variable thrust engines
(4) Acquisition of the propellant surface control technology
(5) Confirmation of the ground effect
(6) Confirmation of the flame protection method
(7) Confirmation of the landing gear design technique
(7) Confirmation of the ground operation method
(9) Confirmation of reuse operation method
(10) Verification of the consistency of the entire takeoff and landing systems

The results obtained from the activities enumerated above are expected to help conduct the concept design for the "Reusable Rockets" and "Lunar Landing Vehicle." Fig. 2. 1-1

2.2. The precondition for examining the concept of experiment

The following items are preconditions for embodying the concept of the experiment. These preconditions could be revised as the examination progresses.

(Time for the experiment)

The flight experiment will be conceptualized with a view to executing it within a three year time frame.

(Cost for the experiment)

The objective experiment cost is the cumulative total of the research budget for three years.

(Site for experiment)

A domestic site where security distance can be secured and no problem with environmental preservation occurs is the current precondition for selecting a site for the experiment; however, an overseas site may be considered as the situation develops.

(Technological level)

The existing technologies within the country are put together for the present takeoff and landing experiment. The introduction of new technologies should be avoided except those associated with the objectives of the experiment.

(Equipment Configuration)

Products that have already been developed or commercially available products should be used as devices and equipment for the experiment as far as possible.

(Significance of the devices and equipment used)

The purpose of the present experiment is to acquire technologies that can be reflected in future activities, such as the confirmation of a design technique; it is not necessarily the purpose of the present experiment to evaluate devices and equipment to be used for a subsequent plan.

(Cost reduction)

In order to reduce the cost for the experiment, products that have already been developed and those that have been retained should be used as extensively as possible.

(Overseas products)

Overseas products may be used provided that details of the associated technologies are fabricated public.

(Propulsion system)

A rocket engine with variable thrust function is to be used as the prime mover for vertical soft landing. The current precondition is that a gas pressure two-fluid propulsion system, which has already been developed, be employed for the main propulsion system.

(Simulation of the lunar environment)

Simulating the "lunar gravity" is not intended.

(Aerodynamic control)

Simulating the aerodynamic control is not intended.

(Flight mode)

The present experimental vehicle aims at demonstrating self-contained free flight not relying on remote piloting.

(Weight control)

Marginal design aiming at weight reduction is excluded as much as possible.

(Ground equipment)

The ground equipment should be minimal and portable.

2.3. Preconditions for Mission

Of the technologies required to realize "Reusable Rockets" and "Lunar and Planetary Landing Vehicle," common technologies concerning "Vertical Takeoff and Landing" are demonstrated as far as possible, with associated problems being identified.

(Takeoff)

The vehicle lifts by self-propulsion.

(Self-contained control)

The position, attitude, and velocity of the vehicle can be controlled by the self-contained control of onboard equipment in the entire flight domain.

(Landing method)

The landing method consists of automatic vertical soft landing with use of rocket engines and does not, as a rule, rely on ground support equipment.

(Flight altitude)

More than 100 m; flight in the visible range is the current precondition for flight altitude.

(Flight distance)

Not specified

(Hovering and moving)

The vehicle is capable of hovering and low velocity lateral movement to simulate obstacles evading flight in the future.

(Flight time)

About 60[s] including hovering time

(Flight pattern)

Fig.2.2-1 shows the draft flight pattern to be flown.

(Data acquisition)

The data to be acquired consist of information concerning the position, attitude, and velocity, control output signals, pressure values, and data from various parts; these data should be capable of being transmitted to the ground in real time by a telemeter system.

(Reusability)

The present experimental vehicle is capable of repeated flight not less than 10 times without replacing major parts.

(Emergency landing)

The present experimental vehicle is capable of making emergency landing on unleveled ground.

2.4. Preconditions for Systems

The system meets the mission requirements. The details and additional function requirements are given as follows:

(Overall structure and configuration)

A freestanding structure using truss construction is the precondition. Not less than three shock absorbing landing gears are installed on the periphery not only to prevent the component devices from being in direct contact with the ground but also to secure clearance between the main propulsion unit outlet and the ground.

(Total height, gross weight)

In order to facilitate handling, the total height is limited to 5[m] and the gross weight to 1[T].

(Ground equipment interface)

The structure system, heat control system, propulsion system and electric power system are interfaced directly with the general purpose portable ground equipment.

2.5. Preconditions for Subsystems

2.5.1. Propulsion System

(Main propulsion system)

Adoption of a gas pressure two-fluid propulsion system is the precondition; either a single or multiple combustion chambers are provided.

(Thrust control system)

The main propulsion system permits an adjustment of thrust by not more than 60[%] (T.B.D.).

(Tank system)

The feed line is provided with a flowmeter. The level gauge is installed as required.

(Attitude control system)

The attitude is controlled by the gas jet, main propulsion control, or main thrust direction control (gimbal) as required, either singly or in combination.

2.5.2. Structure

(Overall structure)

The system is provided with a hoisting mechanism for transportation and transfer.

(Landing gear)

The landing gears are provided with shock absorbers that can be used repeatedly and withstand the gross static weight. The landing point is basically supposed to be a leveled ground; with the possibility of landing on a sandy ground being taken into account, however, the ground striking portion has a structure to allow the vehicle to keep its attitude.

(Thrust support structure)

Not specified.

2.5.3. Thermal Control System

(Flame protection)

To provide protection against flame caused by the reverse jetting at the time of landing, sections of the vehicle requiring flame protection are provided with thermal insulation.

2.5.4. Navigation Guidance and Control System

(Navigation)

The vehicle has a function to implement composite navigation combining IMU and RA (Radio Altimeter); adoption of a speedometer will be considered as the examination of the experiment makes progress.

(Guidance)

The guidance command can be outputted that enables the vehicle to follow a predetermined flight profile; the effect of wind is taken into consideration in carrying out the experiment.

(Control)

The attitude of the vehicle is controlled by combining the necessary elements from engine thrust differential, gimbal and RCS.

(System management)

The function to perform sequence control, input/output management, data management, performance monitoring, and ground inspection is provided.

2.5.5. Communications System

(Communications equipment)

T.B.D.

(Antenna System)

T.B.D.

(Commanded destruction system)

T.B.D.

2.5.6. Power Supply System

(Battery) T.B.D.

(Power distribution board) T.B.D.

2.6. Preconditions for Operations

The "Operational Stage" is defined as one beginning with carrying in of the equipment to the experiment site and ending with withdrawal of the equipment therefrom; the requirements for this stage are stated as follows:

(Operational requirements)

Operations of the vehicle can be performed by a team of about 10 persons. In addition, under the normal condition, inspections to be performed within three days between one operation and another enable repeated operations of the vehicle.

(Conservation requirements)

After undergoing nondestructive inspection, the vehicle is reusable. The design of the vehicle minimizes disassembly for inspection. No permanent building is required for housing the vehicle.

(Reliability requirements)

The reliability required of the vehicle is the same as that required of ordinary experimental vehicles; as a rule, redundant design is not employed

(Safety requirements)

For the occurrence of predicted troubles, modes in which damage occurs and the extent of the damage should be predicted. The vehicle should be, as far as possible, recovered by using a parachute and the like. Equipping the vehicle with a commanded destruction device is to be considered as required.

(Environmental preservation)

For the occurrence of predicted troubles, the degree, extent, and effects of environmental pollution should be predicted.

2.7. Criteria for Selecting the Experiment Site

On the assumption that a two-fluid propellant is used, the criteria for selecting an experiment site are as follows:

(Safety distance)

The safety distance meets the requirements set by AFR-100.

(Impact dispersion area)

The experiment site makes it possible to set the impact dispersion area on land.

(Propellant diffusion area)

Emergency diffusion of propellant does not exert fatal impact on organisms in the vicinity.

(Weather)

On the assumption that operations of the experimental vehicle are performed outdoors, an experiment site with smaller precipitation is desirable.

(Water supply)

Water supply for cooling, sprinkling, and fire-fighting purposes is available.

(Transportation)

To minimize work on the site, the vehicle allows itself to be carried in the experiment site without being disassembled.

(Other requirements)

T.B.D.

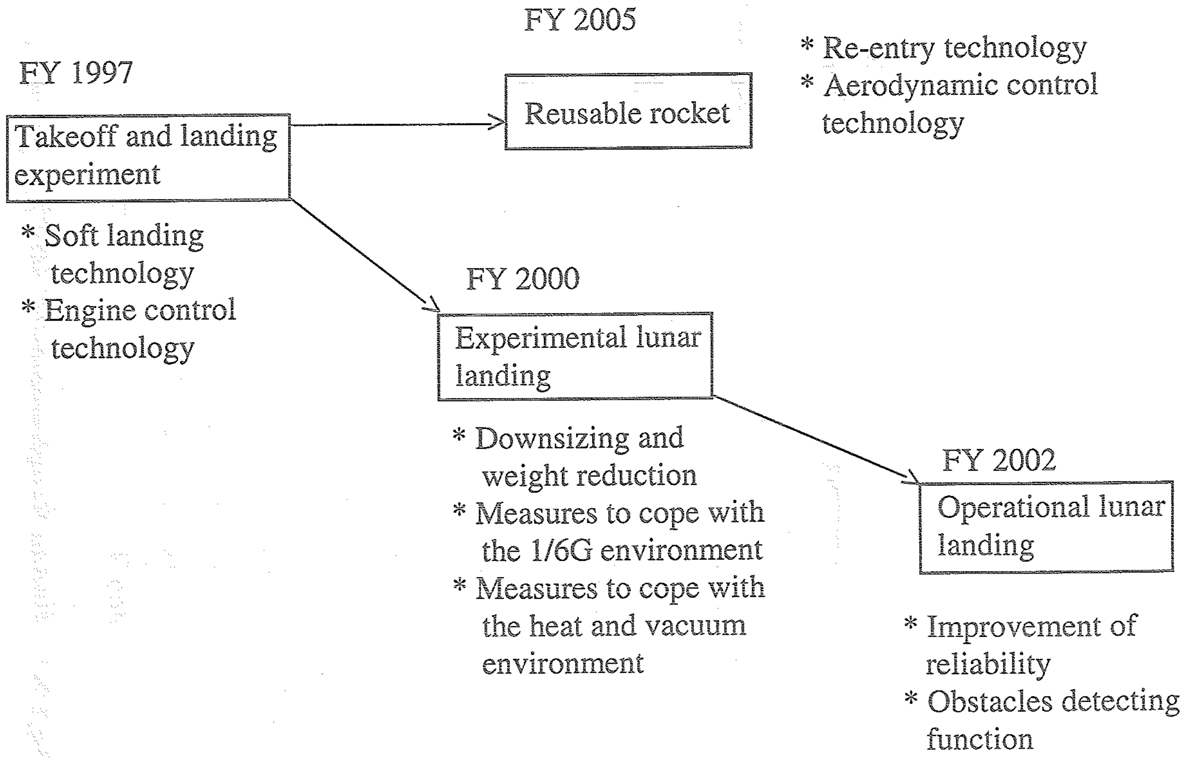


Fig.2.1-1 Technology-reflecting Scenario (draft)

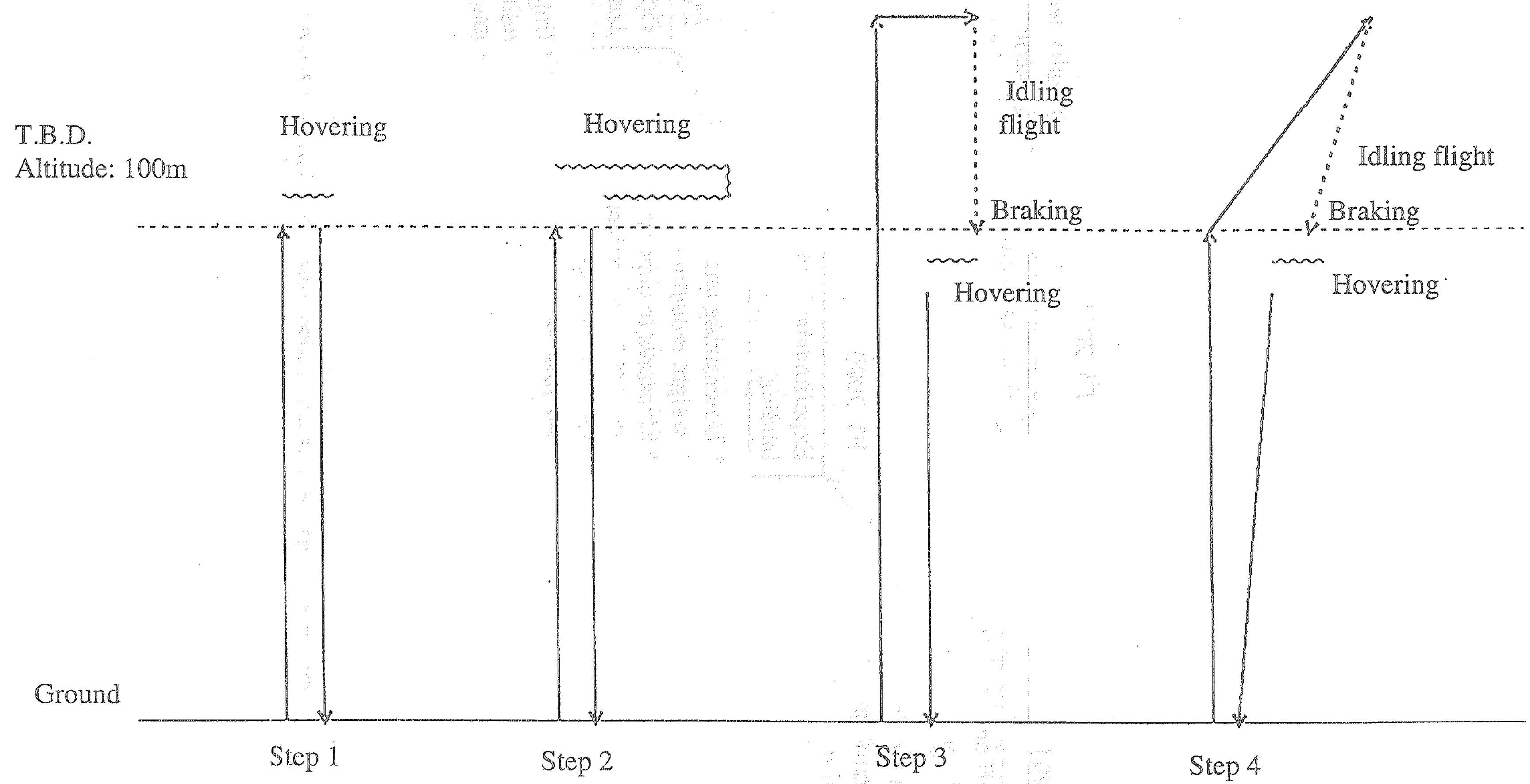


Fig.2.2-1 Mission Requirements (draft flight pattern)

3. Examination of Missions

3.1. Mission Requirements

The vehicle is capable of performing multiple flights in order to demonstrate the vertical landing technology, one of those required of lunar landing vehicles and reusable boosters. In principle, it is sufficient for the purpose of this experiment for the vehicle to be capable of performing transition from the state of having both the vertical and the horizontal velocity component, via halt in the air and hovering, to soft landing.

More specifically, the vehicle is capable of performing multiple flights to allow the following items to be confirmed:

(A) Propulsion system
- (a) Engine thrust control technology
- (b) Reusable engine
- (c) Thermal and fluid dynamic characteristics during reverse jetting
- (d) Propellant surface control technology (liquid level monitoring, rocking prevention, and the like)
- (e) Engine cluster technology

(B) Navigation guidance and control system
- (a) Automatic guidance and control technology
- (b) Control characteristics during reverse jetting

(C) Landing system
- (a) Landing gear technology (shock absorbing, prevention of overturning)

(D) Thermal control system
- (a) Landing flame protection technology

3.2. Mission Design

(A) Flight Phase

A flight consists of the four phases that follow:

(a) Ascent phase

(b) Descent phase

(c) Hovering phase

(d) Soft landing phase

Fig.3.2-1

(B) Flight pattern

The vehicle is put to the flight experiment corresponding to each of Steps 1 though 4 shown in Fig.2.2., in that order. To be more specifically, the ultimate goal is performing the flight transition from the state having both the vertical and the horizontal velocity component, via halt in the air and hovering, to soft landing. See Fig.3.2-2.

(C) Reusability

The reusability is focused on. The vehicle is capable of flying 10 times without replacing parts.

(D) Takeoff and landing ground

The takeoff and landing ground requires a concrete-paved area of about 100[m]x 100[m]; depending on the circumstances, steel plates may be laid to secure necessary areas. In addition, a flat security zone extending at least 100 [m] from the outside edges of the ground is required.

Takeoff and landing are performed on a flat ground.

The takeoff and landing ground is, as a rule, paved; however, depending on the circumstances, a sandy lot may be used.

(E) Landing precision (T.B.D.)

The landing precision is of the same order as that of the DC-X, with the following figures taken as the objective:

Position error:	Within 45[m] of the objective landing point
Velocity error:	Horizontal Within 1.5[m/s]
	Vertical 0.6[m/s]
Attitude angle error:	Not more than 0.6[deg]
Attitude angular velocity error:	Not more than (T.B.D.)

The design conditions for the landing gear are as follows:

Velocity error:	Horizontal Within 3.0[m/s]
	Vertical Within 0.7[m/s]
Attitude angle error:	Not more than 5[deg]
Attitude angular velocity error:	Not more than (T.B.D.)

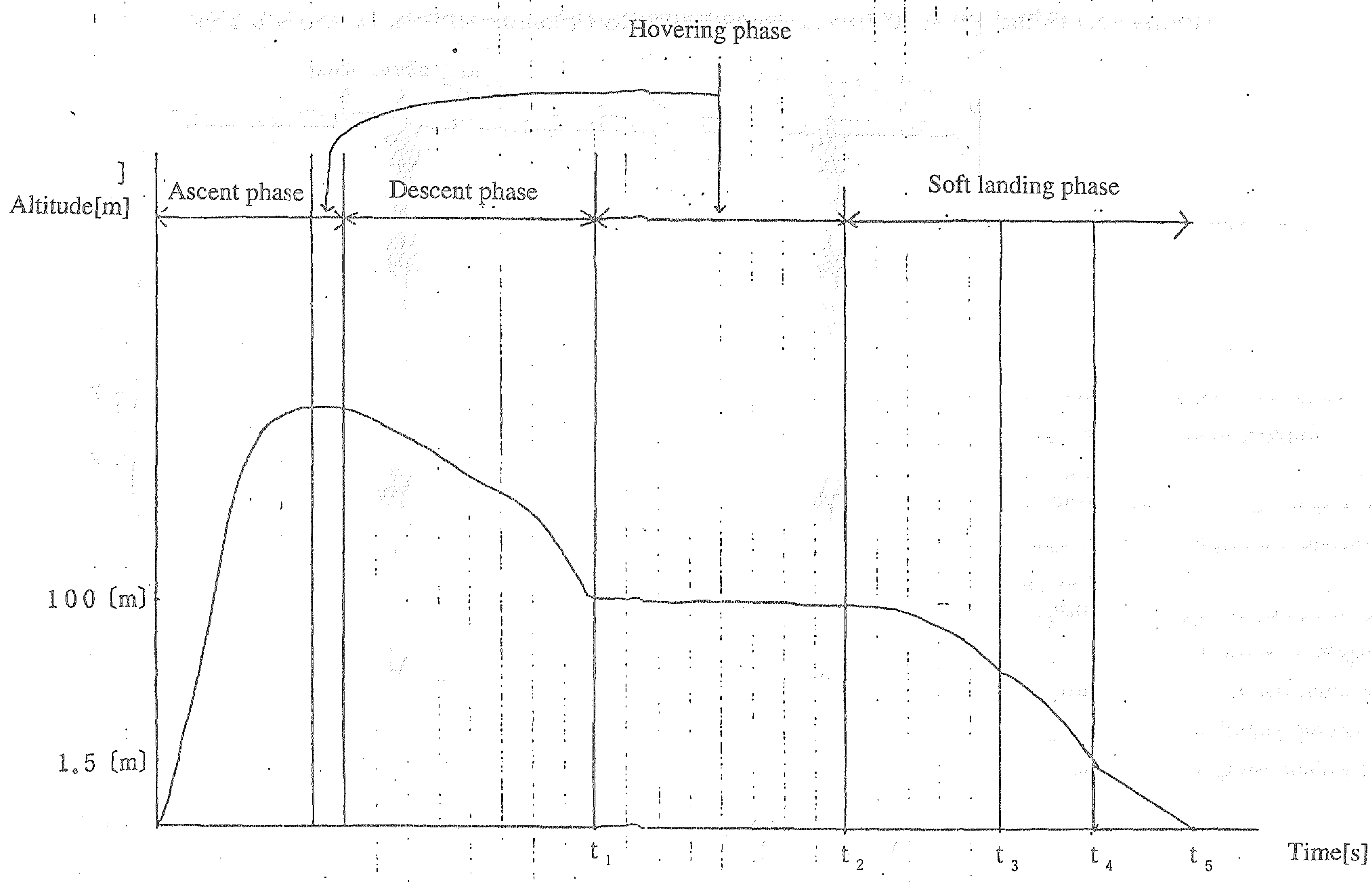
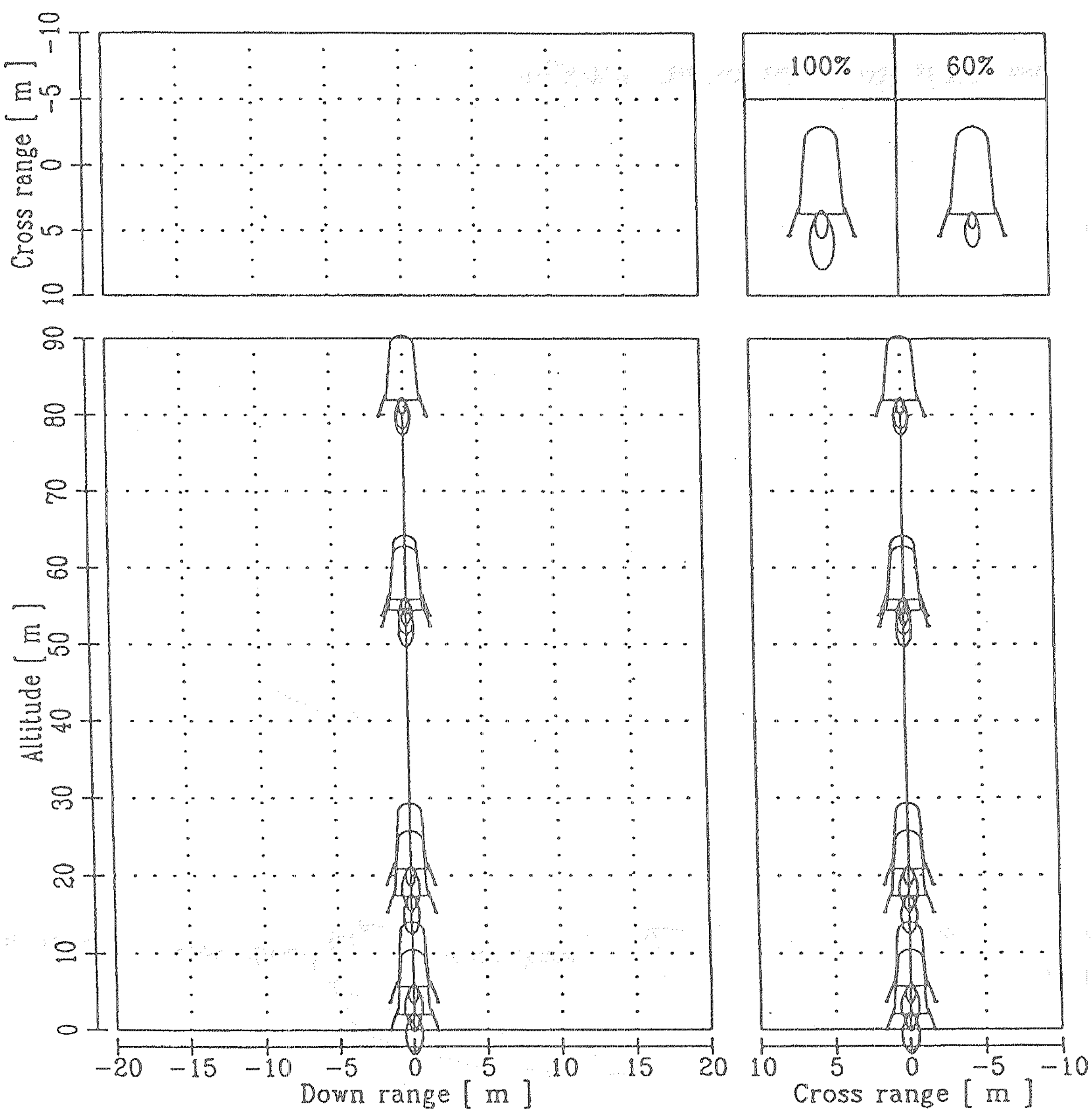


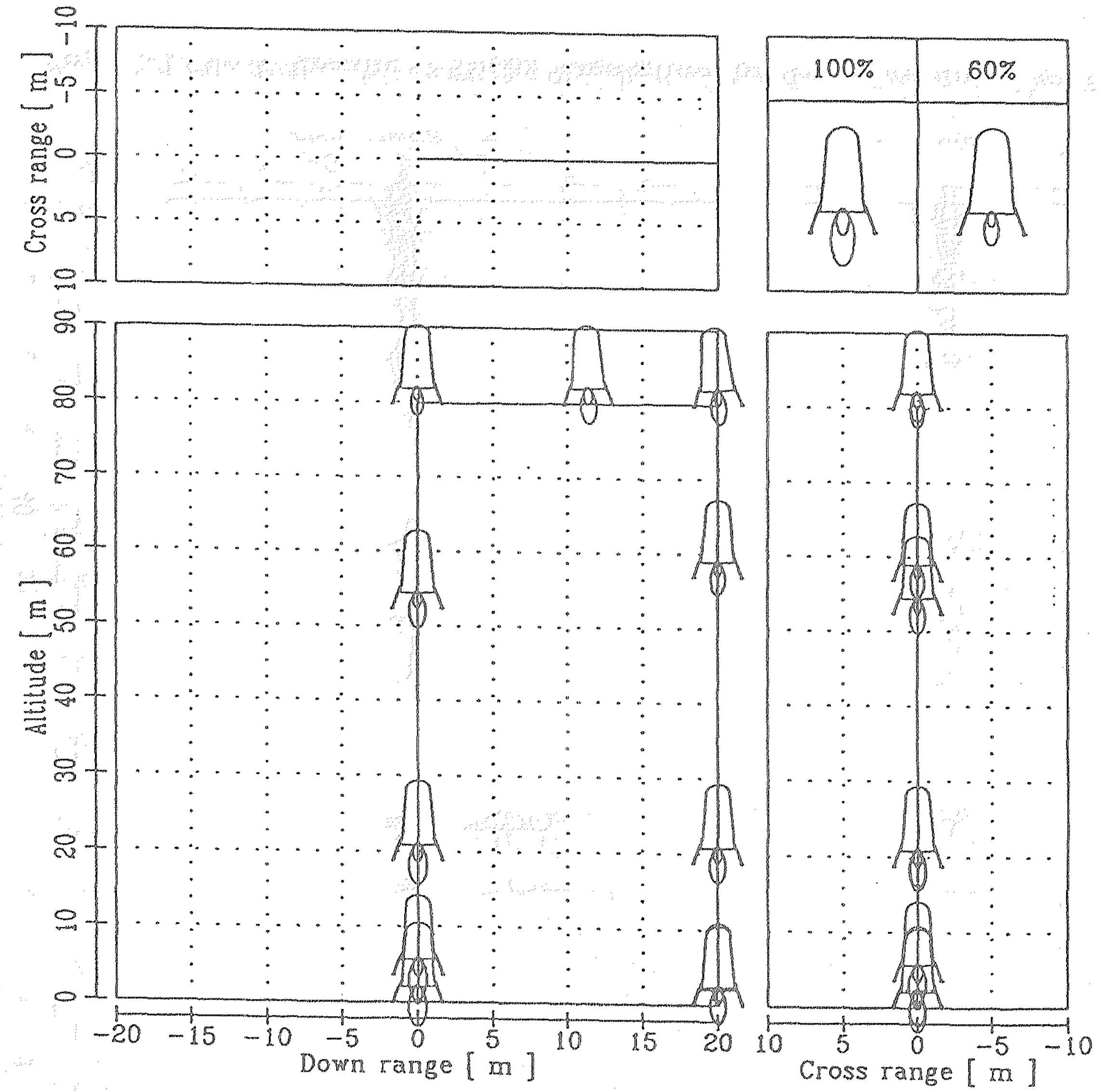
Fig.3.2-1 Definition of Flight Phases



Confirmation of the thrust control performance
Confirmation of the gimbal performance
Confirmation of the attitude static stability
Confirmation of the altimeter performance
Confirmation of the design of the shock absorbing landing gears
Confirmation of the flame protection measures
Demonstration of the soft landing and recovery of the vehicle
Confirmation of the reusability
Demonstration of the standard descent
Demonstration of the ground operability

Interval between vehicle position plots: 5 sec

Fig.3.2-2 (No.1) Results of Flight Simulation for Vertical Jump Flight (No Wind)



Checking of controllability and dynamic stability during hovering
Confirmation of dynamic stability

Interval between vehicle position plots: 5 sec

Fig.3.2-2 (No.2) Results of Flight Simulation for Hovering Horizontal Movement (No Wind)

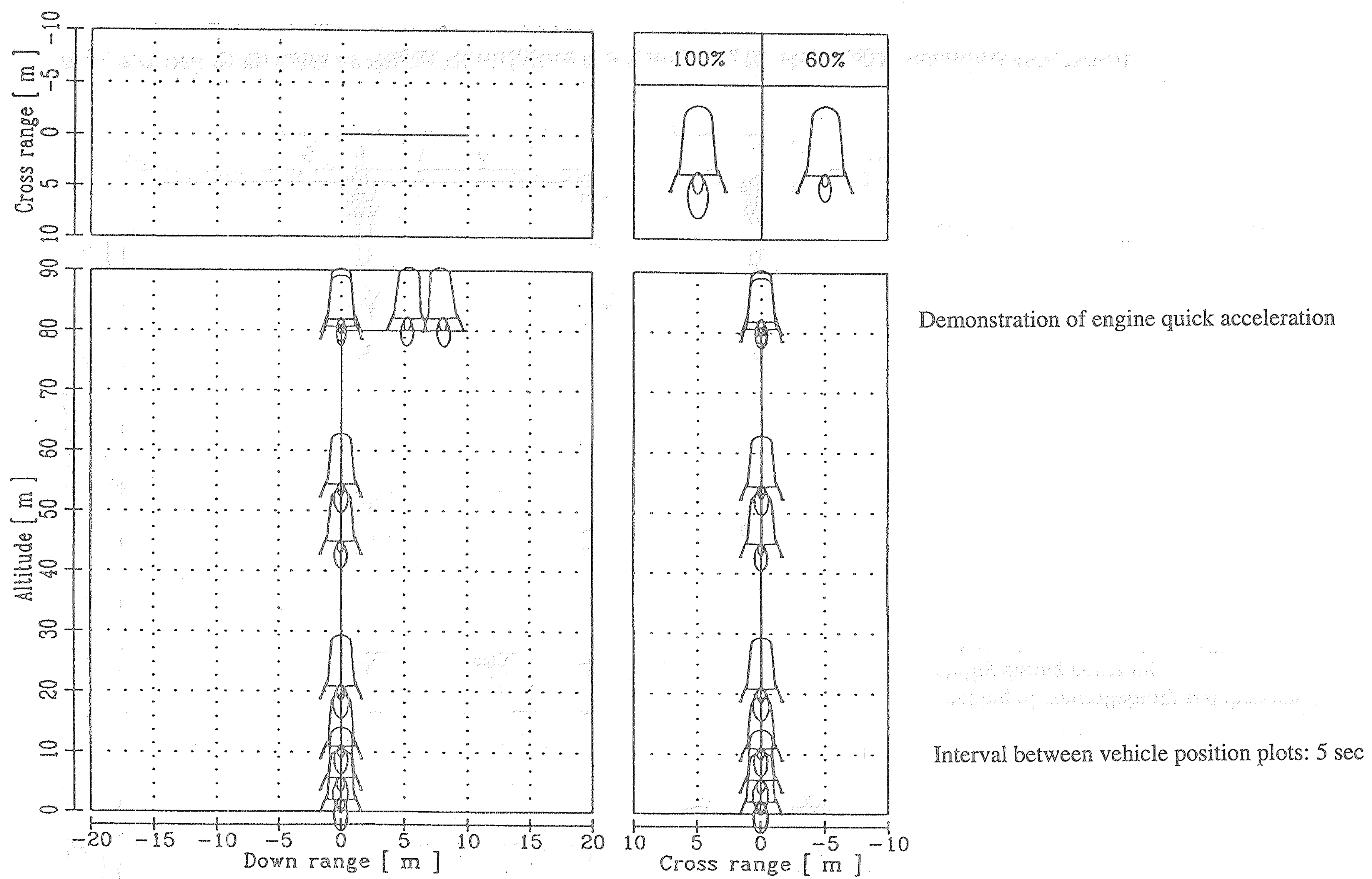


Fig.3. 2-2 (No.3) Results of Flight Simulation for Drop Braking (No Wind)

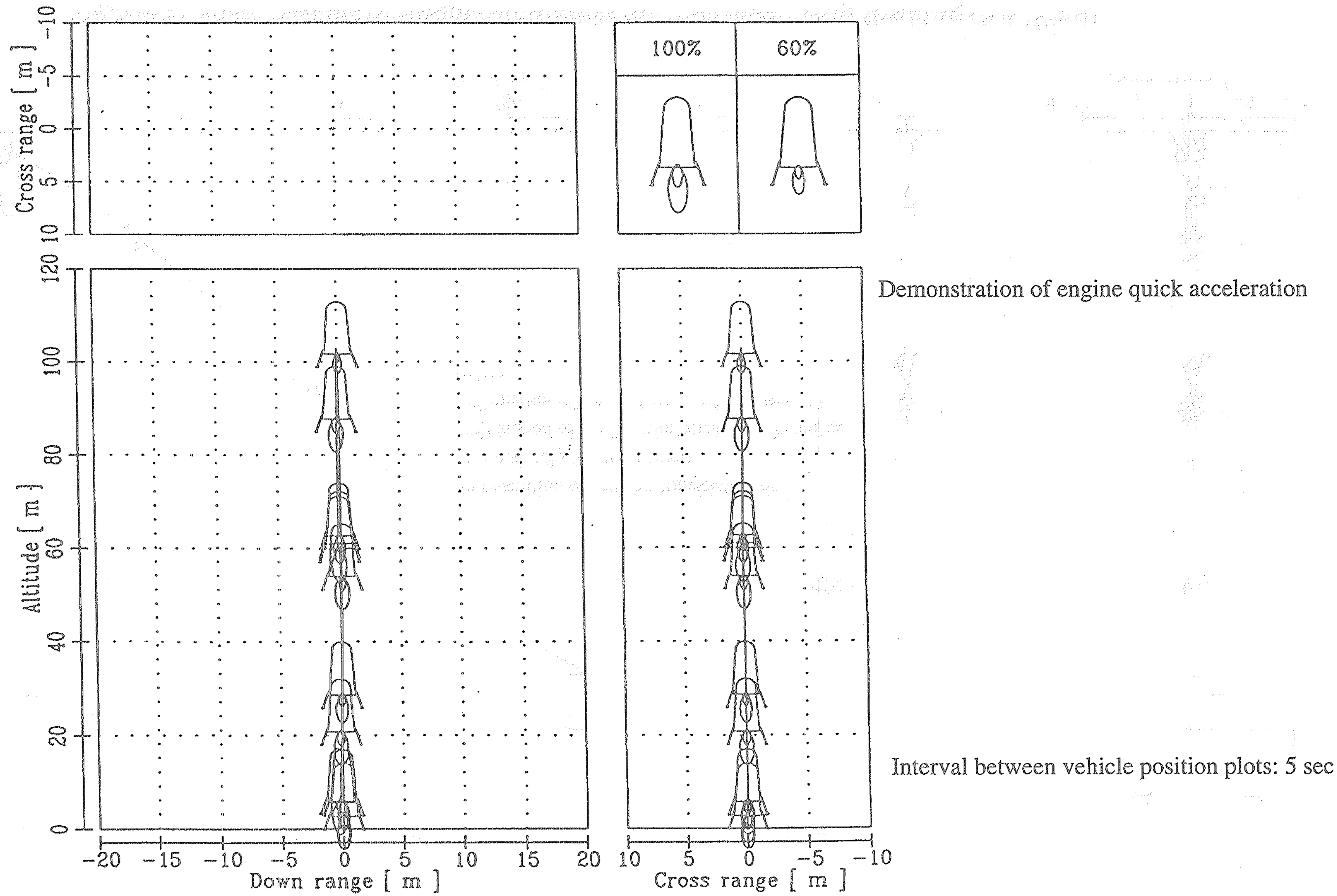


Fig.3. 2-2 (No.4) Results of Flight Simulation for Drop Braking (No Wind)

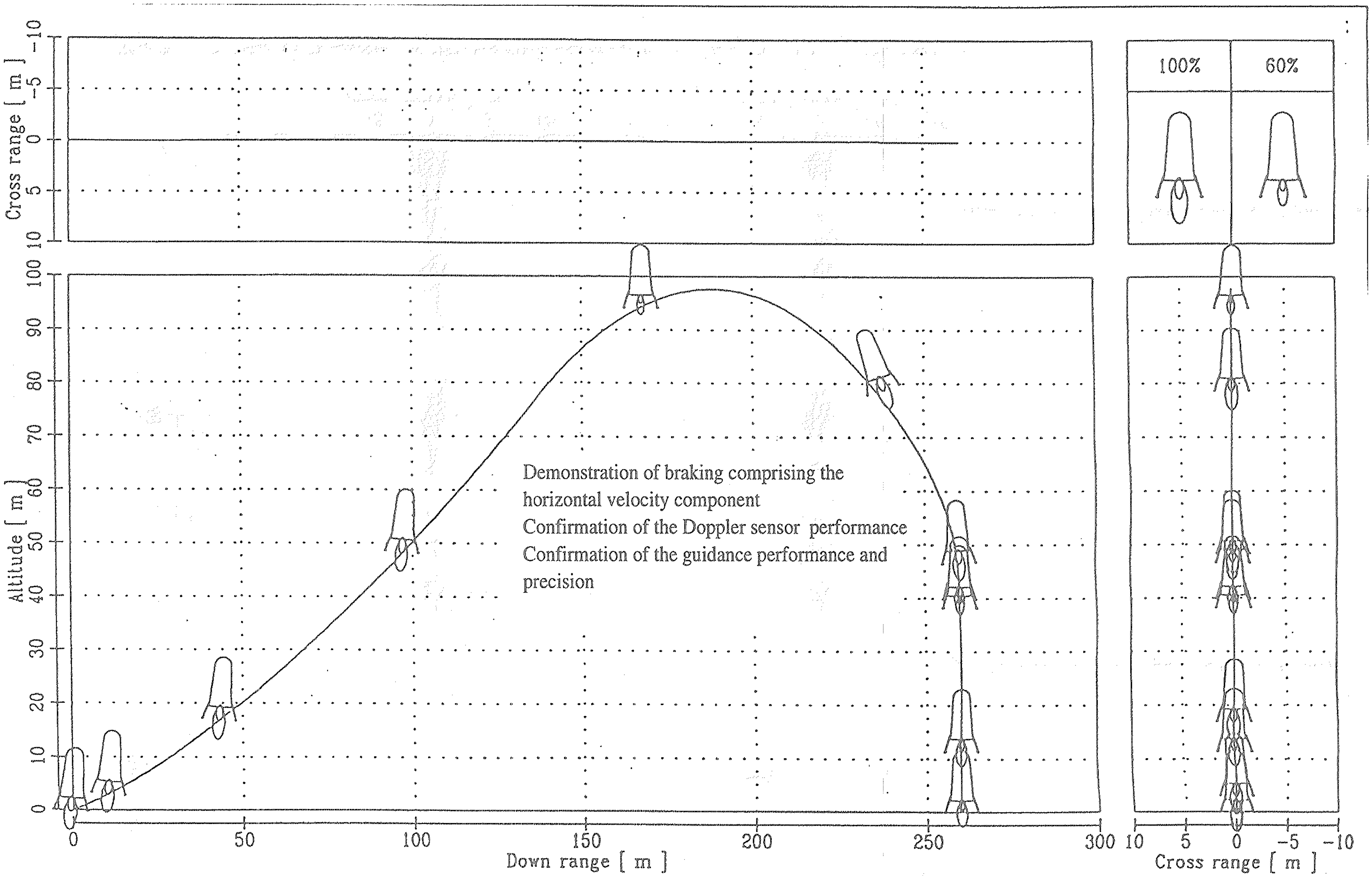


Fig.3.2-2 (No.5) Results of Flight Simulation for Parabolic Drop Braking (No Wind)

4. Examination of the Systems

4.1. System Requirements

In addition to the general items described in 2.4 Preconditions for Systems, the following items have been considered:

(a) Superior maintainability;
(b) Shorter turnaround time;
(c) The experimental vehicle can easily be transported from the landing point to the takeoff point by means of a crane or the like;
(d) The possibility of transporting the experimental vehicle to the takeoff and landing test site by land or sea is considered.

4.2. System Design

4.2.1. Engines, Number of Engines, Configuration, Gross Mass, and Structural Mass

The results of the examination of the draft system design for the experimental vehicle are shown below.

The number of engines are reduced to reduce the development risk and development cost. For this reason, two J-1 rocket external vernier engines (EVE: External Vernier Engine) are used.

(Preconditions for examination)

The candidate engine is a gas pressure engine using storable propellant and one for which the degree of completion is likely to be confirmed during the fiscal 1994; Refer to Table 4.2.1-1.

With the items below (a) through (e) taken into account, the engines, number of engines, configuration, gross mass, and structural mass were determined:

(a) The vehicle is capable of takeoff, braked drop from a position with an altitude of about 200[m] to a position with an altitude of about 100[m], hovering at a position with altitude of 100[m], and soft landing.
(b) An engine with a larger thrust is favorable because a fewer number of engines allows the experimental vehicle to be configured.
(c) The throttling feature is required for soft landing.
(d) Nozzles extension matching the atmospheric pressure are needed.
(e) To minimize the development risk and development period, an engine already developed is preferable.

(Results of examination)

The gross mass is decided on 541[kg] and the structural mass on 417[kg].
The vehicle is equipped with two J-1EVE.
Both engines have the throttling function.

4.2.2. Overall Configuration

* The two engines are installed on the bottom;
* The two tanks are vertically installed.
Refer to Figs.4.2.2-1 and 4.2.2-2.

(A) Oxidant tank and fuel tank

Regarding the relative position of the NTO tank and the N2H4 tank, a higher degree of dynamic stability is obtained with the NTO, which has a higher density, installed above;

(B) Electric and electronic equipment

To alleviate the effect of vibration and heat from the engines, electric and electronic equipment is installed as far away from the engines as possible.

4.2.3. System Configuration and Distribution of Weight and Electric Power

See Table 4.2.3-1.

The system configuration and distribution of weight and electric power are determined by referring to those for lunar landing vehicles and reusable boosters.

4.2.4. Items Concerning the Attitude Control (center of gravity, moment of inertia, product of inertia, RCS arrangement, gimbal mechanism, and so forth)

4.2.4.1. Center of Gravity, Moment of Inertia, and Product of Inertia

Values for the center of gravity, moment of inertia, and product of inertia are given by the overall arrangement described in Paragraph 4.2.2. as follows:

			Dry	Wet (maximum)
Center of Gravity	X	[cm]	85	83
	Y	[cm]	0	0
	Z	[cm]	0	0
Moment of Inertia	Ixx	[kgm2]	95	100
	Iyy	[kgm2]	350	380
	Izz	[kgm2]	350	380
Product of Inertia	Ixy	[kgm2]	0	0
	Iyz	[kgm2]	0	0
	Izx	[kgm2]	0	0

4.2.4.2. Attitude Control Methods

As is shown below, a few options for attitude control are available.

(a) Gimbal mechanism (with two 2-axis units)

(b) Gimbal mechanism (with two 1-axis units)

(c) RCSx4

With Option (a), a gimbal mechanism alone is capable of controlling all of rolling, pitching, and yawing; however, the yawing control could functionally become redundant because there is a possibility of the yawing motion being controlled by the thrust differential control;

With Option (b), a gimbal mechanism alone can principally control all of rolling, pitching, and yawing; regarding the yaw control, however, there is a possibility of the vehicle being overturned against strong cross wind if the variable thrust response speed is too small or the thrust/weight ratio is too small.

With Option (c), duplication of pitch control thrusters could reduce the control force; in addition, the attitude control speed cannot be increased much, and it would be very difficult to handle the deviation of the airframe center of gravity.

With emphasis placed on the capability of addressing disturbances, Option (a) is adopted as the primary option; however, the results of future detailed analysis of controllability may lead to switching to the adoption of Option (b).

4.2.4.3. Attitude Requirements

The inclination of the airframe axis against disturbances is kept within 5[deg].

4.2.4.4. Responsiveness Requirements

(a) Gimbaling requirements (T.B.D.)

Gimbaling angular velocity	(T.B.D.)
Delay time	(T.B.D.)

(b) Throttling requirements

Throttling changing rate	(T.B.D.)
Delay time	(T.B.D.)

4.2.5. Requirements for Velocity/Attitude at the Time of Landing

(a) Horizontal velocity/attitude angle

Fig.4.2.5-1 shows the maximum horizontal velocity for the airframe not to be overturned which is obtained under the condition in which the airframe angular velocity immediately before striking is assumed to be zero. With a 25% margin added to the results obtained, the horizontal velocity and inclination of the airframe axis are set at the following values:

When the inclination of the airframe axis is assumed to be 5[deg] at the maximum, the horizontal velocity is not more than 0.7[m/s]; and

When the inclination of the airframe axis is assumed to be 1[deg] at the maximum, the horizontal velocity is not more than 1.0[m/s].

(b) Horizontal velocity/angular velocity

Fig.4.2.5-2 shows the maximum horizontal velocity for the airframe not to be overturned which is obtained under the condition in which the inclination of the airframe axis immediately before striking is assumed to be zero. With a 25% margin added to the results obtained, the horizontal velocity and airframe angular velocity are set at the following values:

When the horizontal velocity is assumed to be 0.7[m/s] at the maximum, the pitch/yaw angular velocity is not more than 45[deg/s]; and

When the horizontal velocity is assumed to be 1.0[m/s] at the maximum, the pitch/yaw angular velocity is not more than 8[deg/s].

4.2.6. Aerodynamic Effects (T.B.D.)

From the technological and cost viewpoint, the flying velocity of the experimental vehicle is limited so that aerodynamic fairing may not be necessary and the flight pattern is modeled after that of a lunar landing vehicle.

Flying velocity About 20[m/s]

Approximate aerodynamic center About 10[cm] above the center of the gravity

However, the vehicle is designed so as to be capable of responding properly to gusts.

4.2.7. Reusability

The combustor is capable of being used over 20 times.

The tank is capable of being used over 50 times.

The life of the air reservoir is determined depending on the maximum working pressure.

The thermal protection is used over 10 times.

4.2.8. Electric Equipment System

Fig.4.2.8-1

Fig.4.2.8-2

4.2.9. System Ratings

Table 4.2.9-1

Fig.4.2.9-1

The equipment and the systems are not redundantly configured except the emergency command system (engine stop command/forced descent in an emergency)

4.2.10. Ground Interfaces

Main Interfaces:

Structure system	Hard point (for sling/handling)	
Thermal control system	Air conditioning (for onboard electronic and electric equipment) inlet and outlet	
Propulsion system	Fuel load/unload openings	
	Oxidant load/unload openings	
	Pressure gas charge/discharge openings	
	Function checkout	Communications umbilical
Navigation guidance and control system	Function checkout	Communications umbilical
Communications system	Function checkout	Communications umbilical
Electric power system	Charging of the onboard battery	Power umbilical
	Supply from external power supply	Power umbilical

* Outer dimension

Total height	325cm
Total width	140cm
Airframe diameter	110cm
* Structural mass	417kg
* Gross mass	541kg
Propellant mass	124kg
NTO	60kg
N_2H_4	60kg
GHe	4kg
* Average electric power	1149W (Nominal), 1846W (Peak)

* Umbilical line/external interface

1) Air conditioning duct	1 pc.	GL+2.5m
2) Air conditioning outlet	2 pcs.	GL+1.3m
3) Pressure gas charge inlet	1 pc.	GL+1.5m
4) Pressure gas discharge outlet	2 pcs.	GL+1.5m
5) Hydrazine charge inlet	1 pc.	GL+1.5m
6) NTO charge inlet	1 pc.	GL+1.5m
7) Hydrazine discharge outlet	3 pcs.	GL+1.5m
8) NTO discharge outlet	2 pcs.	GL+1.5m
9) Signal and electric power umbilical	1 pc.	GL+2.0m

(All of these items are manually removed before launching.)

* With the surface shell removed, access to the entire periphery of the airframe is secured using a platform for servicing the airframe.

4.3. Systems Analysis

Tentative Examination of Flight Capability/Flight Performance

(a) Vertical/horizontal flight capability

Fig.4.3-1

(b) Hovering flight capability

Fig.4.3-2

Table 4.2.1-1 Comparison between various engines

	J-1 EVE Combustor		LAPS, UPS AKE		550[N] For Lunar-A		1200[N] Subscaled model for HOPE		500[N] For PLANET-B	
Propellant	NTO/N2H4		NTO/N2H4		NTO/N2H4		NTO/MMH		NTO/N2H4	
Thrust	2,900 at SL	○	2,000 in vacuum	○	550 in vacuum	×	1,200 in vacuum	△	500 in vacuum	×
Specific impulse	213 at SL	○	320 in vacuum	○	317 in vacuum	○	322 in vacuum	○	312 in vacuum	○
Weight [kg] (engine alone)	7	○	6	○	7.4	○	8	○	4.5	○
Nozzle	For use under the atmospheric pressure	○	For use in a vacuum	×	For use in a vacuum	×	For use in a vacuum	×	For use in a vacuum	×
Throttling capability [%]	55~100	○	60~100	○	100% only	×	89~100	△	100% only	×
Development status	1994FY Development completed in the fiscal 1994	○	QUALIFIED	○	Flight scheduled in 1997	△	For research purpose only	△	Flight scheduled in 1998	△

Overall evaluation	○	△	×	×	×

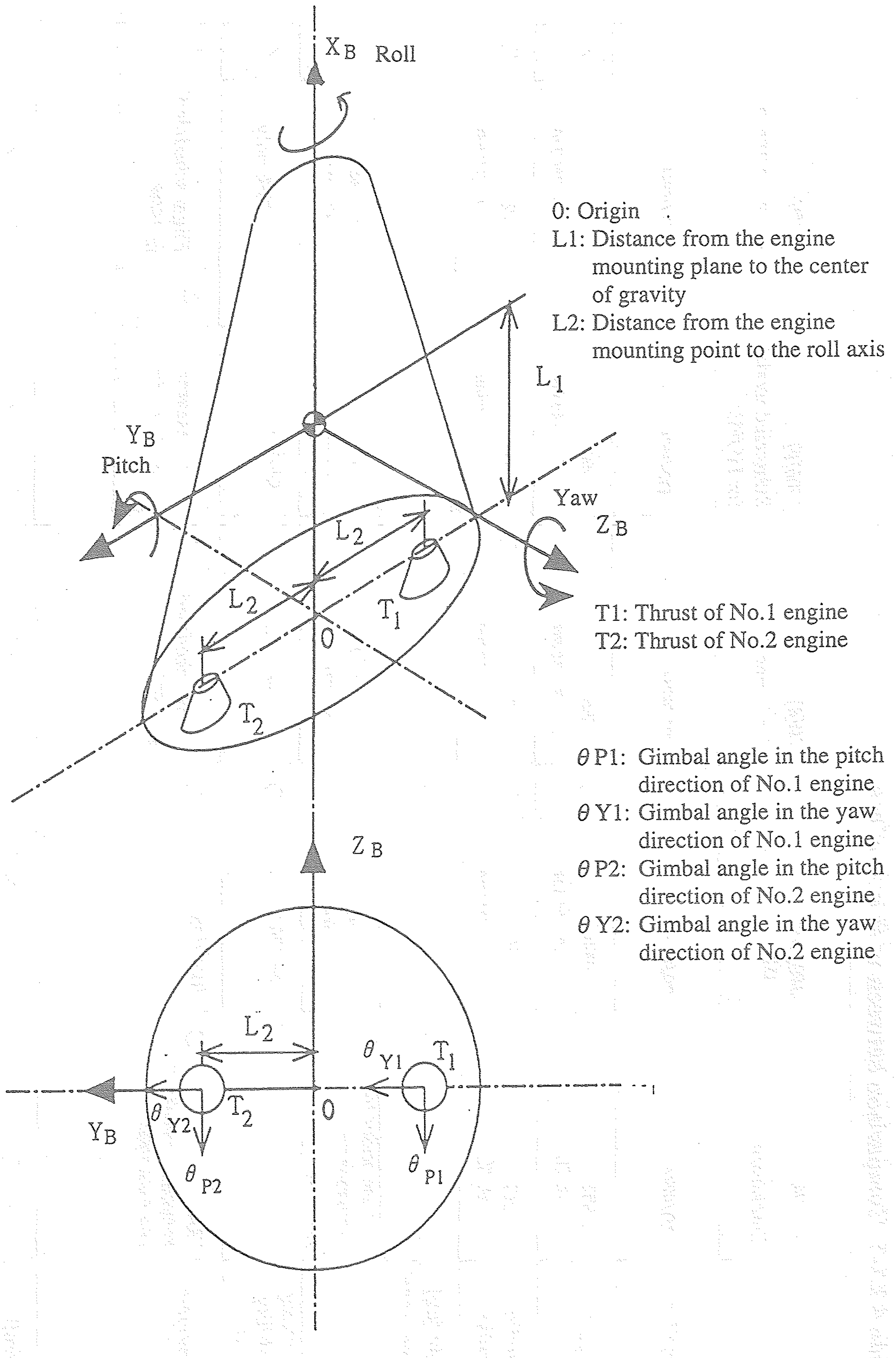


Fig.4.2.2-1 Airframe model image (two-axis gimbal)

An option for discussion

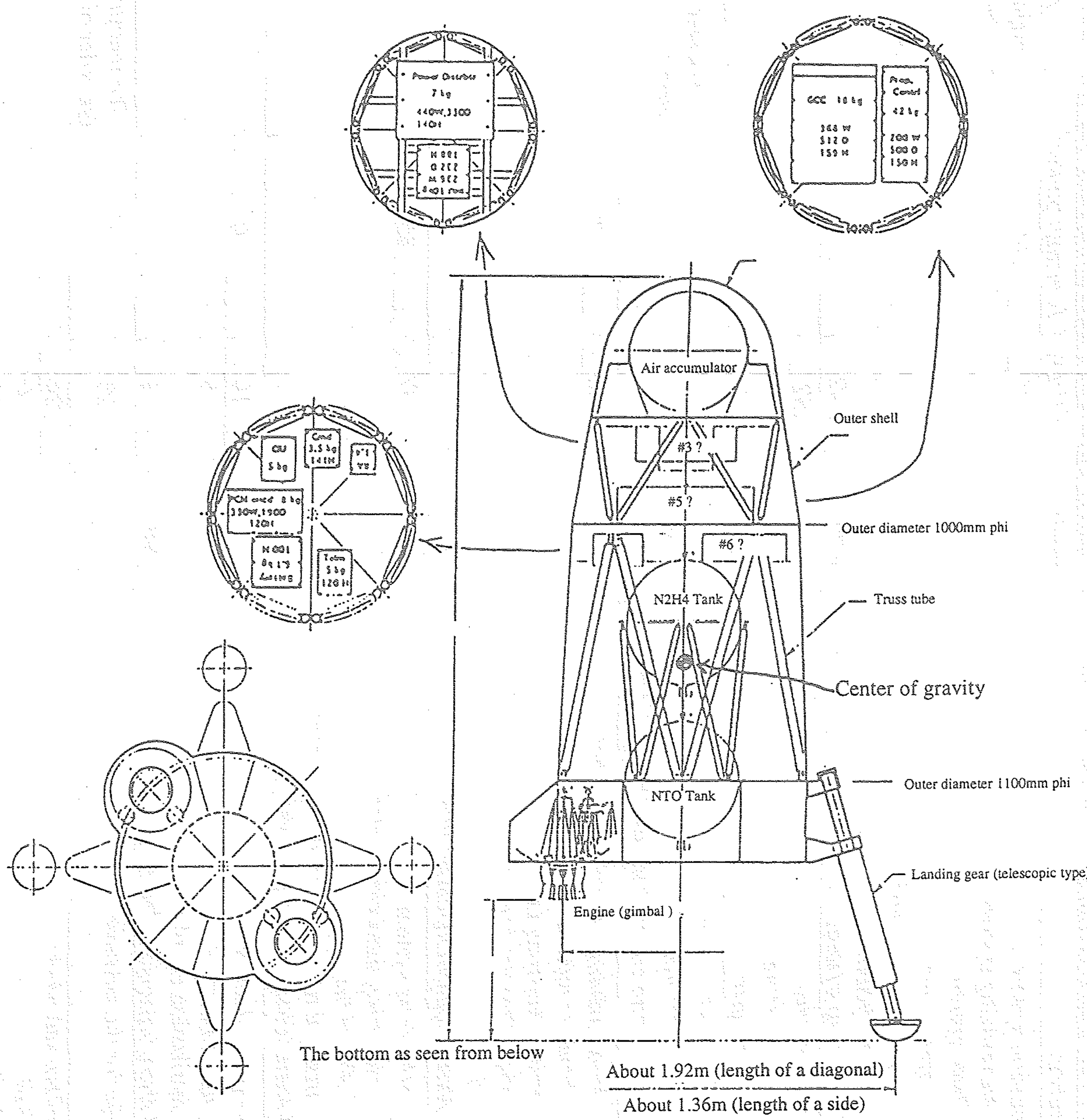


Fig.4.2.2-2 Overall arrangement

Table 4.2.3-1 Plan for takeoff and landing flight experiment   Airframe system configuration, weight, and electric power

System configuration		Quantity	Weight	Average power	Remarks
Airframe system Dry weight 417kg Gross weight 541kg Average power 1200W Peak power 2000W	Propulsion system		184	914	The weight of the propulsion system does not include a margin of 3kg. The peak power for the propulsion system is 1611W.
	* Pressurizing system	1 set	27		
	* Tank system	1 set	40		
	* Engine system	1 set	24		
	* Thrust control system	1 set	26		
	* Gimbal system	1 set	25		
	* Controller system	1 set	42		
	Structure system		132	0	
	* Main structure	1 set	85		
	* Landing gear	4 sets	50		
	Thermal control system		7	0	The power for the heat control system is included in the margin.
	* Flame protection	1 set			
	* Temperature control/temperature measurement	1 set			
	Navigation guidance and control system		38	150	
	* Onboard computer	1	18		
	* Inertial measuring unit	1	10		
	* Radio altimeter/antenna	1	2		
	* CIU	1	5		
	* Others (connectors/harnesses)	1 set	3		
	Communications system		20	85	
	* Telemetering transmitter/antenna	1	5		
	* PCM encoder	1	8		
	* Command receiver/demodulator/antenna	1	4		
	* Others (connectors/harnesses)	1 set	3		
	Power supply system		16	0	
	* Onboard battery	1	6		
	* Distribution board	1	7		
	* Others (connectors/harnesses)	1 set	3		
	Margin for the airframe		20	51	The margin for the airframe is in the order of 5%.
	Propellant and others		124		
	* Fuel		60		
	* Oxidant		60		
	* Pressurizing gas		4		

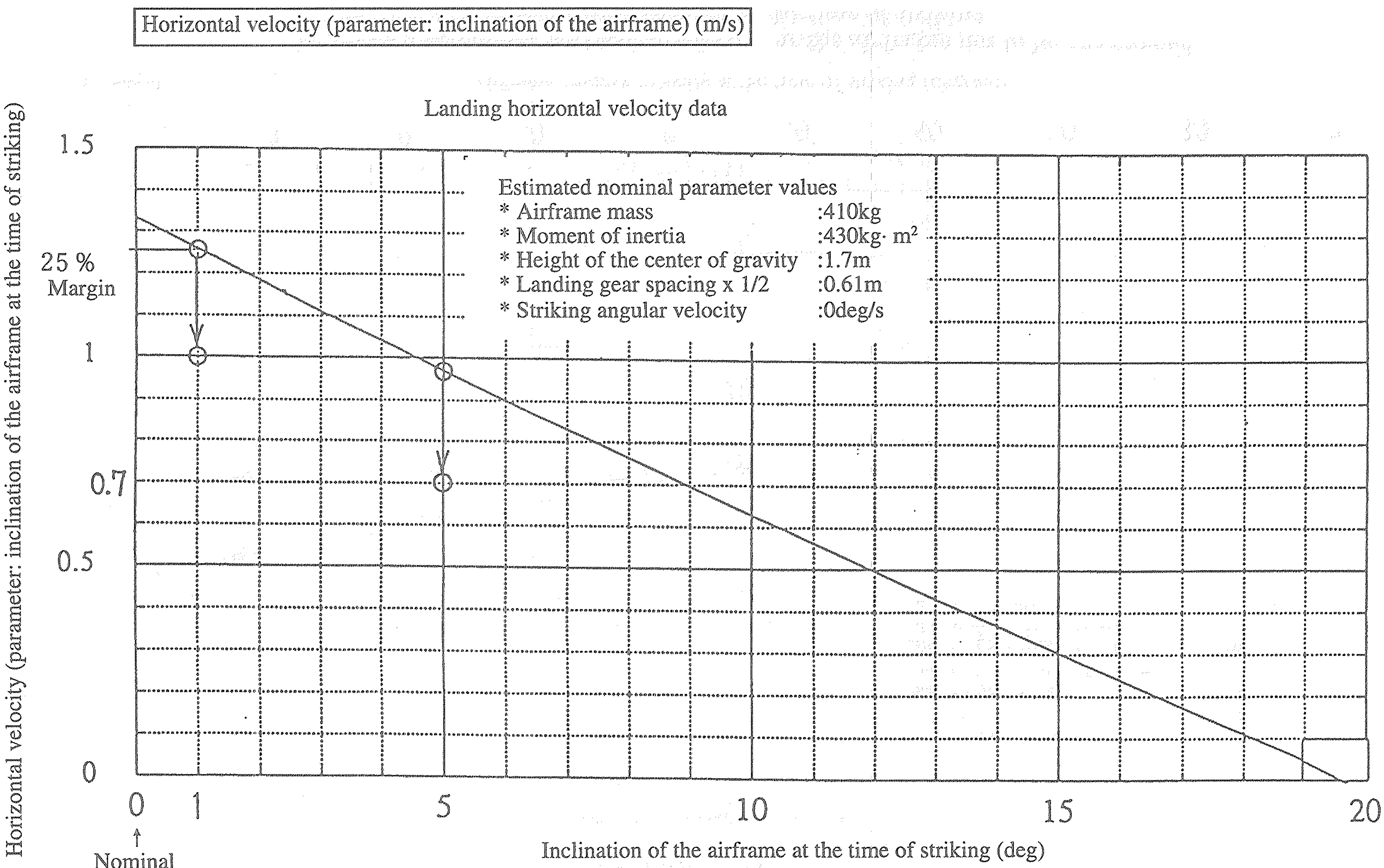


Fig.4.2.5-1 Maximum horizontal velocity for the airframe not to be overturned
Parameter: Inclination of the airframe at the time of striking

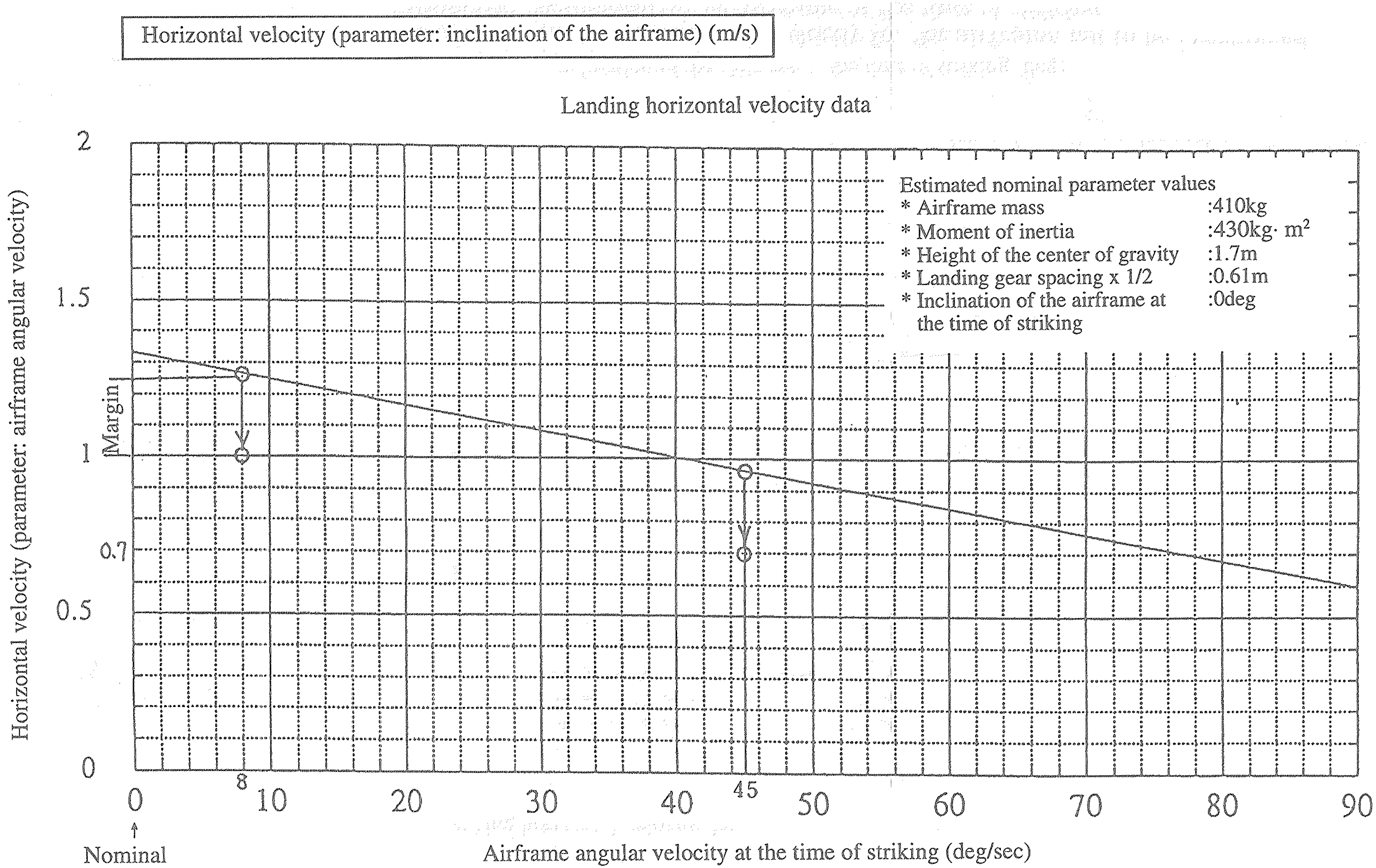


Fig.4.2.5-2 Maximum horizontal velocity for the airframe not to be overturned
Parameter: Airframe angular velocity at the time of striking

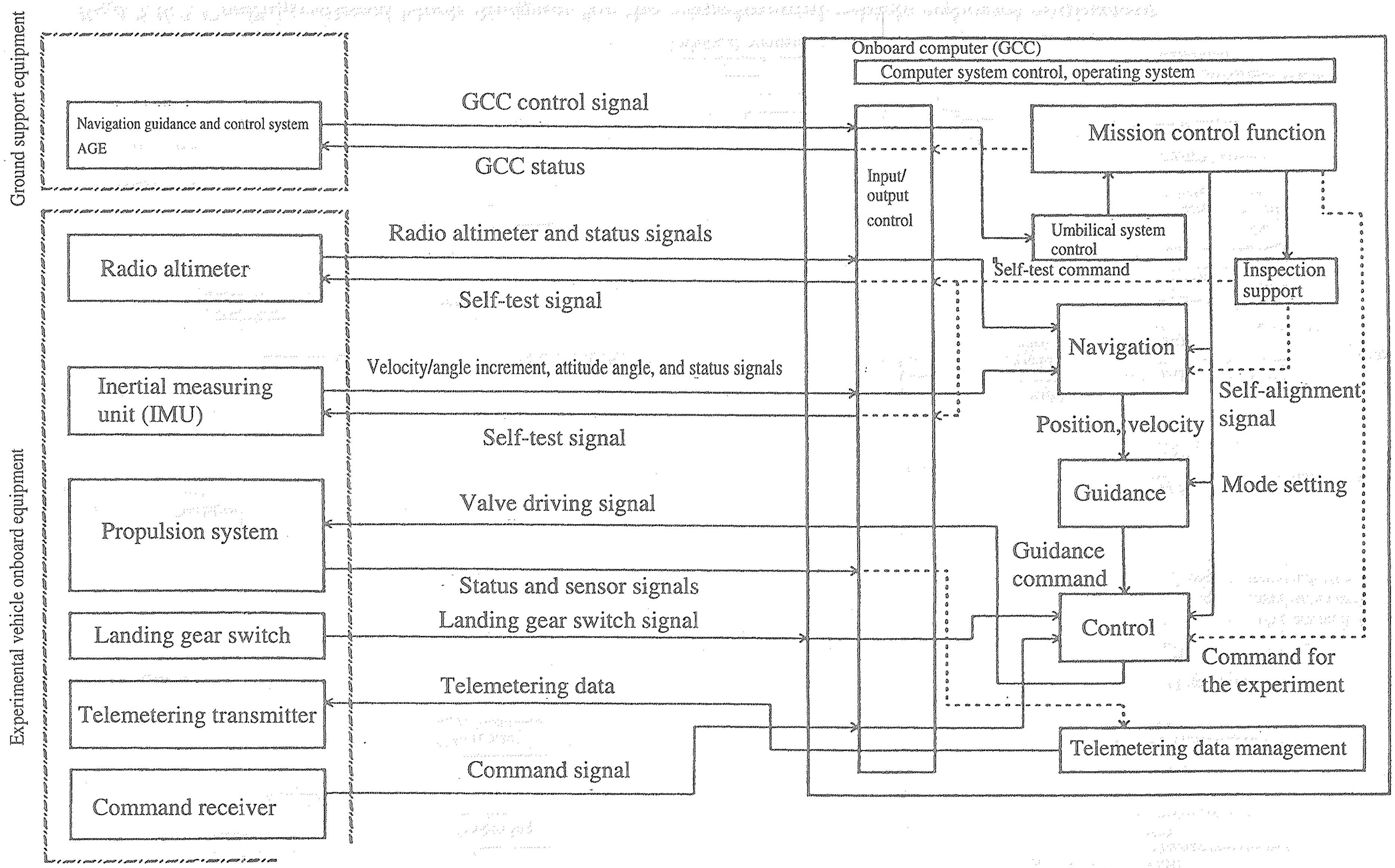


Fig.4.2.8-1 Functional block diagram for the experimental vehicle onboard systems

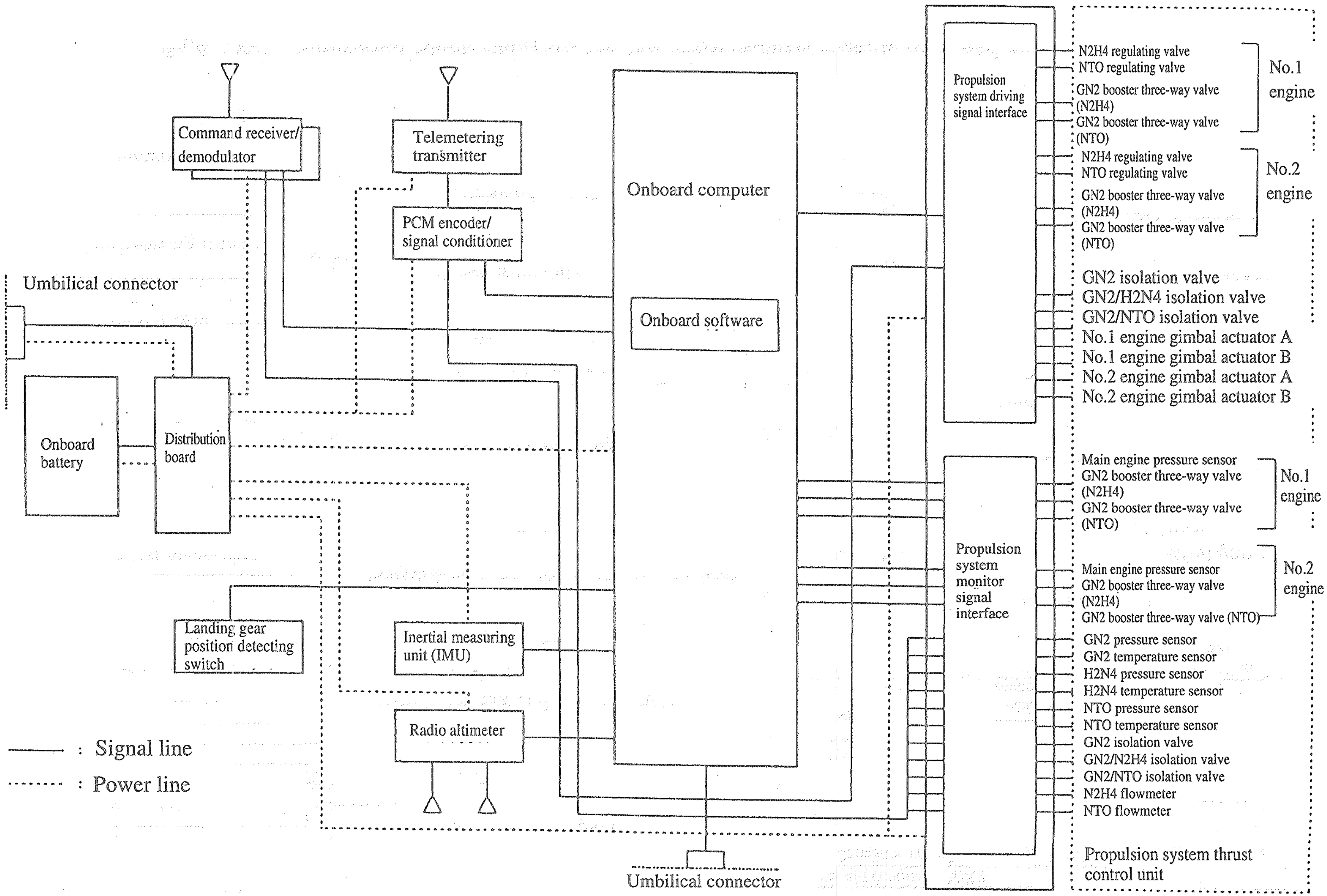


Fig.4.2.8-2 Configurational block diagram for the experimental vehicle onboard equipment

Table 4.2.9-1 Outline of the system

(A) Weight	Gross mass 541[kg] Structural mass 417[kg]
(B) Propulsion system	Engine Number of engines installed 2 Propellant NTO/N2H4 Thrust F= not less than 280[kgf] at SL Specific impulse Isp=not less than 200[s] at SL Throttling capacity 60 - 100[%] (for both engines) Total thrust range 560[kgf] (both engines at 100% thrust) 336[kgf] (both engines at 60% thrust) Gimbal 2 axis for both engines Equipped with no RCS
(C) Airframe outer dimension	Overall height 3.3[m] Overall width 1.4[m] (with the stationary state landing gear spacing) Airframe diameter 1.1[m]

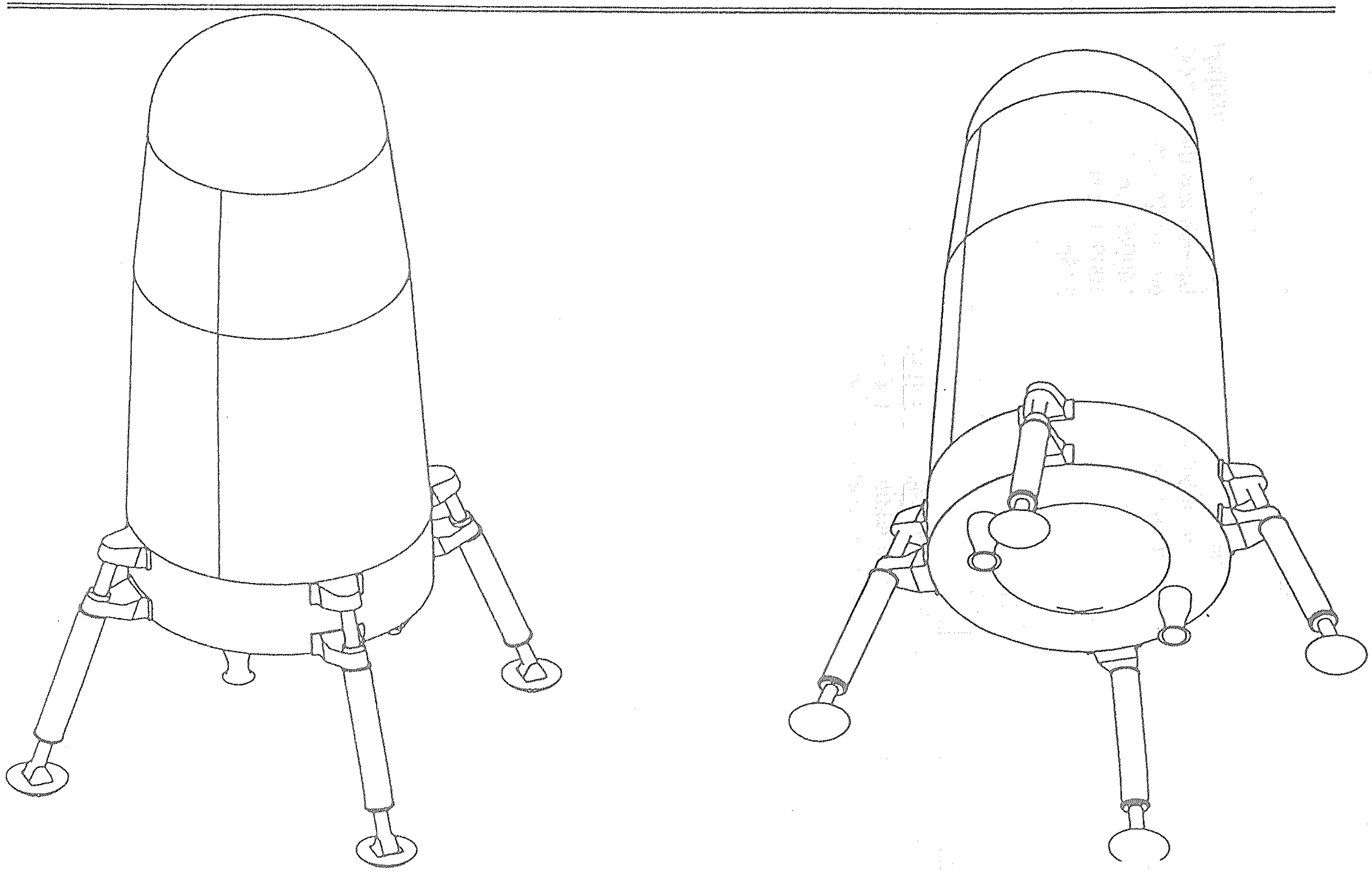
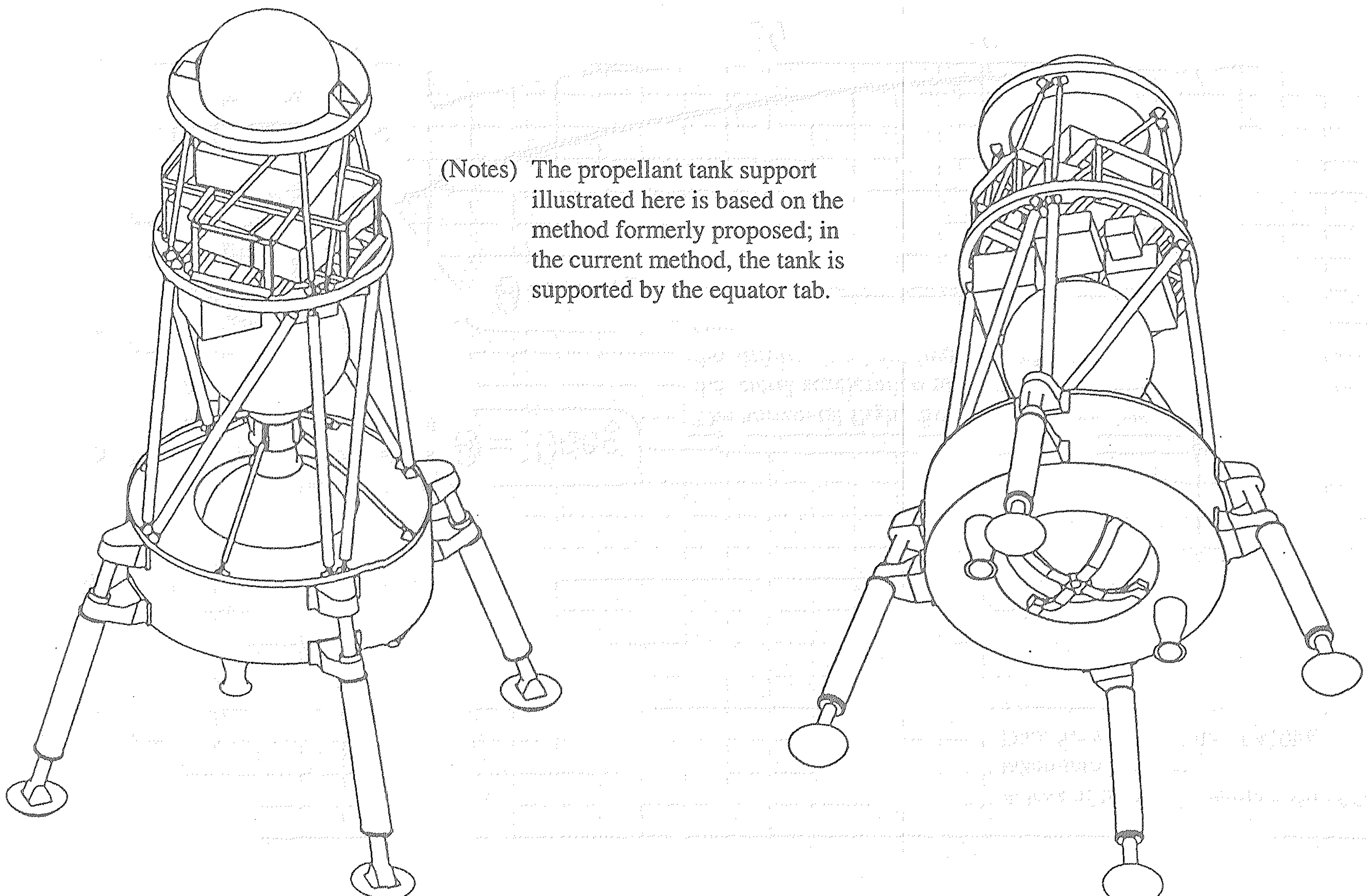


Fig.4.2.9-1 Diagrammatic sketch of the airframe (1/2)



(Notes) The propellant tank support illustrated here is based on the method formerly proposed; in the current method, the tank is supported by the equator tab.

Fig.4.2.9-1 Diagrammatic sketch of the airframe (2/2)

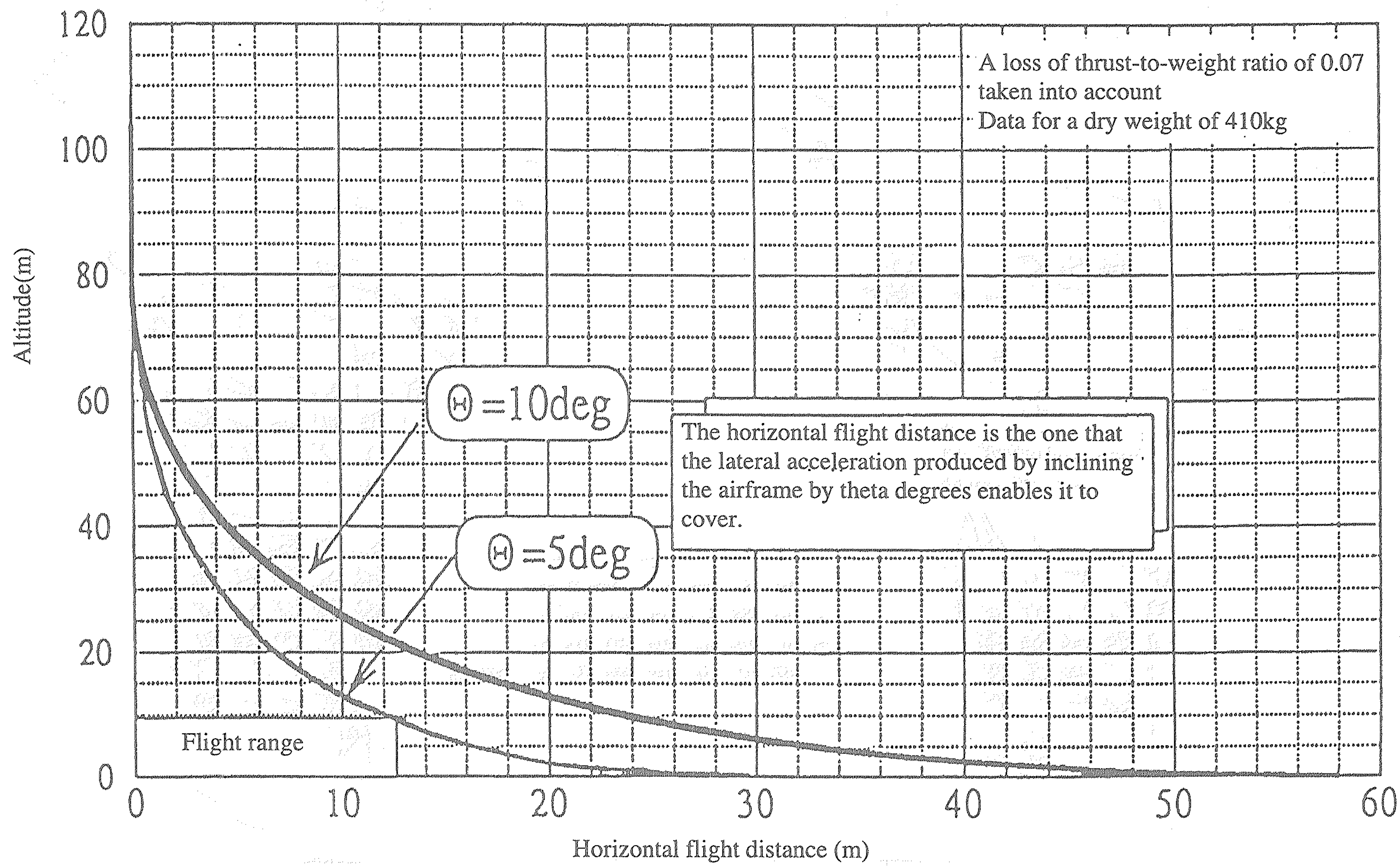


Fig.4.3-1 Flight range

Plan for takeoff and landing flight experiment   Hovering time and attained altitude (hovering capability)

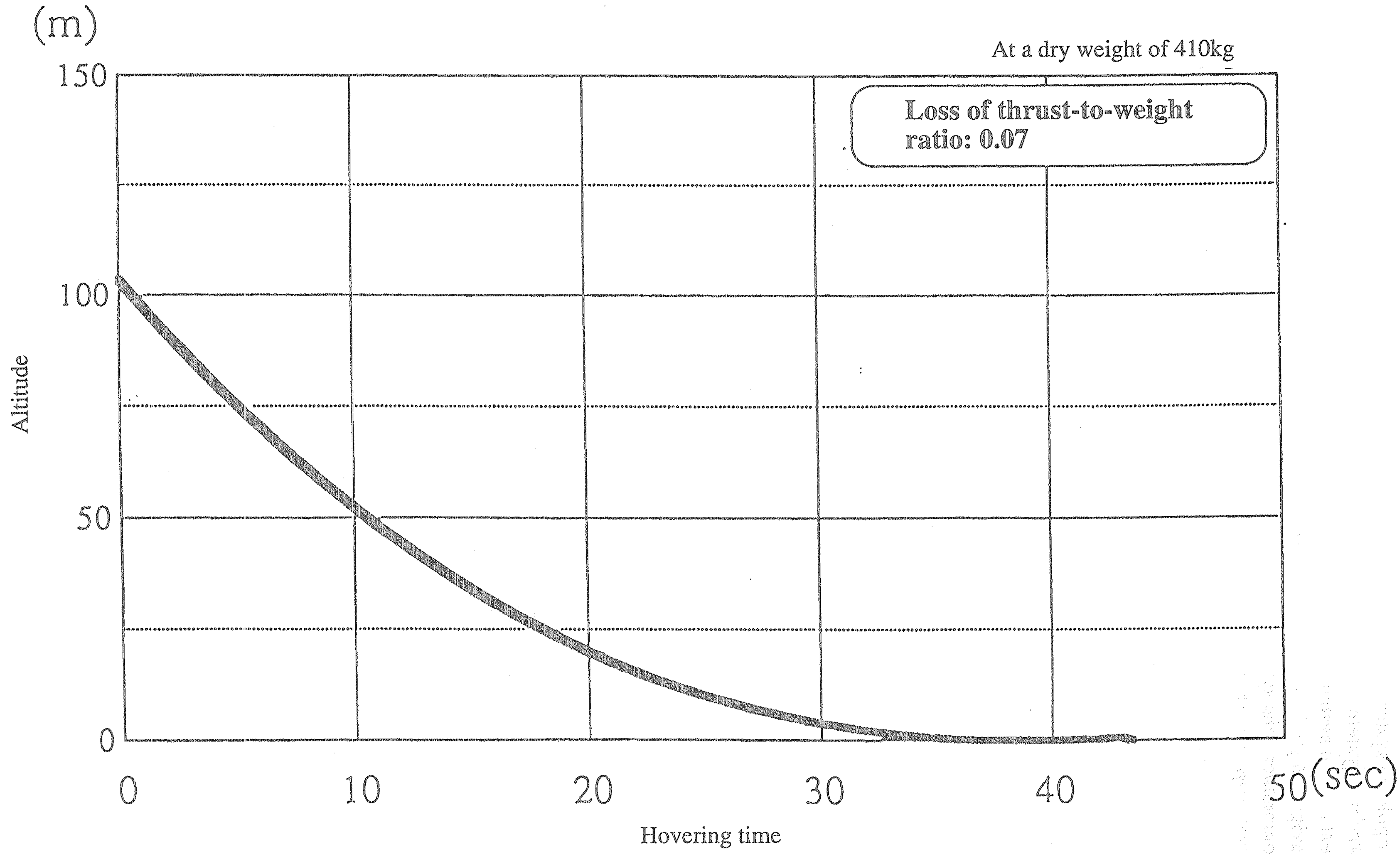


Fig.4.3-2 Hovering time and attained altitude

5. Examination of Subsystems

Propulsion system
Structure system
Heat control system
Navigation guidance and control system
Communications system
Power supply system

The following are the results of the examinations fabricated on the subsystems enumerated above:

5.1 Propulsion System

5.1.1 Development Policies

The development policies for the takeoff and landing experimental vehicle are as follows:

(1) Attempts to modularize the propulsion system (shortening of the development phase, cost reduction, and improvement of maintainability);

(2) Maximum utilization of divertible components from among those that have been already used as development test models and the like for development programs of other rockets and satellites (cost reduction);

(3) Maximum utilization of consumer use components that are proven in facility applications to bypass the development and test of components as a rule (cost reduction);

(4) Use of proven components in high pressure sections such as the pressure feed system (risk reduction);

(5) Vibration tests at the component and the subsystem level are not conducted but verification is fabricated through CFT (cost reduction);

(6) Sufficient consideration is given to workability in loading and unloading propellant (maintainability).

5.1.2 Required Specifications

(1) Preconditions

Two main engines for J-1 EVE are used.

Basic ratings of J-1 EVE

Propellant NTO/N2H4

(2) Required specifications

After the systems examination, the specifications required of the propulsion system for the vertical takeoff and landing experimental vehicle have been decided on as follows:

Thrust 560 - 336 [Kgf] (for 2 engines, at sea level)

Responsiveness not more than 0.7s

(time required for the change from a thrust of 560 to 336[Kgf])

Gimbal Angle not less than ±7[deg]

Angular velocity not less than 6[deg/s]

Environmental conditions In the direction of the airframe axis 3G

In the direction perpendicular to the airframe axis 1G

Life not less than 90s x 20 times

5.13 Examination of the Propulsion System

(1) Configuration of the propulsion system

The pressure control method, which forms the basis of the throttling engine system now being developed, is adopted and additionally provided with the gimbal function. Table 5.1-1 shows the configuration and functions of the propulsion system.

(2) Overall arrangement

Fig.5.1-1 shows the overall appearance of the propulsion system built up on the basis of the following points:

(a) Attempts to modularize the propulsion system. (adjusting and testing the operations of the propulsion system alone by using CFT is required; it is also required to develop the propulsion system mounting structure at the same time.)

(b) Convenience in assembling the propulsion system

(c) Important components such as the tank and flow control valves are not mounted on the engine mount plane.

The overall arrangement depends on the tank arrangement greatly; Table 5.1-2 shows the results of the trade-off between the number of propellant tanks and their arrangement; on the basis of the results, the arrangement of 2 tanks placed in tandem has been adopted.

(3) System diagram

Fig.5.1-2 shows the system diagram for the propulsion system drawn from the viewpoint of the maintainability in loading and unloading the propellant.

(4) Electric system for the propulsion system

Fig.5.1-3 provides a functional block diagram for the electric system for the propulsion system. The controller system has an interface function with the propulsion system, navigation guidance and control system, and measurement and communications system; the vertical takeoff and landing experimental vehicle is the first to be equipped with a controller including the thrust control and gimbal control functions. The flow control valve driver and gimbal actuator driver add to the vehicle weight if used as they are, some modification is added to them to reduce their weight.

(5) Mass characteristics

Table 5.1-3

(6) Power consumption

Table 5.1-4 shows the summation of the power consumption; it is allowable power consumption from the viewpoint of systems operation.

5.1.4. Specifications for the Propulsion System

The propulsion system is equipped with 2 engines.

5.1.4.1. Pressurizing System

Requirements: The pressurizing system is capable of supplying gas to the tank pressuring and pneumatic mechanisms.

Because the system's pressure is high, the HYFLEX pressure control valves are used to reduce risk and cost.

For the accumulator, EM items of COMETS/UPS are diverted.

Capacity	: 57.9[l]
Initial charging pressure	: 140[kgf/cm2a] (T.B.D.)

5.1.4.2. Tank System

Requirements:

The tank system has a propellant retaining function.

Examination:

A Ti alloy tank is used for NTO and N2H4 each.

To fabricate the tanks, the superplasticity (SPF) method, which has been proven in the fabrication of the tanks for H-2G/J, is used to reduce cost. Although a higher safety factor is desirable, it seems that a safety factor value of 2 is sufficiently consistent with a life of 50 times and the weight reduction requirement. Because the High Pressure Gas Regulation is applicable, coordination with the authorities concerned becomes necessary.

Fig.5.1-4

Materials	Ti-6Al-4V
Configuration	Bare tank
	Baffle plate
	Aireddy prevention plate
	Diffuser

5.1.4.3. Engine System

Requirements: The engine system has a thrust and Isp required to control the flight trajectory.

Engines equivalent to the combustor for J-1 EVE are used.

Number of engines	2		
Isp	not less than 200[s]	at SL.	Refer to Fig.5.1-5.
Thrust	not less than 280[kgf]	at SL.	Refer to Fig.5.1-6.

The nozzle extension ratio is determined so that the gas expands in an optimum condition at an atmospheric pressure of 1[kgf/cm2a].

$\varepsilon = 2.0$ (T.B.D.)

Throttling

The thrust is controlled by changing the flow rate of the propellant with the flow control valve. By giving feedback on the combustion pressure, the 2 flow control valves on the fuel and the oxidizer line are properly controlled.

Fig.5.1-7

Fig.5.1-8

Fig.5.1-9

Life

In the order of 90[s] x 20 times (T.B.D.)

Misalignment T.B.D.

5.1.4.4. Gimbal System

Requirements: The gimbal system has speeds of response required to cancel aerodynamic misalignments and shift in the center of gravity and to control the airframe actively.

Configuration	Engine gimbal assembly	1 set
	Motor-operated actuator	2 sets
	Engine gimbal mount	1 set
	Actuator driver	1 set

Required ratings for the gimbal actuator

Gimbal	
Type of gimbal	Pivot ball mount type
Maximum gimbal angle	±7[deg]
Gimbal velocity	6[deg]/s

Fig.5.1-10

5.1.4.5. Safety System

Should an abnormality occur to the experimental vehicle in some way or other, the safety system is capable of securing safety of the surrounding environment and facilities.

The propulsion system controller interfaced directly with the telemetry command system makes it possible for commands from the ground to cause the propellant valve or flow control valve to be shut off to shut down the engine.

5.1.5 Assembling and Testing of the Propulsion System

Fig.5.1-11 shows the flow in the assembling and testing of the propulsion system. In order to reduce cost, subsystem tests are not conducted; tests are conducted on the entire propulsion system only.

5.1.6 Development Plan and Component Procurement Method for the Propulsion System

Table 5.1-5 shows the development plan for the propulsion system; Table 5.1-6 shows the component procurement method for the propulsion system.

Regarding the technology acquisition for the propulsion system, the throttling technology is to be acquired through the throttling combustion test using an engine while the 2-engine throttling and the cluster technologies are to be acquired through the cluster combustion test to be performed in the fiscal 1996. The propulsion system for the vertical takeoff and landing experimental vehicle will be developed on the basis of the technologies mentioned above that help reduce development risk.

To reduce cost, in addition, the components for the throttling combustion and cluster combustion tests are diverted, as extensively as possible, to the vertical takeoff and landing experimental vehicle.

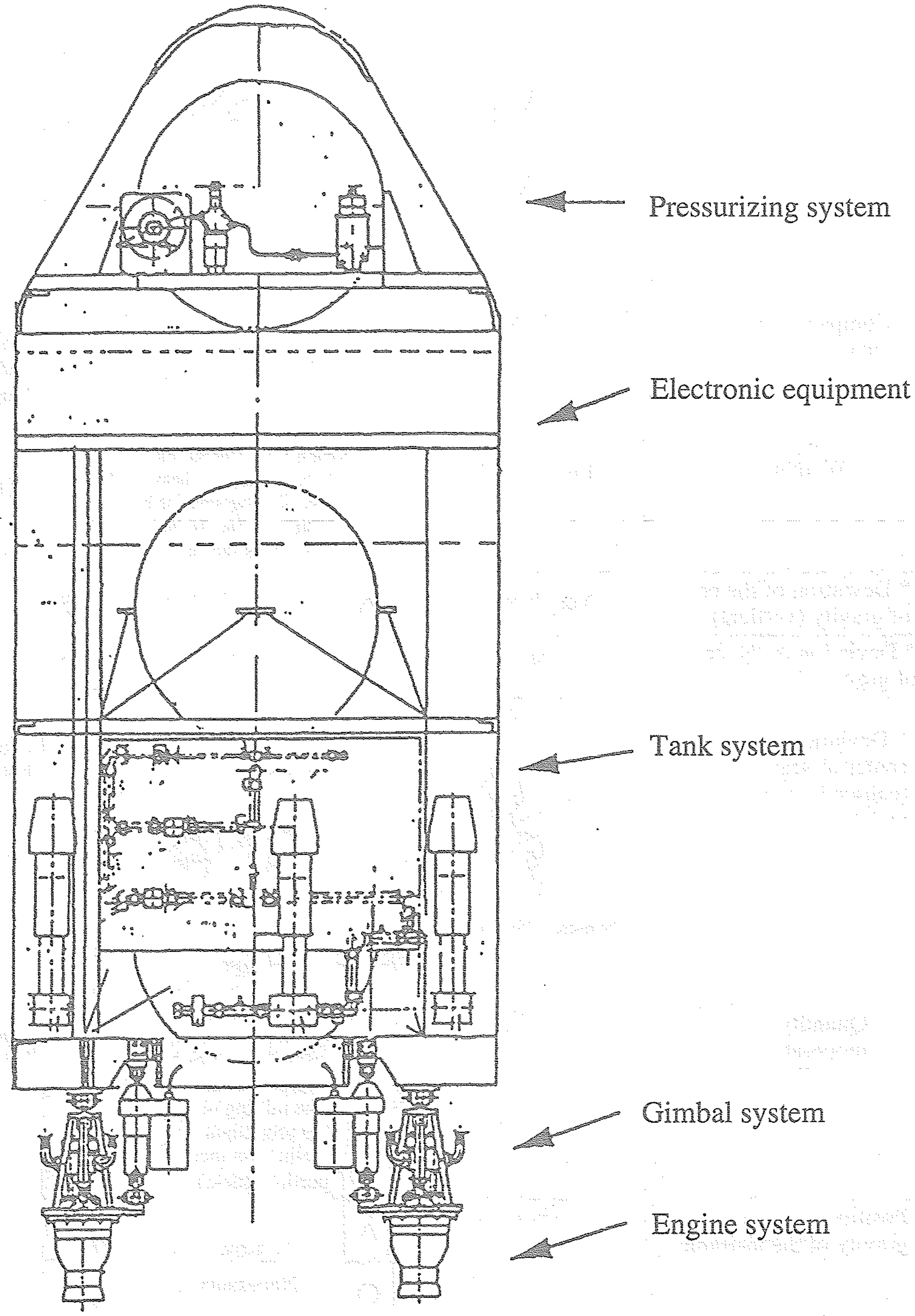
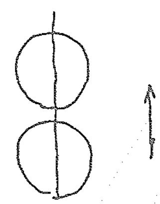
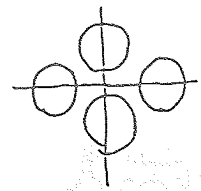
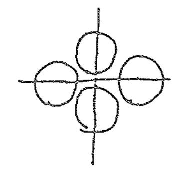
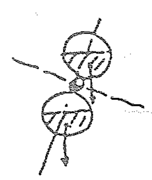
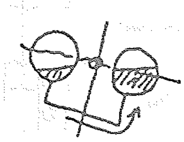
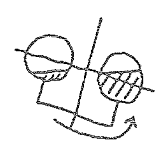
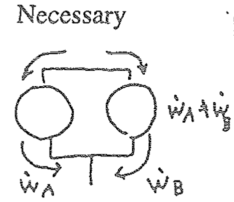


Fig.5.1-1 Overall appearance of the propulsion system
(Ground Test Configuration)

Table 5.1-2 Results of the trade-off between the number of propellant tanks and their arrangement

Trade-off items	Tandem (fabricated of Ti alloy)		Horizontal opposed position (fabricated of Ti alloy)		Horizontal opposed position (fabricated of Al alloy)	
Arrangement/number of tanks	 2 tanks		 4 tanks		 4 tanks	
Compatibility of the material with the propellant	Ti－6Al－4V Performance proven	○	Ti－6Al－4V Performance proven	○	Al alloy (compatibility with NTO under the condition of a long period of use)	?
Weight	Light weight	○	Compared with the system using only 2 tanks, the weight is slightly heavier. (by several kilograms due to addition of the weight of accessories such as tabs.)	△	The same as the left	△
* Deviation of the center of gravity (vertical)	Yes (large ?)	△	Yes (small ?)	○/△	Yes (small ?)	○/△
* Deviation of the center of gravity (horizontal)	No	○	No	○	No	○
* Deviation of the center of gravity (caused by the inclination of the airframe)	Small  The restoring force generated ?	○	Large (movement of the fluid between the tanks) 	△	Large (movement of the fluid between the tanks) 	△
Quantity of inoperative propellant	Small	○	Large (movement of the fluid between the tanks) (the portion of the propellant stuck to the tank is 1.25 times more.) (the mixing of gas into the propellant due to inclination increases the portion stuck.)	△	Large (movement of the fluid between the tanks)	△
Position of the center of gravity of the airframe	High	△	Low	○	Low	○
Pressure adjustment between the tanks (discharging characteristics of the propellant)	Not necessary	○	Necessary 	△	Necessary The same as the left	△
Overall evaluation	○		×		×	

(Note) Although the satellite use propulsion system uses multiple tanks, sufficient measures to counter the problems described above have been taken.

* Depends on the position of the center of gravity.

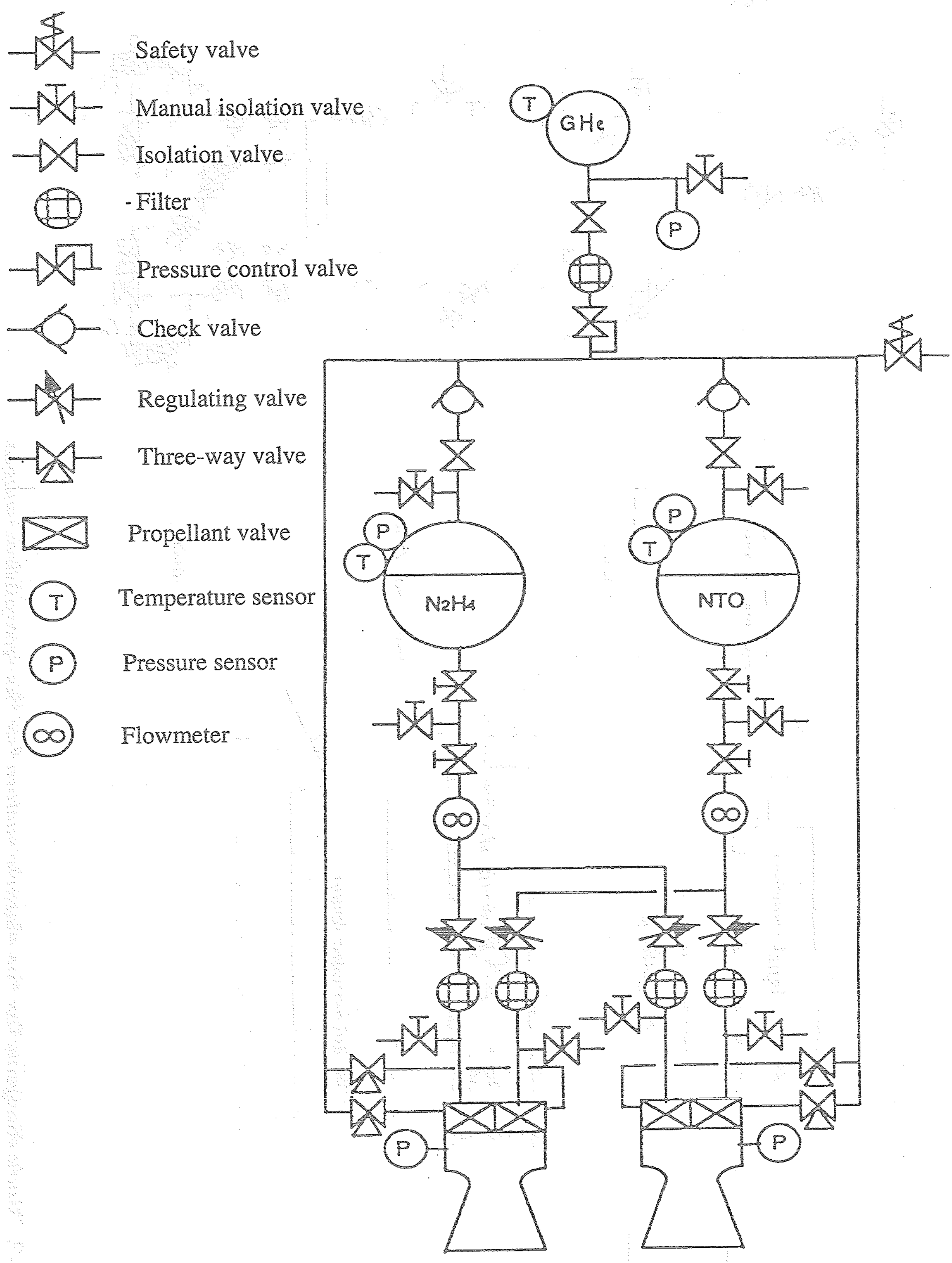


Fig.5.1-2 System diagram

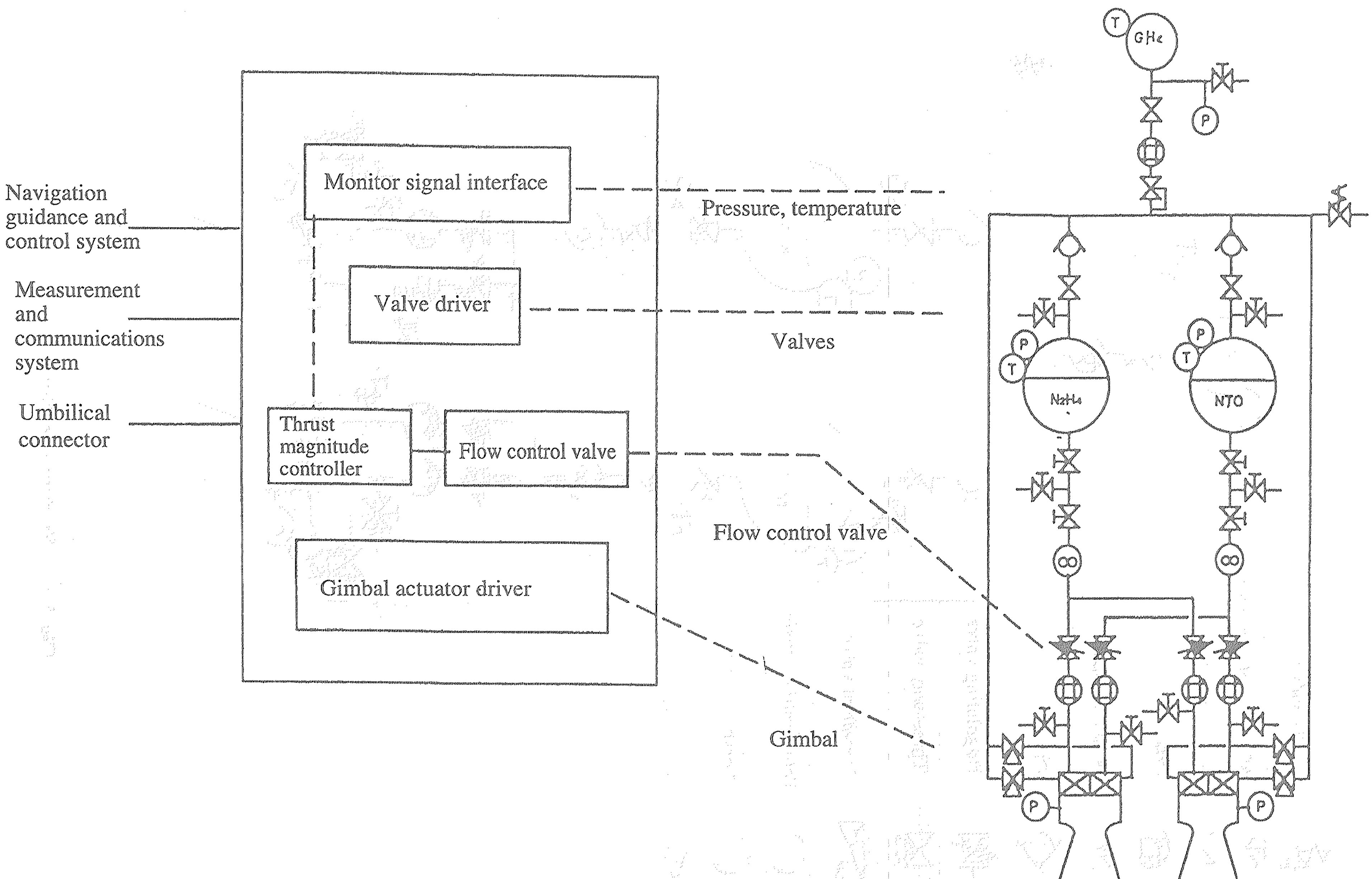


Fig.5.1-3 Block diagram for the electric system for the propulsion system

Table 5.1-3 Weight of the propulsion system components

		Ground plan	Results of the 1st review
Propulsion system	Pressurizing system	26	27
	Tank system	50	40
	Engine system	44	24
	Thrust control system	-	26
	Gimbal system	48	25
	Controller system Thrust magnitude controller Flow control valve driver Gimbal actuator driver Valve driver, CIU	-	42 (17) (12) (12) (1)
	Dry weight	168	184
	N2H4 weight	60	60
	NTO weight	60	60
	Pressure gas weight	10	4
	Total weight	298	308

Table 5.1-4 Power consumption of the propulsion system components

	Normal power consumption (per unit)	Maximum power consumption (per unit)	Number of units	Normal power consumption (total)	Maximum power consumption (total)	Remarks
Pressure feed system						
High-pressure pressure sensor	8W	8W	1	8W	8W	
High-pressure isolation valve	0W	47W	1	0W	47W	Latch valve
Pressurizing isolation valve	0W	9W	2	0W	18W	Latch valve
Propellant supply system						
Low-pressure pressure sensor	8W	8W	2	16W	16W	
Flowmeter	5W	5W	2	10W	10W	
Regulating valve	0W	160W	4	320W	640W	At the maximum load
Engine system						
Three-way valve	37W	37W	4	312W	624W	
Gimbal system						
Actuator	0W	156W	4	312W	624W	
Controller system						
Thrust controller	50W	50W	2	100W	100W	
				914W	1611W	

Addition of heater power may become necessary depending on the environment.

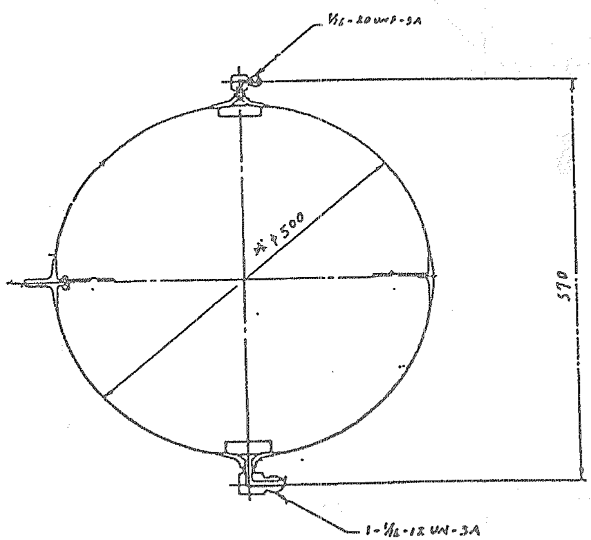


Fig.1 ø 500 NTO tank

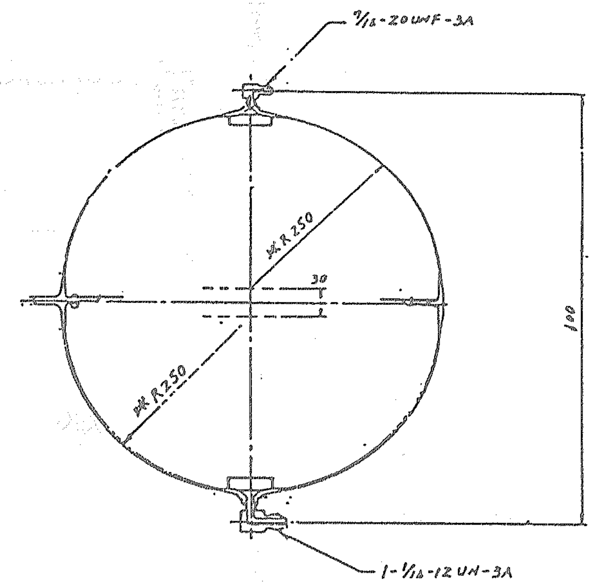


Fig.2 ø 500 N2H4 tank

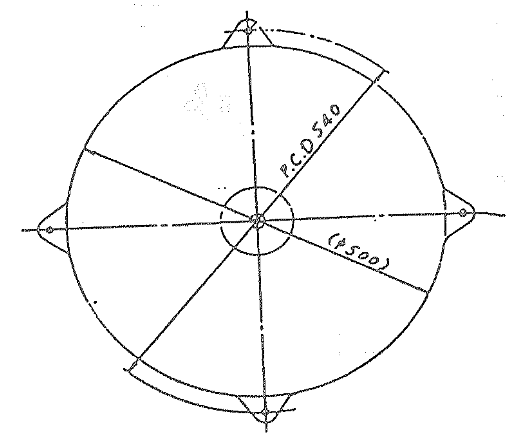


Fig.3 ø 500 NTO/N2H4 tank

Fig.5.1-4 Major dimension of the tank

Calculation of the optimum expansion ratio for J-1 EVE combustor

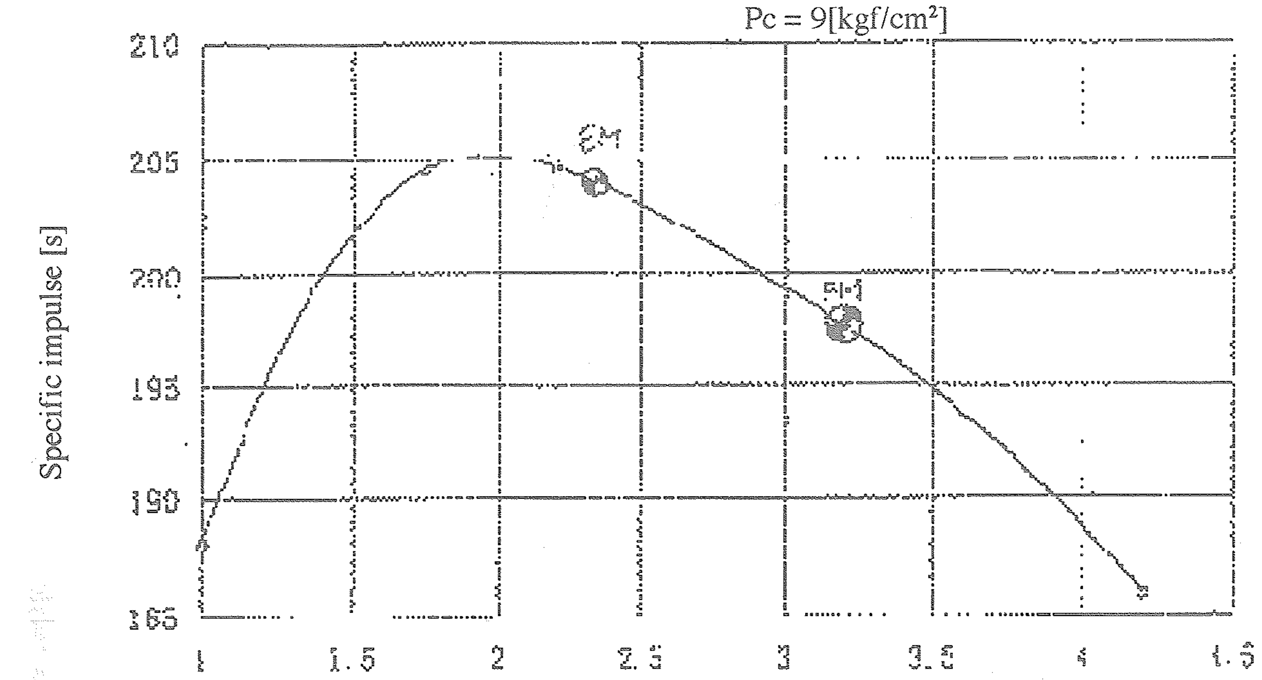


Fig.5.1-5 Nozzle expansion ratio ε

Calculation of the optimum expansion ratio for J-1 EVE combustor

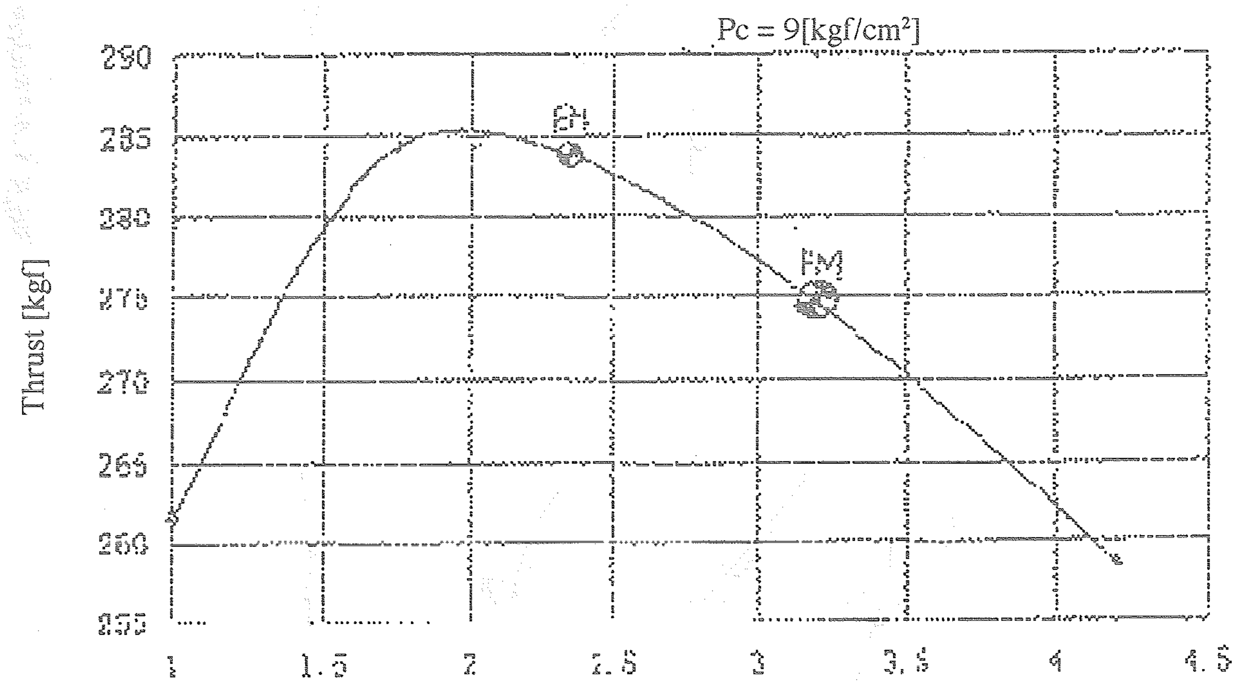


Fig.5.1-6 Nozzle expansion ratio ε

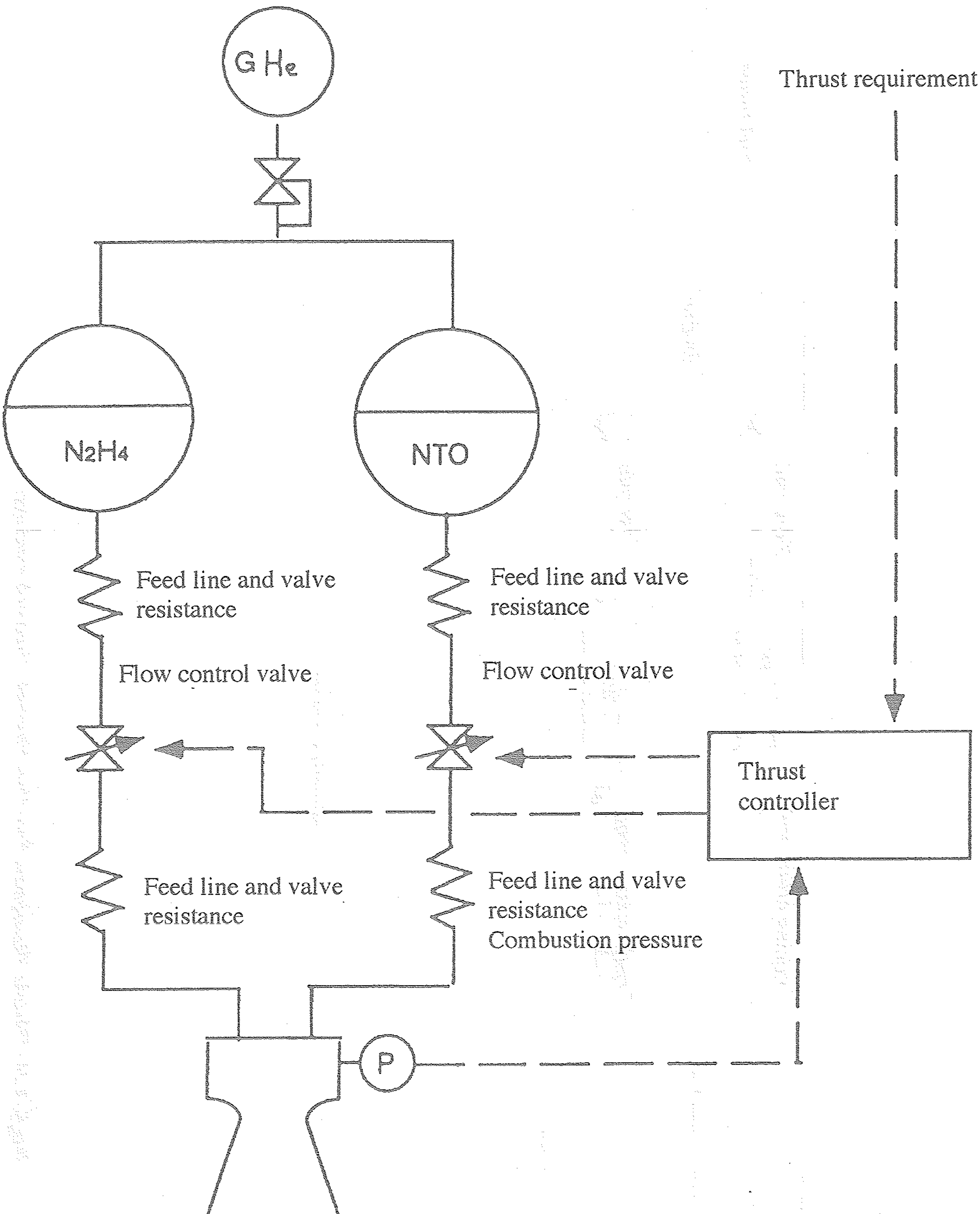


Fig.5.1-7 Functional configuration of the thrust control system

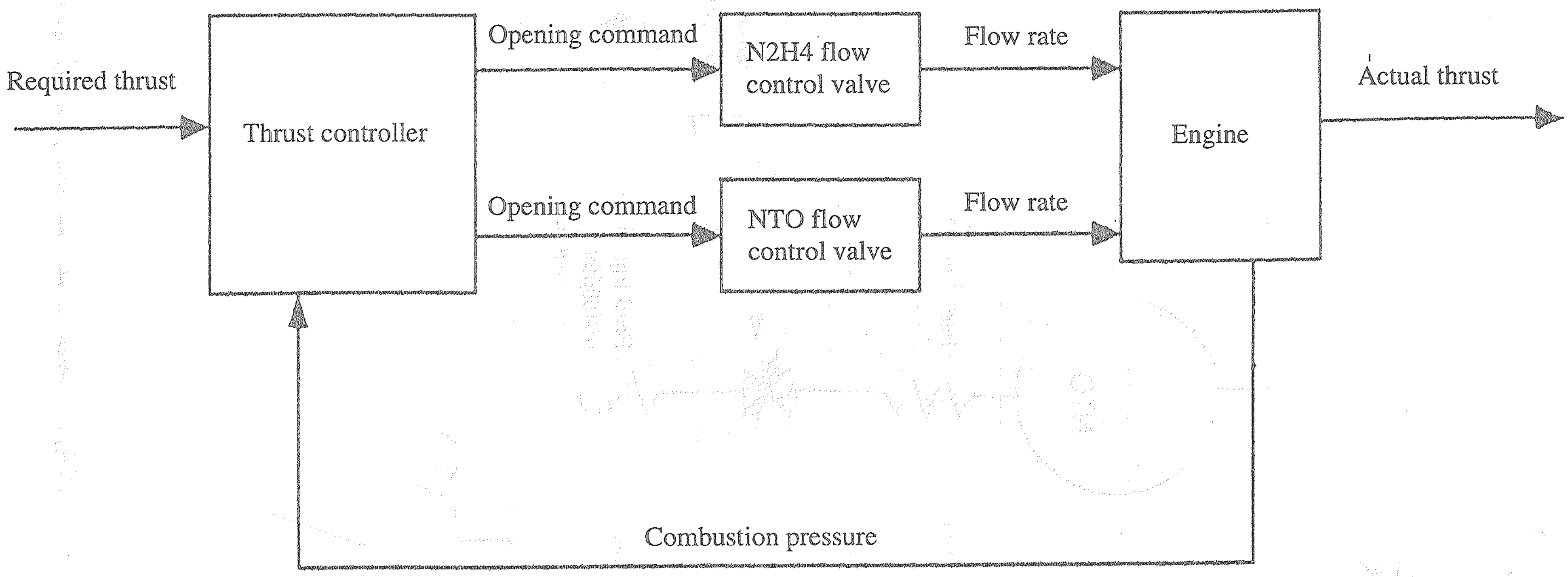


Fig.5.1-8 Block diagram for the thrust control system

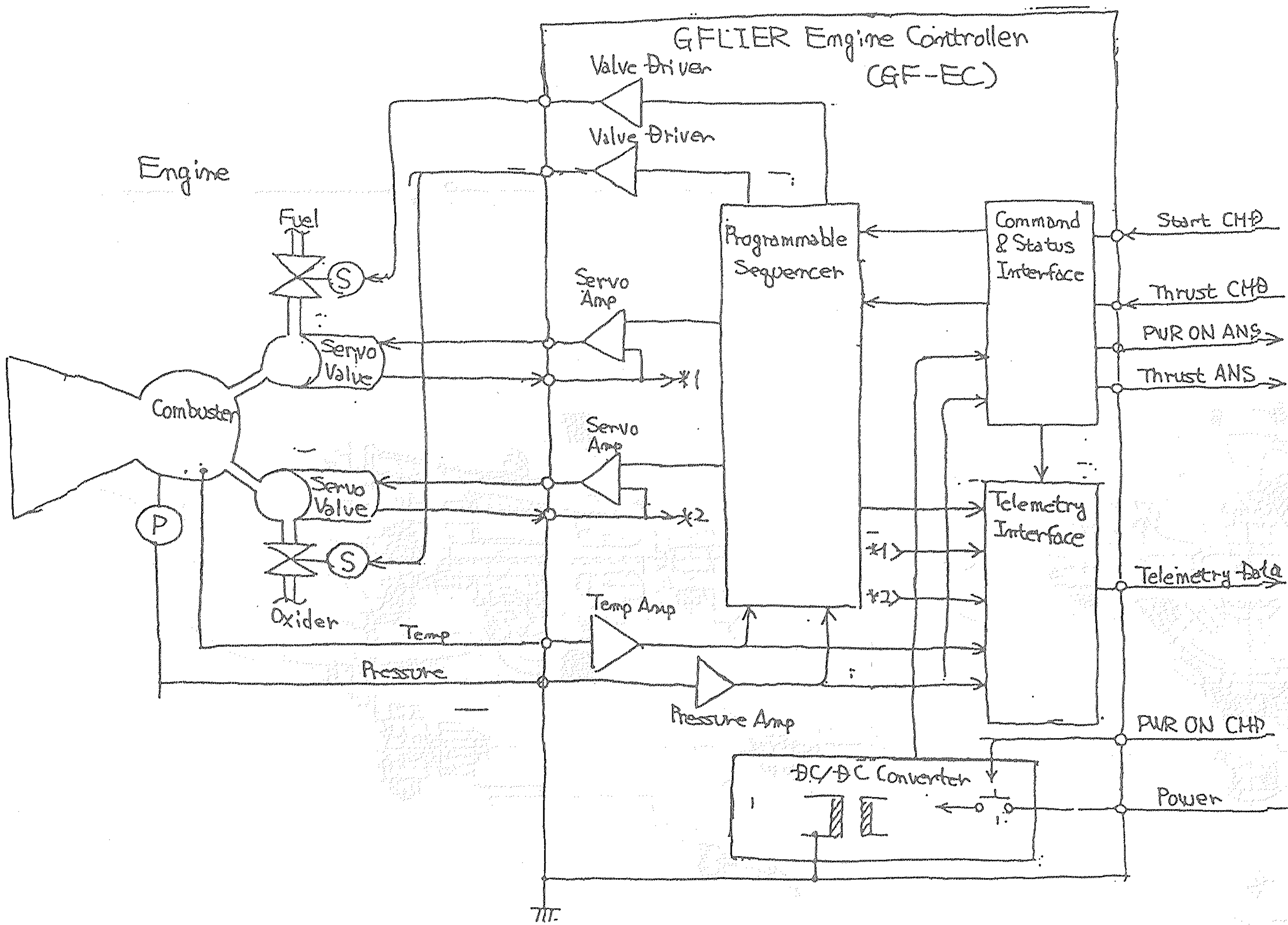


Fig.5.1-9 Block diagram for the thrust controller

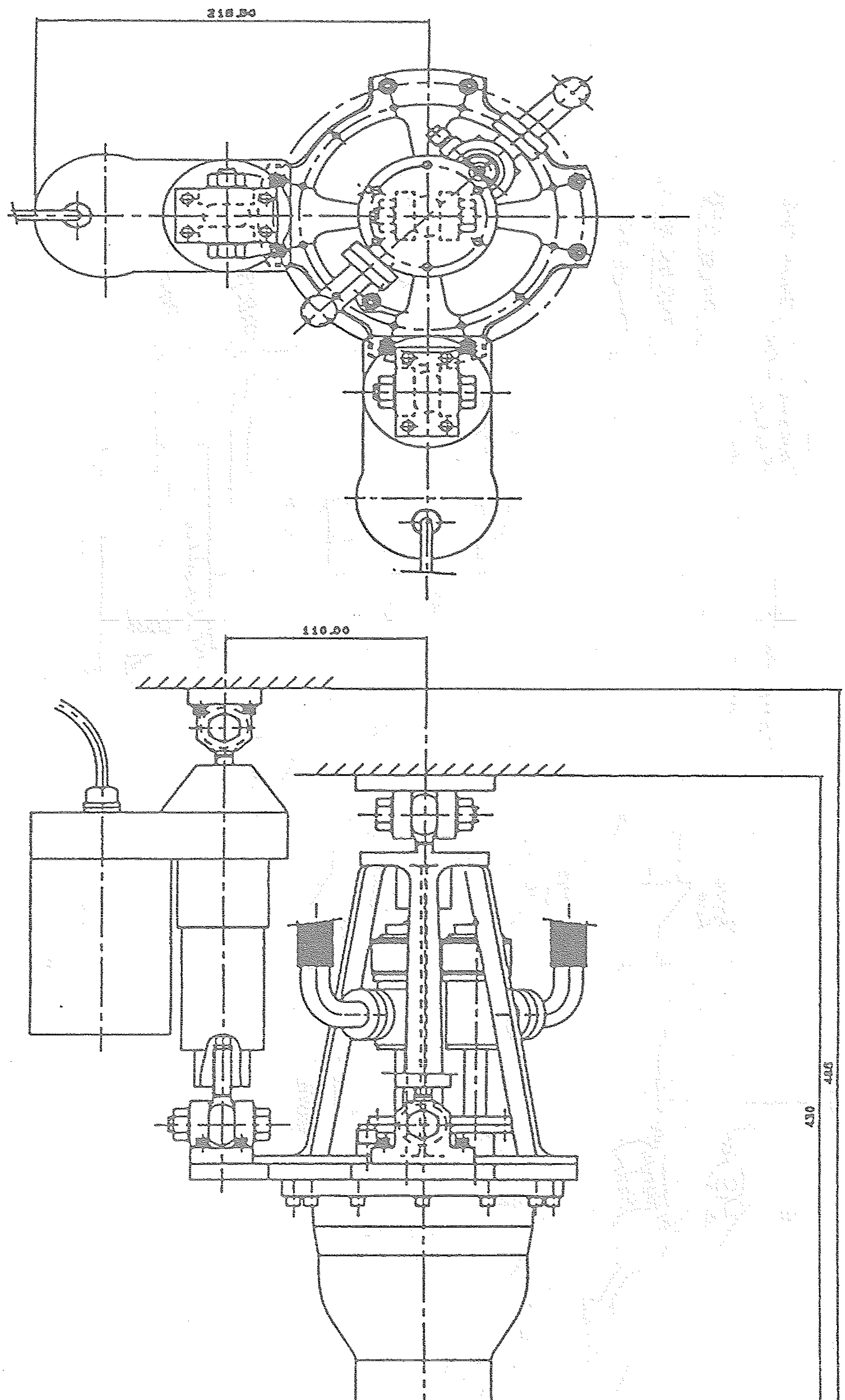


Fig.5.1-10 Major dimensions of the gimbal system

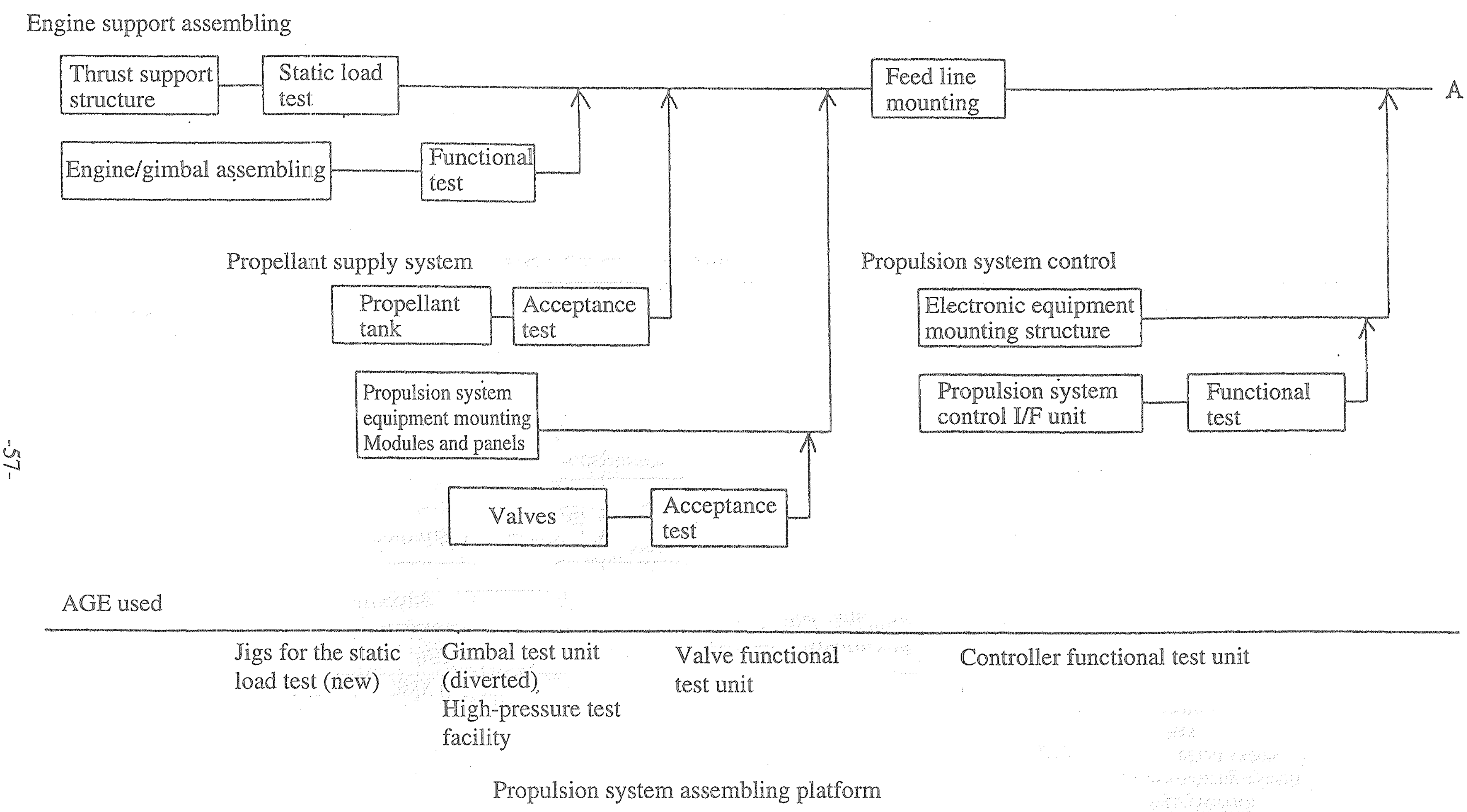


Fig.5.1-11 Flow in the assembling and testing of the propulsion system (1/4)

Fig.5.1-11 Flow in the assembling and testing of the propulsion system (2/4)

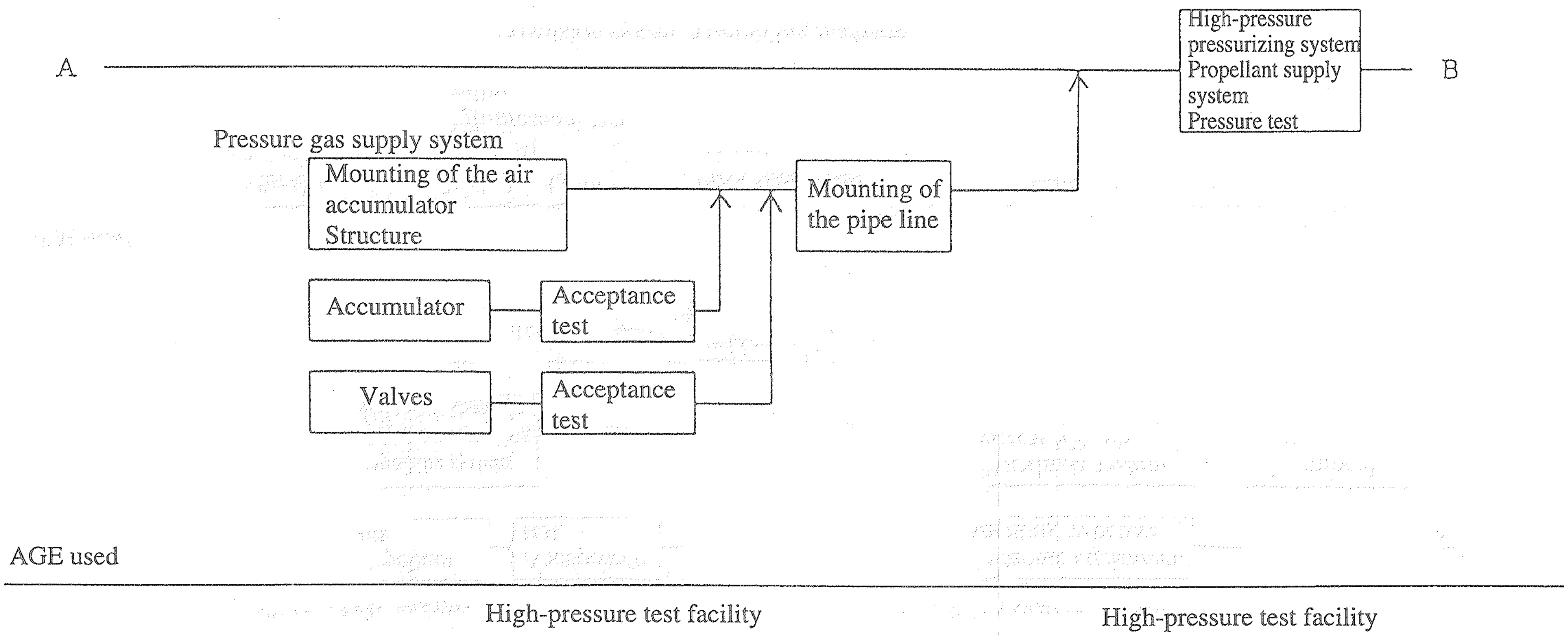


Fig.5.1-11 Flow in the assembling and testing of the propulsion system (3/4)

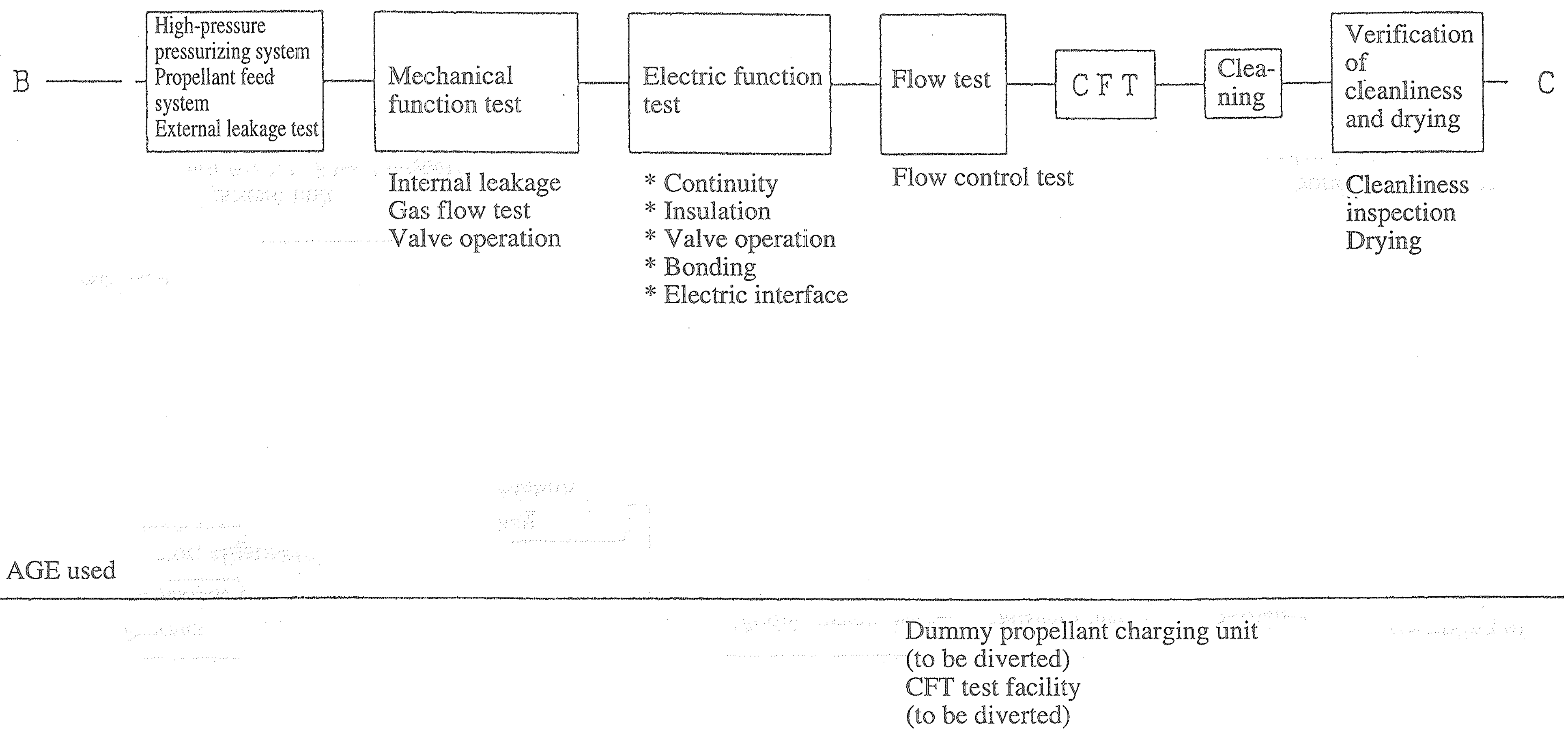


Fig.5.1-11 Flow in the assembling and testing of the propulsion system (4/4)

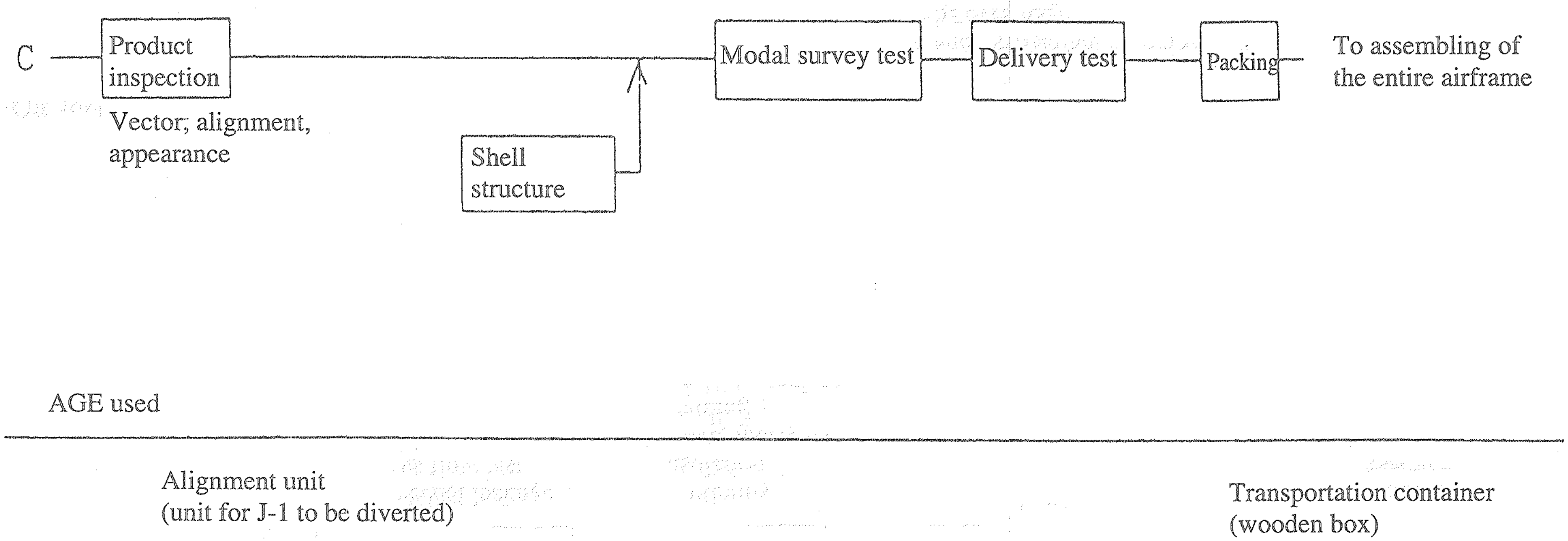


Table 5.1-5 Propulsion system development schedule

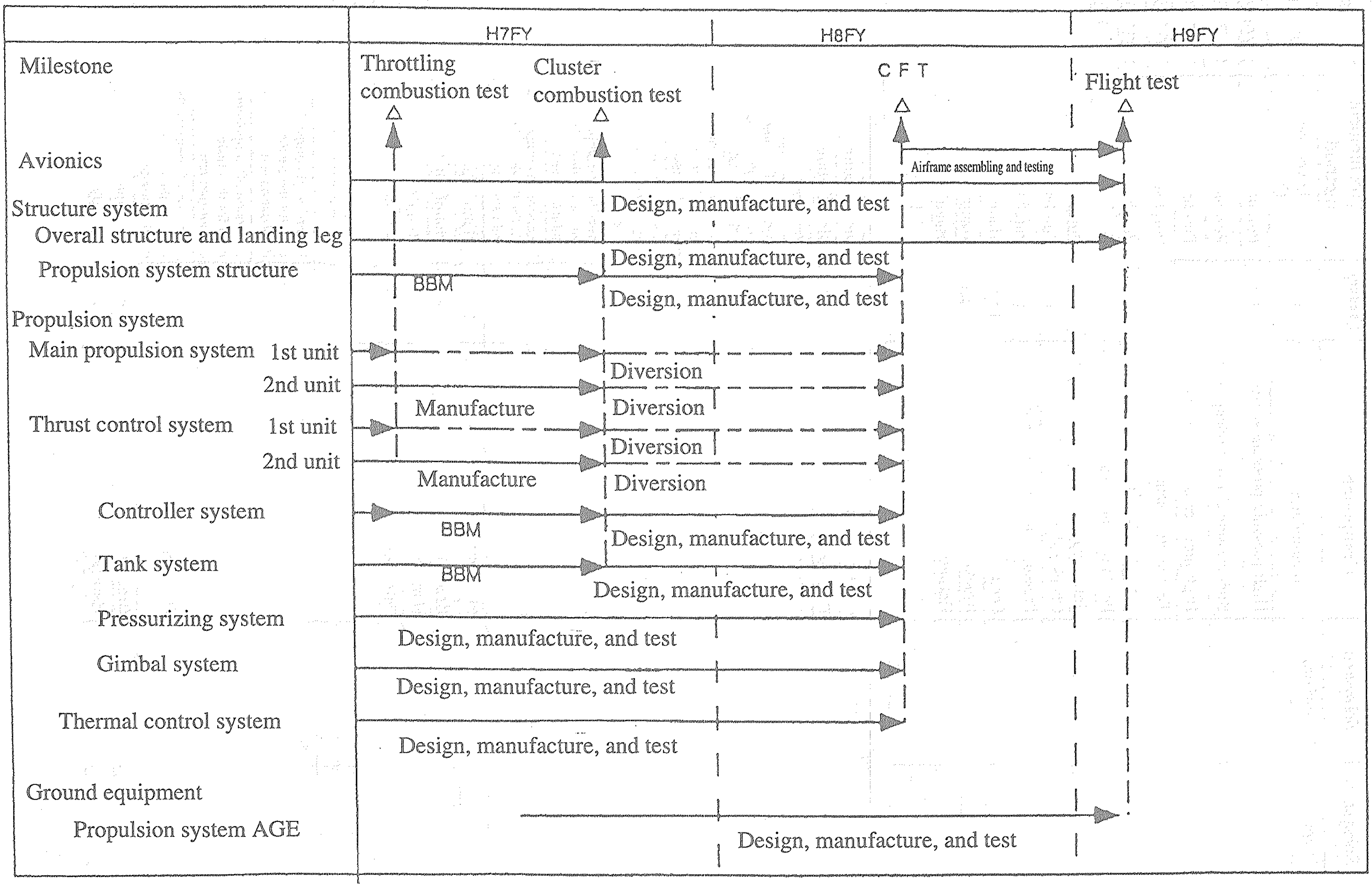


Table 5.1-6(1/2) Component procurement method

Subsystem	Subsystem	Component	Quantity	Procurement method	Past history
Propulsion system	Pressurizing system	Accumulator	1	Diversion (SFT2)	COMETS
		High-pressure pressure sensor	1	Purchase	HYFLEX
		High-pressure charge/discharge valve	1	Purchase	HYFLEX
		High-pressure isolation valve	1	Purchase	HYFLEX
		Temperature sensor	1	Purchase	HYFLEX
		High-pressure filter	1	Purchase	HYFLEX
		Pressure control valve	1	Purchase	HYFLEX
		Safety valve	1	Purchase	
		Check valve	1	Purchase	
		Pressurizing isolation valve	1	Purchase	
		Feed line	1 set	Manufacture	
		Bracket	1 set	Manufacture	
		Harness	1 set	Manufacture	
		Fasteners		Purchase	
	Tank system	Oxidizer tank	1	Manufacture	
		Fuel tank	1	Manufacture	
		Temperature sensor	2	Purchase	
		Low pressure sensor	2	Diversion of components used for the throttling combustion test	
		Exhaust valve	2	Diversion of components used for the throttling combustion test	
		Load/unload valve	2	Diversion of components used for the throttling combustion test	
		Manual isolation valve	4	Diversion of components used for the throttling combustion test	
		Flowmeter	2	Diversion of components used for the throttling combustion test	
		Main propellant feed line	1 set	Manufacture	
		Bracket	1 set	Manufacture	
		Harness	1 set	Manufacture	
		Fasteners	1 set	Purchase	
	Thrust control system	Regulating valve	4	Diversion of components used for the cluster combustion test	
		Filter	4	Diversion of components used for the cluster combustion test	
		Discharge valve	4	Diversion of components used for the cluster combustion test	
		Bracket	1 set	Manufacture	
		Harness	1 set	Manufacture	
		Fasteners	1 set	Purchase	

a. Throttling combustion test to be conducted in the fiscal 1994
b. Cluster combustion test to be conducted in the fiscal 1995

Table 5.1-6(2/2) Component procurement method

	Engine system	Engine	2	Diversion of components used for the cluster combustion test	J-1EVE
		Propellant	4	Diversion of components used for the cluster combustion test	COMETS
		Propellant pilot valve	4	Diversion of components used for the cluster combustion test	
		Pressure sensor	4	Diversion of components used for the cluster combustion test	
		Engine feed line	4	Manufacture	
		Bracket	1 set	Manufacture	
		Harness	1 set	Manufacture	
		Fasteners	1 set	Purchase	
	Gimbal system	Actuator	4	Purchase	
		Gimbal support	2	Manufacture	
	Controller system	Thrust controller	2	Manufacture	
		Regulator valve driver	7	Modification of consumer products	
		Gimbal driver	4	Modification of consumer products	
		Valve driver	1	Manufacture	

5.2. Structure System and Thermal Control System

5.2.1. Main Structure

(1) Development policy

* Utilization of existing technologies

Development of a reliable structure based on the design, manufacture, and operation technologies for various structures including aircraft, rockets, and satellites.

* Exclusion of marginal design

Without placing too much emphasis on weight reduction, the design provides just enough margin of strength to bypass the design verification through the development tests.

* Modularization

In order to ensure that various subsystems are developed in parallel, the development work places emphasis on modularization and interfaces.

* Maintainability

Development of a structure that is excellent in maintainability by combining the modularization technology for aircraft and the assembling and mounting technology for space propulsion units

(2) Required specifications

* The main structure supports all onboard equipment and withstands the environmental load arising from all operational sequences.
* Maintenance and inspection of propellant and onboard equipment are easy.
* In order to prevent the transportation load conditions from becoming excessive, proper interfacing with AGE is provided.
* In order to be transported by a crane and the like, the structure is provided with hoisting fixtures and support fixture for transportation with the vehicle laid on its side.

(3) Design Specifications

Buckling strength is the major criterion for the main structure and the repeated load does not pose specific problems with respect to the strength. Here, 2 options are proposed for the basic structure. There is no difference in weight and cost between the tow options; both pose no problem in feasibility and embody useful ideas in conducting reviews on the structure in the future; therefore, the selection is left to the final phase review .

Components and materials commonly used for aeronautics and space applications will be adopted therefore, no problem of commercial availability is foreseen.

(A) Upper structure

Consists mainly of the portions supporting the equipment and tanks; adopts the truss construction. The tanks are positioned at the upper and lower poles.

Truss: CFRP
Panel: Al honeycomb
Coupling: AL alloy

An equipment mounting structure is placed between the propellant tank and the pressurizing tank.

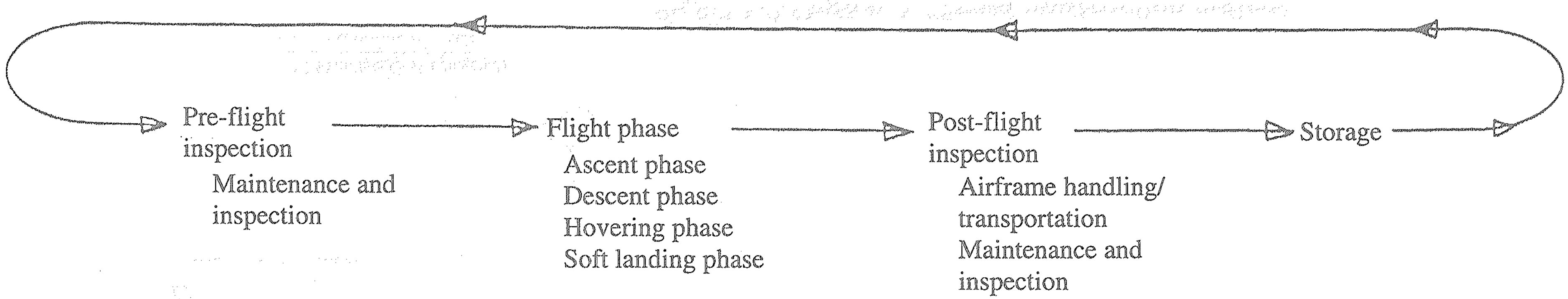
(B) Lower structure

Consists mainly of the portions to which the legs and engines are attached; adopts the box beam construction.

Truss: CFRP
Panel: Al honeycomb
Coupling: AL alloy

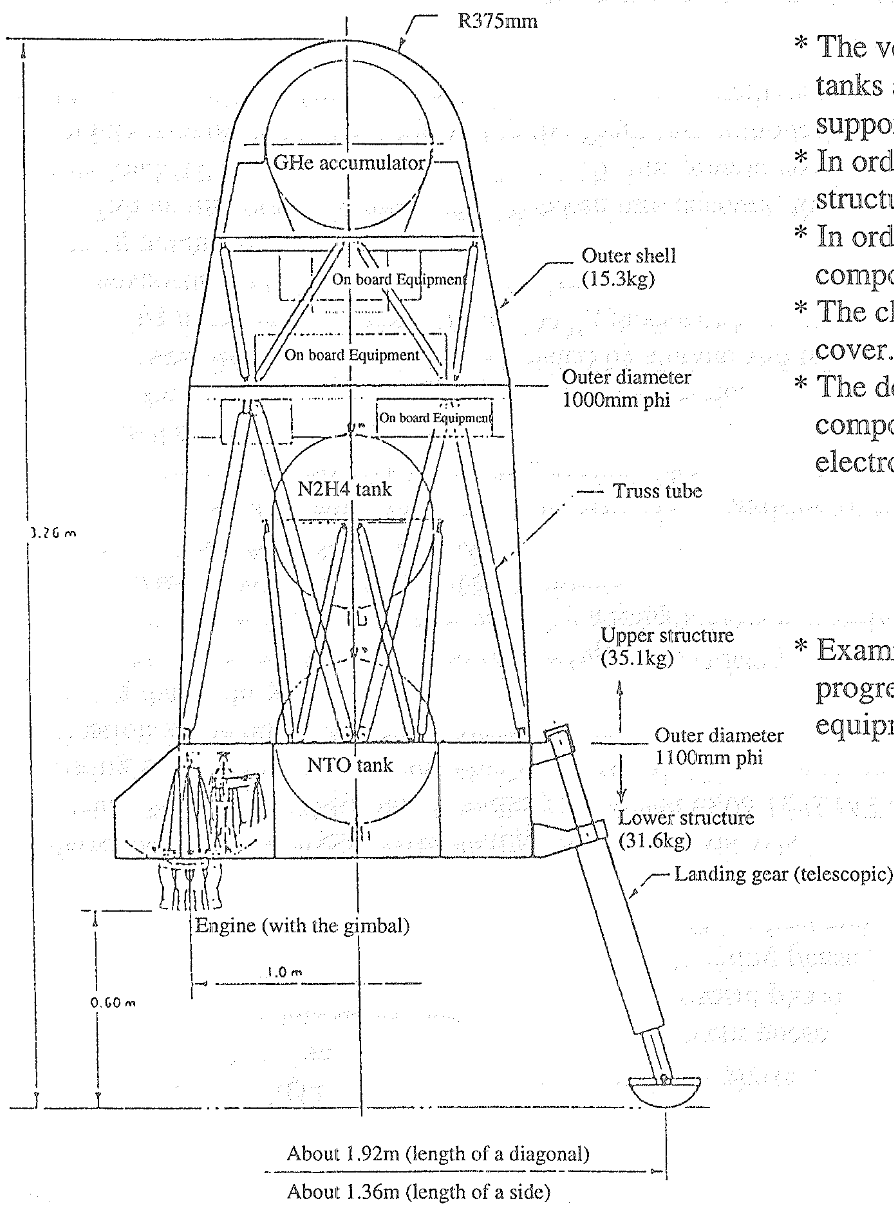
(4) Test Plan

The development test is not conducted on the main structure system.



* Acceleration in the direction perpendicular to the airframe axis
  - During flight and at the time of landing: Assumed to be 1g (LMT) on the safe side.
  - During transportation with the vehicle laid on its side on the truck: 3g (LMT)(AGE for transportation to be taken into consideration)
* Acceleration in the direction of the airframe axis
  - During the flight phase:
    - Flight phase except at the time of landing: 0-2g (LMT)
      - Determine the acceleration value by adding a certain margin to the value (maximum thrust-to-weight ratio)+(acceleration caused by 10 m/sec cross-wind at an inclined attitude).
    - At the time of landing: 3g (LMT)
      - To be set at a value within the load resistance condition of the onboard equipment with the balance among the weight, action of the landing gear, and ease of landing control taken into consideration; at the moment, however, the acceleration is assumed to be the same as that for aircraft, i.e. 3g.
  - During the handling and transportation of the airframe:
    - It is assumed that the airframe is hoisted by a crane and transported laid on its side on a truck.
    - During hoisting by a crane: 0-2g (LMT); during transportation with the airframe laid on its side on a truck: 3g (LMT)(AGE for transportation to be taken into consideration)
  - During maintenance and inspection:
    - With an uncertain variation of 20% taken into account, the acceleration is assumed to be 0.8-1.2g (LMT).
* Ultimate load (ULT) = Limit load (LMT) x 1.25 (the general conditions for the rocket structure apply.)
* The vehicle is expected to be subjected to the flight test without the preliminary structural strength test; with a view to avoiding increase in the weight, the objective safety margin for the structural strength is set at about 0.5.

Fig.5.2.1-1 Load conditions and structural strength



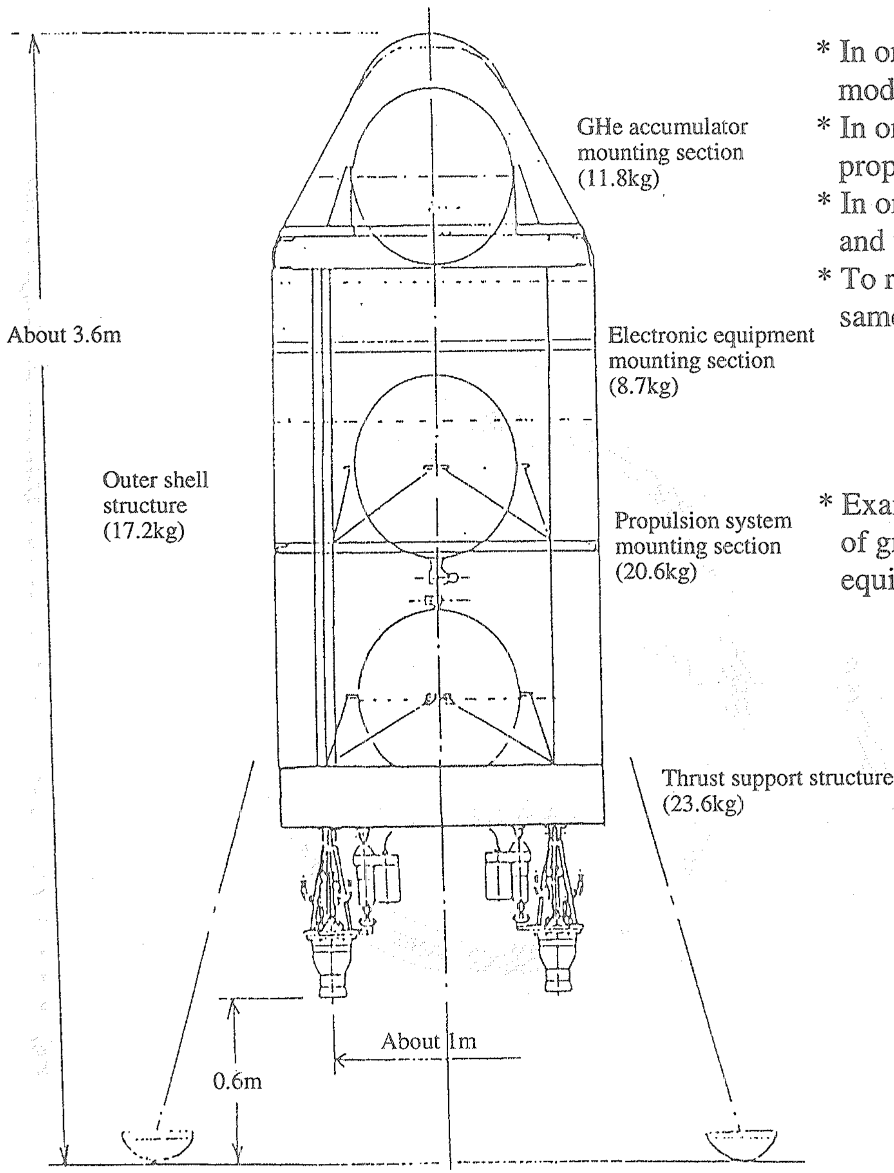
Concept behind and features of this construction

* The vehicle consists of the truss construction upper structure that supports the tanks and various equipment and the box beam construction lower structure that supports the engines and landing gear.
* In order to ensure maintainability, the outer shell of the upper structure is non-structural.
* In order to ensure good formability, the cap is composed of a FRP cover. Other components are made of common aluminum materials.
* The clearance between the engine and the bottom skin is filled with a spherical cover.
* The design aims at a structure consistent with the modularization; for example, components are arranged so that the propulsion system equipment and the electronic equipment do not interfere with each other.

Problems to be addressed in the future

* Examination and review of the details of the structure in harmony with the progress of the design of various onboard equipment and propulsion system equipment.

Fig.5.2.1-2 Option A (Truss construction option)



Concept behind and features of this construction

* In order to shorten the development stage and reduce cost, the structure is modularized (to enable the subsystems to be developed simultaneously.)
* In order to improve the facility of the process of assembling and equipping of the propulsion system, the structure is modularized.
* In order to reduce the development risk, important components such as the tanks and flow control valves are separated from the engine mounting plane.
* To reduce the weight, an outer shell is adopted as the structural members; at the same time, the fasteners are removed to ensure maintainability.

Problems to be addressed in the future

* Examination of the details of the system construction with the weight and centers of gravity of the landing legs, propulsion system equipment, and electronic equipment taken into consideration.

Fig.5.2.1-3 Option B (Outer shell construction option)

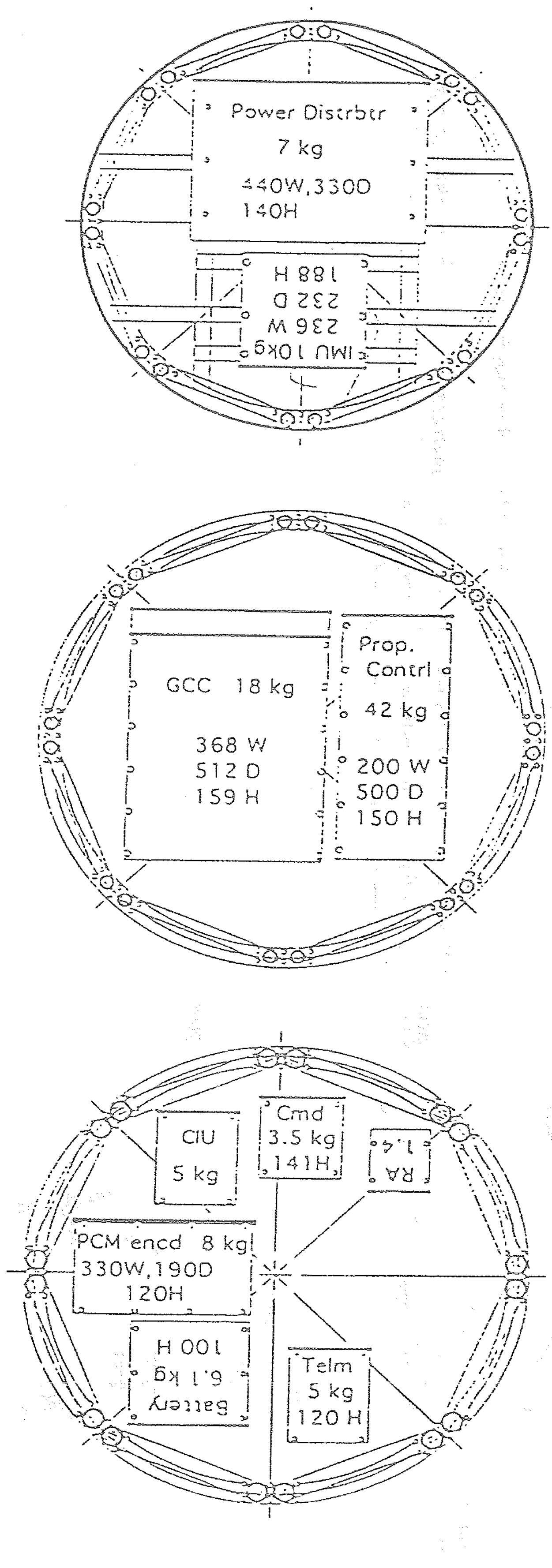


Fig.5.2.1-4 Example of onboard equipment arrangements

Table 5.2.1-1 (1/3) Breakdown of weight by component

G - FLIER Structure system Weight estimation 4th (tank arrangement according to the B1 option, gimbal) REV.A H7.2.20 (In kg)

*** AL structure/telescopic leg Lower structure=1100mm Equipment mounting section=1000 Cap radius=375mmR

	Component	Quantity	Weight	Total weight	Remarks
	Structure system weight		Total =	132.000	Upper structure + Lower structure + Outer shell / Painting + Landing leg
	Main structure			66.745	Upper structure + lower structure
	Upper truss structure			35.108	
	Upper ring frame	1	2.305		AL ring machined, thickness 2t
	GN2 tank support fixture	2	0.249		AL angle machined and extension fitting
			0.249		
	Rod connecting fixture, upper A	2	0.140		AL fitting
			0.140		
	Rod connecting fixture, upper B	2	0.140		AL fitting
			0.140		
	Rod assembly, upper	8	0.468		AL tube 1 inφ×.040int×500L with two AL end-fittings
			0.468		
			0.468		
			0.468		
X	Equipment mounting section ring frame	1	2.922	13.747	AL ring machined, thickness 2t Including connecting fixtures for transportation
	Channel A	2	0.302		AL formed channel (25w*40h*1.6t) L=750
			0.302		
	Channel B	2	0.323		AL formed channel (25w*40h*1.6t) L=800
			0.323		
	Channel C	2	0.161		AL formed channel (25w*40h*1.6t) L=400
			0.161		
	Clip	16	0.252		AL machined T section
	Gusset plate A	4	0.086		AL sheet (.063in t)
	Gusset plate B	8	0.271		AL sheet (.063in t)
	Rib	14	0.204		AL formed channel 1.6t
	Equipment mounting panel Skin/doubler		1.828	7.680	.012 skin & .012 doubler
	Film adhesive		0.647		.06 psf
	Core		1.561		3.1 pcf
	Foam adhesive		0.352		2.93 kg/m2/ply
	Potting		2.093		0.58 g/cm3
	Mount		1.200		20 g * 60 ea
			0.000		
			0.000		
			0.000		
			0.000		
	GHe accumulator support structure	2	0.379		AL formed pannel
			0.379		
			0.000		
			0.000		
			0.000		
			0.000		
			0.000		
			0.000		
	Rod connecting fixture, middle and upper	4	0.272		AL fitting
			0.272		
			0.272		
			0.272		
	Rod connecting fixture, middle and upper	4	0.272		AL fitting
			0.272		
			0.272		
			0.272		
	Rod assembly, lower	8	1.455		AL tube 1.5 inφ×.058int×1100L with two AL end-fittings
			1.455		
			1.455		
			1.455		
	Tank support rod assembly	8	0.564		AL tube 1 inφ×.040int×700L with two AL end-fittings
			0.564		
			0.564		
			0.564		
	Rod connecting fixture, lower	4	0.544		AL fitting
			0.544		
			0.544		AL fitting
			0.544		
	Tank connecting fixture	4	0.200		AL plate
			0.200		
			0.200		
			0.200		
	Panel mounting member A	4	0.939		AL machined T section bar with nutplates
			0.939		

Table 5.2.1-1 (2/3) Breakdown of weight by component

G-FLIER Structure system Weight estimation 4th (tank arrangement according to the B1 option, gimbal) REV.A H7.2.20 In kg

*** AL structure/telescopic leg Lower structure=1100mm Equipment mounting section =1000mm Cap radius=375mmR***

Component	Quantity	Weight	Total weight	Remarks
Hoisting fixture	4	0.127		AL fitting
		0.127		
		0.127		
		0.127		
M S P	1	0.500		bolt, nut, washer etc
Lower structure			31.637	
Equipment mounting panel	1	2.999		AL skin (.032t) & angle(20×20×.050t, (24+8)ea) build up
Upper frame A	1	1.703		AL ring machined, 50×30×2t (T)
Upper frame B	1	0.604		AL ring formed, 30×30×2t (angle)
Outer surface skin	1	3.028		AL skin (.032t)
Outer surface stiffener	1	1.769		Al formed angle (20×20×2t)
Web	1	2.714		AL formed web (.032t) with bead & hole
Inner surface skin	1	1.431		AL skin (.032t)
Inner surface stiffener	1	0.767		Al formed angle (20×20×2t)
Lower frame A	1	1.277		AL ring formed, 30×30×2t (angle)
Lower frame B	1	0.604		AL ring formed, 30×30×2t (angle)
Bottom panel A	1	2.999		AL skin (.032t) with stiffner
Bottom panel B	1	1.391		AL skin (.032t) with stiffner , removable panel
	0	0.000		
Truss support structure reinforcement	4	0.600		beam, doubler, etc
Leg mounting structure reinforcement	4	1.200		beam, doubler, etc
Leg mounting fixture A	4	1.600		AL fitting
Leg mounting fixture B	4	2.400		AL fitting
Main engine mounting structure/reinforcement	2	0.800		beam, doubler, etc
Thruster mounting structure/reinforcement	2	0.600		beam, doubler, etc
Bulge structure/reinforcement and sealing	2	0.800		beam, doubler, etc
Altimeter mounting structure/reinforcement	1	0.350		beam, doubler, etc
Inspection door installing structure reinforcement	1	1.000		beam, doubler, etc
M S P	1	1.000		bolt, nut, washer etc
Outer shell/painting			15.256	
Cap	1	3.415		FRP CAP
Panel A	2	2.959		.020t AL skin and .020t Al former, build up with spot welding
Panel B	2	8.137		.020t AL skin and .020t Al former, build up with spot welding
Weight of the thermal insulators included in the weight of the heat control system	1	0.000		insul coat 3t (Density 0.35) & top coat 0.08t (Density 1.5)
Weight of painting on the skin included in the weight of the heat control system	1	0.000		0.17 kg / m2
M S P	1	0.744		bolt, nut, washer etc
Landing leg	4	12.500	50.000	For the detailed listing, refer to the reference material regarding the examination of the legs.
		12.500		
		12.500		
		12.500		
Onboard equipment			74.000	
		10.000		
GCC Radio altimeter		18.000		
RA Antenna		1.400		
RA		0.060		
CIU		5.000		
Telemetering transmitter		5.000		
PCM encoder/signal converter		8.000		
Command receiver/demodulator		3.500		
Telemetering transmitter antenna		0.100		
Command receiver antenna		0.050		
Battery	1	6.100		
Switchboard		7.000		
Shock mount		2.000		
Wire harnesses, connectors, and clamps		7.790		
Propulsion system			142.000	
Pressure feed system		27.000		
Propellant feed system		20.000		
		20.000		
Engine system		12.000		
		12.000		
Gimbal system		12.500		
		12.500		
Thrust control system		6.500		

Table 5.2.1-1 (3/3) Breakdown of weight by component

G-FLIER Structure system Weight estimation 4th (tank arrangement according to the B1 option, gimbal) REV.A H7.2.20 In kg

***AL structure/telescopic leg Lower structure=1100mm Equipment mounting section=1000mm Cap radius=375mmR

Component	Quantity	Weight	Total weight	Remarks
		6.500		
		6.500		
		6.500		
Control system		42.000		
Heat control system		1.750	7.000	
		1.750		
		1.750		
		1.750		
Margin		5.000	20.000	
		5.000		
		5.000		
		5.000		
Propellant		60.000	124.000	
		60.000		
Pressurizing agent		4.000		
		[DRY]	[WET]	
Weight of the experimental vehicle		417.0	541.0	
Position of the centers of gravity				
CGx		7	6	
CGy		-2	-2	
·CGz		1597	1577	Expressed in terms of the height from the ground
Moment of inertia				
Ixx,cg		3.49E+08	3.82E+08	Estimated on the assumption that Ixx=Ixx calc*1.2.
n		3.50E+08	3.83E+08	Estimated on the assumption that Iyy=Iyy calc*1.2.
Izz,cg		9.52E+07	9.99E+07	Estimated on the assumption that Izz=Izz

5.2.2 Landing Gear

(1) Associated documents

(a) GAF-94012: "NASDA Draft for the Conception of Vertical Takeoff and Landing Experiment" November 4, 1994

(b) ——— "NASDA Draft for the Conception (Reference Model)"

(2) Design policies

With the associated document (a) in Item (1) used as a reference, the following design policies have been established and applied:

(a) To utilize existing technologies while minimizing use of novel technologies;

(b) To maximize the utilization of ready-made or consumer products as functional components to reduce cost;

(c) The marginal design for weight reduction is not used; however, the weight reduction design is introduced only when required from the viewpoint of realizing the airframe; and

(d) The redundant design is not introduced.

(3) Design Conditions

Table 5.2.2-1 shows the basic design conditions (design requirements posed by the systems) for the leg set of the vertical takeoff and landing experiment vehicle.

Table 5.2.2-1 Design conditions for the Langing gear assy.

Item			Design requirements posed by the systems	Design conditions adopted in the current examination
Required functions			<1> To alleviate landing shock <2> To support the airframe <3> To prevent the vehicle from being overturned <4> The retractable mechanism is not necessary. <5> To secure the ground clearance for the engine nozzle	* The same as the left; however, the MAXIMUM structure weight is limited to 420kgf.
Maximum takeoff and landing weight			550kgf (structure weight =390kgf MAX.)	
Height of the airframe center of gravity from the airframe bottom			TBD	0.82 m MAX.
Landing conditions	Attitude	At the time of takeoff	Vertical — The vehicle is capable of takeoff and landing using the landing gear.	The same as the left, provided that the inclination of the airframe at the time of landing is 1 degree at MAX.
		At the time of landing	Vertical (the striking attitude is T.B.D.) — The vehicle is capable of takeoff and landing using the landing gear.	
	Velocity	Vertical	TBD	3 m/s MAX
		Horizontal	TBD	* Normal horizontal landing on a concrete pavement is considered. * Emergency horizontal landing on sandy terrain and the like is not considered. *2
	Limit load factor		TBD	
	Engine thrust		* The engine is assumed to be in operation at the time of landing.	* The engine thrust is assumed to balance with the weight empty until the airframe stops when striking the ground for landing.
	Landing surface		* Normally, a concrete pavement * The vehicle is capable of landing on unpaved flat land in case of emergency (landing on sandy terrain to be considered)	* The striking portion is capable of holding the airframe attitude even when it is landing on sandy terrain.
Ground clearance		At the time of takeoff	TBD	* About 0.4m *3 Height of the engine jet nozzle from the ground
		At the time of landing	TBD	
Heat resistance			* Heat resistance to the flame jetted from the engine to be considered	* Heat to be controlled so that the ambient temperature may not rise above the heat withstanding temperature of the leg set.
Engine flame temperature			TBD	
Aerodynamic heating			* Consideration not necessary	The same as the left
Life			Not less than 10 times (to be capable of being used repeatedly)	
Flight time			* About 60 sec.	
Operational altitude			* Not more than 300m	
Launching time			* In the fiscal 1997, objective	
Manned/unmanned			* Unmanned	

Notes
* 1 Excerpts from the associated document (a) in Item (1)
* 2 Cross-wind at the time of striking (50m/s) is taken into consideration.
* 3 Clearance in the freestanding condition under 1G; the minimum clearance against sinking at the time of landing is not less than 0.1m.

(4) Examination

(a) Leg spacing

(A) Required minimum leg set spacing

A leg set is provided with a minimum spacing required to meet the requirements stated in Conditions <1> through <4> in Item (i) and Item (ii); however, the values shown below are used for each corresponding condition:

(i) Prevention of overturning of the airframe

	Values used for examination
<1> Whether or not the side slip speed exists at the time of striking	Normal landing: To be taken into consideration Emergency landing: Not taken into consideration Cross-wind not taken into consideration (Refer to Table 5.2.2-1.)
<2> Magnitude of the coefficient of friction at the time of striking	0.4 *1 MAX
<3> Inclination of the airframe at the time of striking	1 degree MAX
<4> Height of the airframe center of gravity from the airframe bottom	0.82mMAX

(ground clearance for the engine nozzle: not less than about 0.4m)

(ii) Provision of the shock absorbing function of the shock strut
(This must be taken into consideration when swing – arm type leg set is used.)

As Table 5.2.2-2 indicates, a leg set spacing *2 of not less than 0.61x2=1.22m is necessary.

---

*1 Maximum coefficient of friction of a concrete pavement
*2 Length of a side of a regular square

Table 5.2.2-2 Required leg set spacing

μ *4	θ (°) *4	B (m) *1.2 *4	B ' (m) *1.2 *4	Drawings to be referred to
0.4	67.2	0.61	0.86	Fig.5.2.2-1
0.6	58.0	0.91	1.29	
0.8	50.3	1.20	1.70	Fig.5.2.2-2

Notes:

*1. The experimental vehicle is equipped with a leg set composed of 4 legs.

*2. The values listed above are for the case in which the height of the center of gravity from the ground is 1.45m.

*3. The angle of inclination of the airframe at the time of landing is 1 deg. at maximum.

*4. μ: Coefficient of friction of the striking portion
θ: Airframe overturning angle for the two-point striking
2B: Leg set spacing when each of the four legs is located at each vertex of a corresponding regular square. (length of a side)
2B': Leg set spacing when each of the four legs is located at each vertex of a corresponding regular square. (length of a diagonal)

((B)) Coefficient of friction and required leg set spacing

The greater the coefficient of friction of the striking portion becomes, the greater leg set spacing is required.

As is shown in Fig.5.2.2-2, increase in the leg set spacing required affects the design of the telescopic type leg set in particular.

Regarding the coefficient of friction, therefore, the minimum values meeting the leg set design conditions shown in Table 5.2.2-1 are adopted.

μ is equal to or greater than 0.4.*

* Maximum coefficient of sliding friction of the striking portion against the concrete pavement

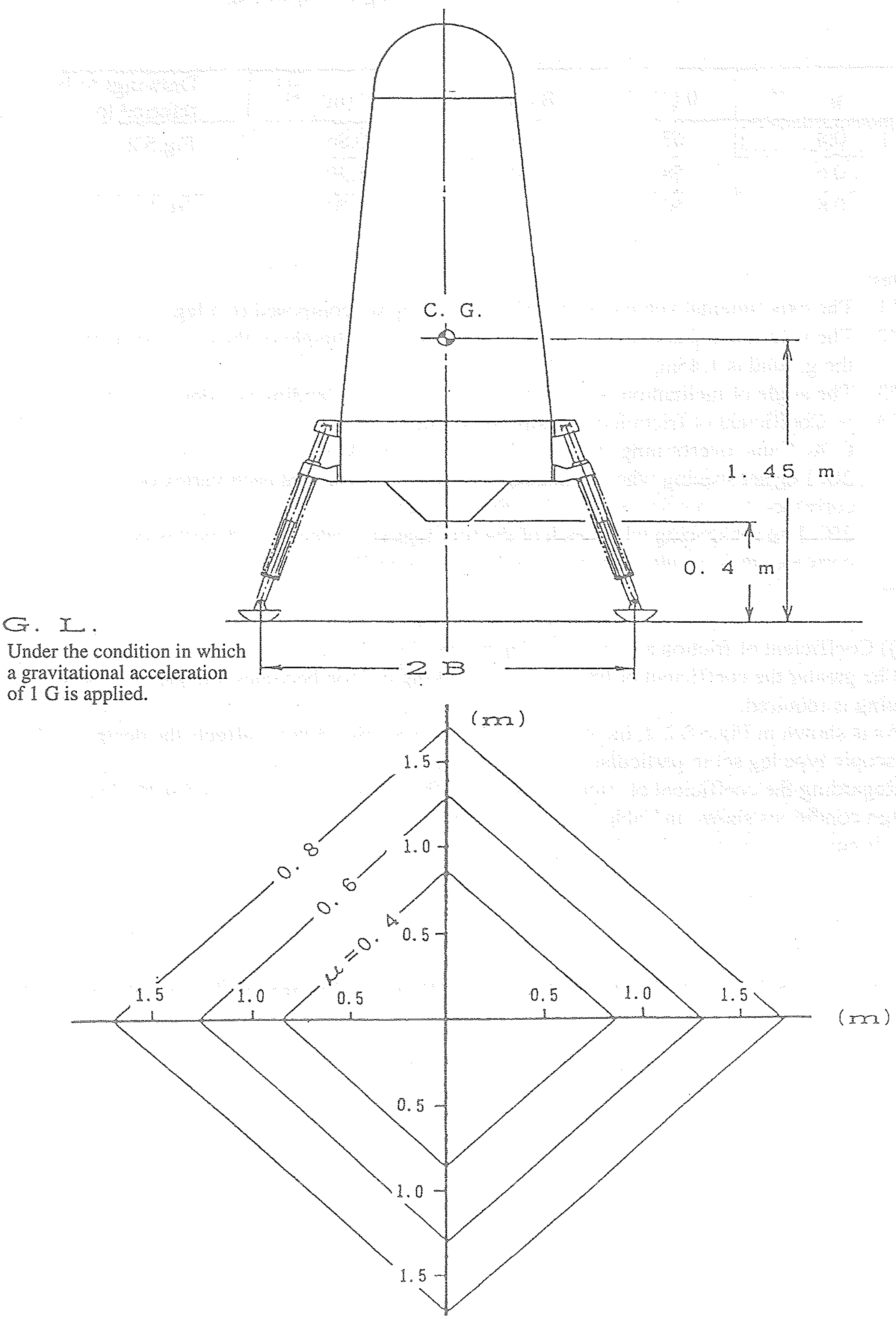


Fig.5.2.2-1 Required leg set spacing

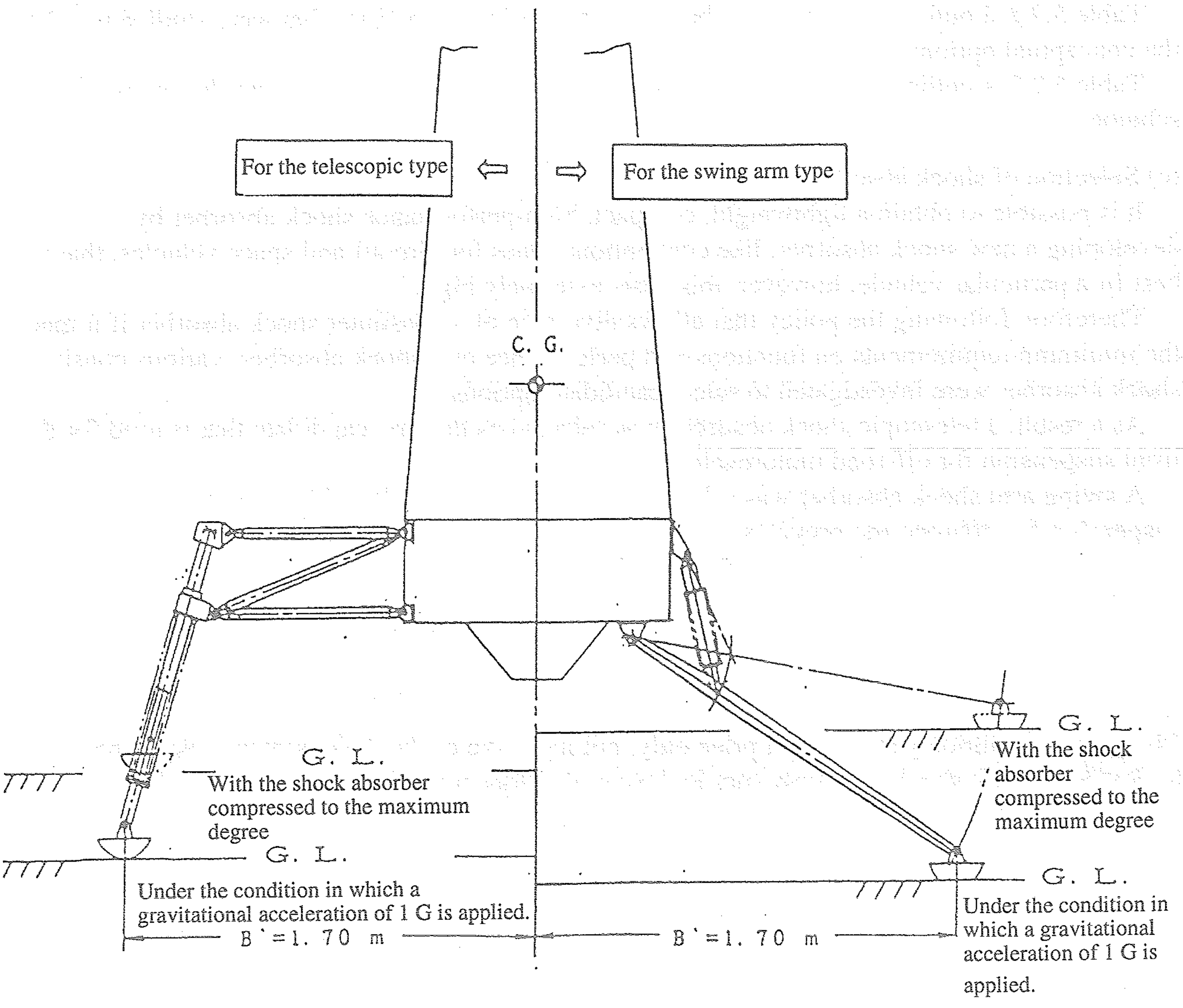


Fig.5.2.2-2 Required leg set spacing for μ=0.8

(b) Selection of leg set options

In order to draft conceptual options for the leg set to be adopted for the experimental vehicle, examination of and comparison between various candidate options were conducted.

As a result, the telescopic scheme was selected as the first candidate leg set option. (refer to Fig.5.2.2-3.)

In addition, the swing arm scheme was selected as the second candidate leg set option. (refer to Fig.5.2.2-4.)

Table 5.2.2-3 outlines the trade-off between various leg set options that were studied to select the conceptual options.

Table 5.2.2-4 outlines the results of comparison between the telescopic and the swing arm scheme.

(c) Selection of shock absorbers (oleo)

It is possible to obtain a lightweight, compact, high-performance shock absorber by developing a new shock absorber, like conventional ones for aircraft and space vehicles, that will best fit a particular vehicle; however, this costs extremely high. [*1]

Therefore, following the policy that allows diversion of a consumer shock absorber if it meets the minimum requirements on functions and performance of a shock absorber, various consumer shock absorber were investigated to select candidate options.

As a result, a telescopic shock absorber was selected as the first candidate that is used for the front suspension for off-road motorcycles.

A swing arm shock absorber was selected as the second candidate that is used also for the rear suspension for off-road motorcycles.

Table 5.2.2-5 outlines the results of the trade-off.

---

*1. About 10 million yen (product price only, not inclusive of the development cost) is required to develop a nose shock strut only (not inclusive of a drag strut, a wheel, a tire, and all other components) for a small fixed wing airplane (such as a trainer).

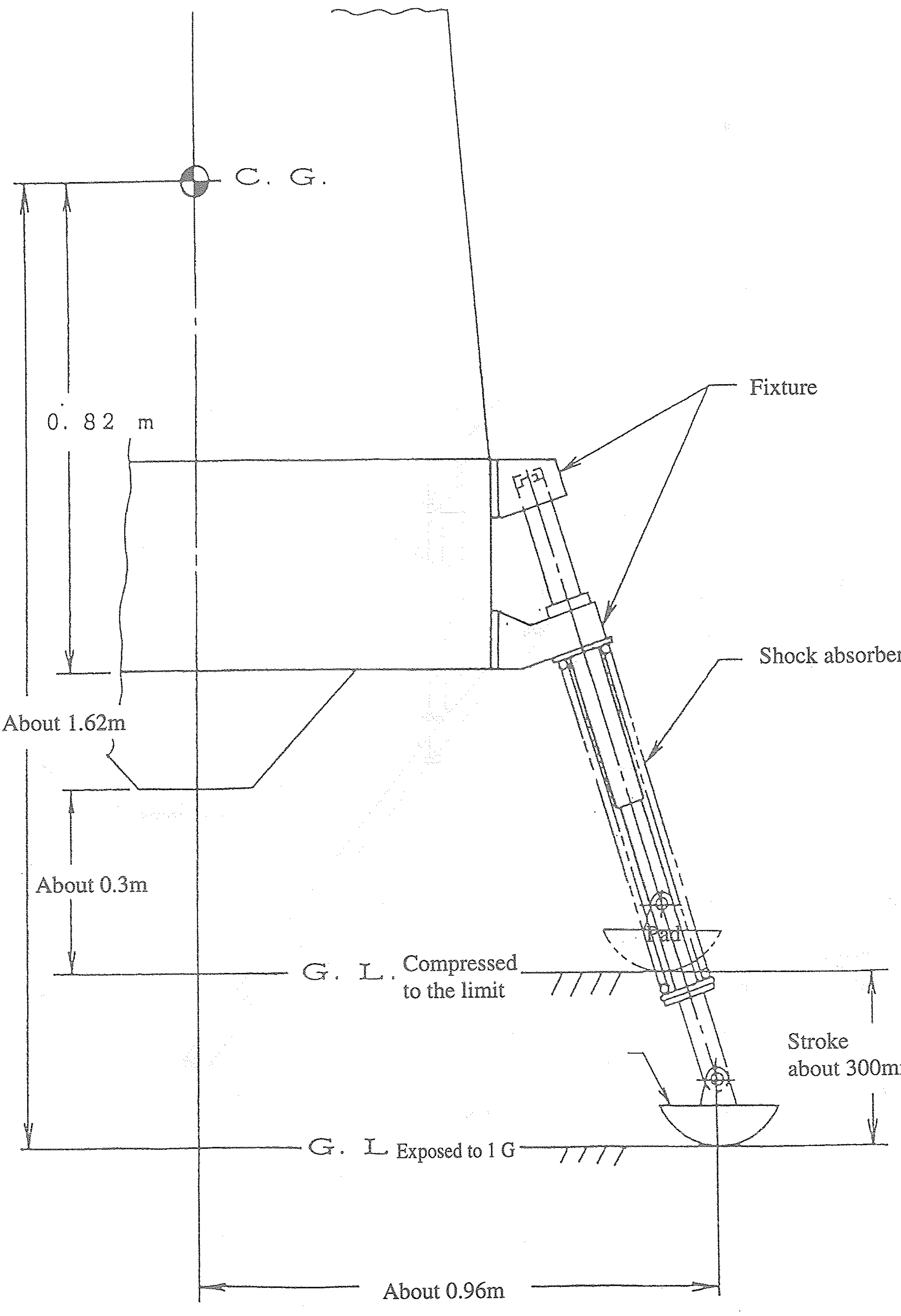


Fig.5.2.2-3 First candidate leg set option (telescopic scheme)

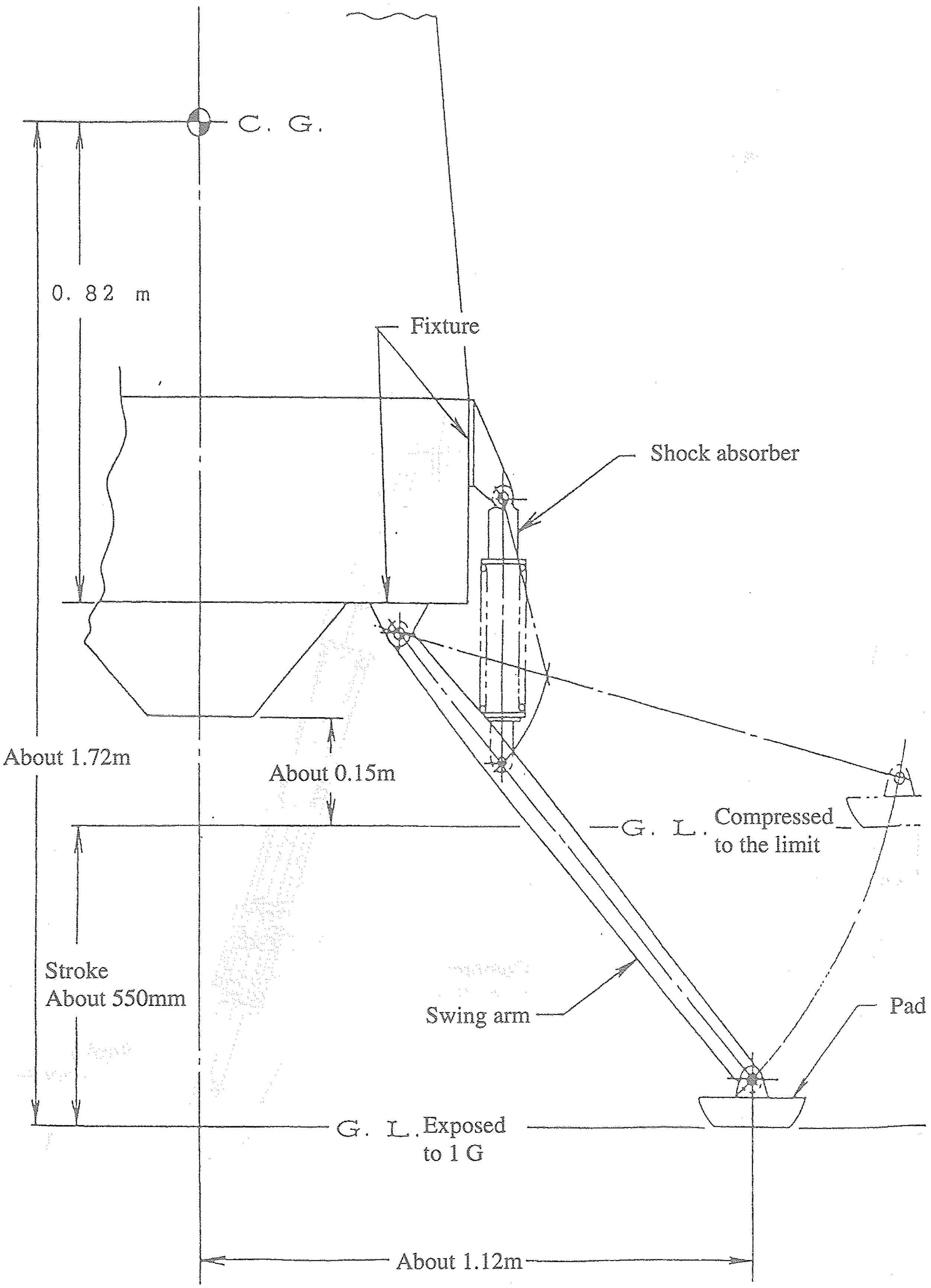


Fig.5.2.2-4 Second candidate leg set option (swing arm scheme)(1/2)

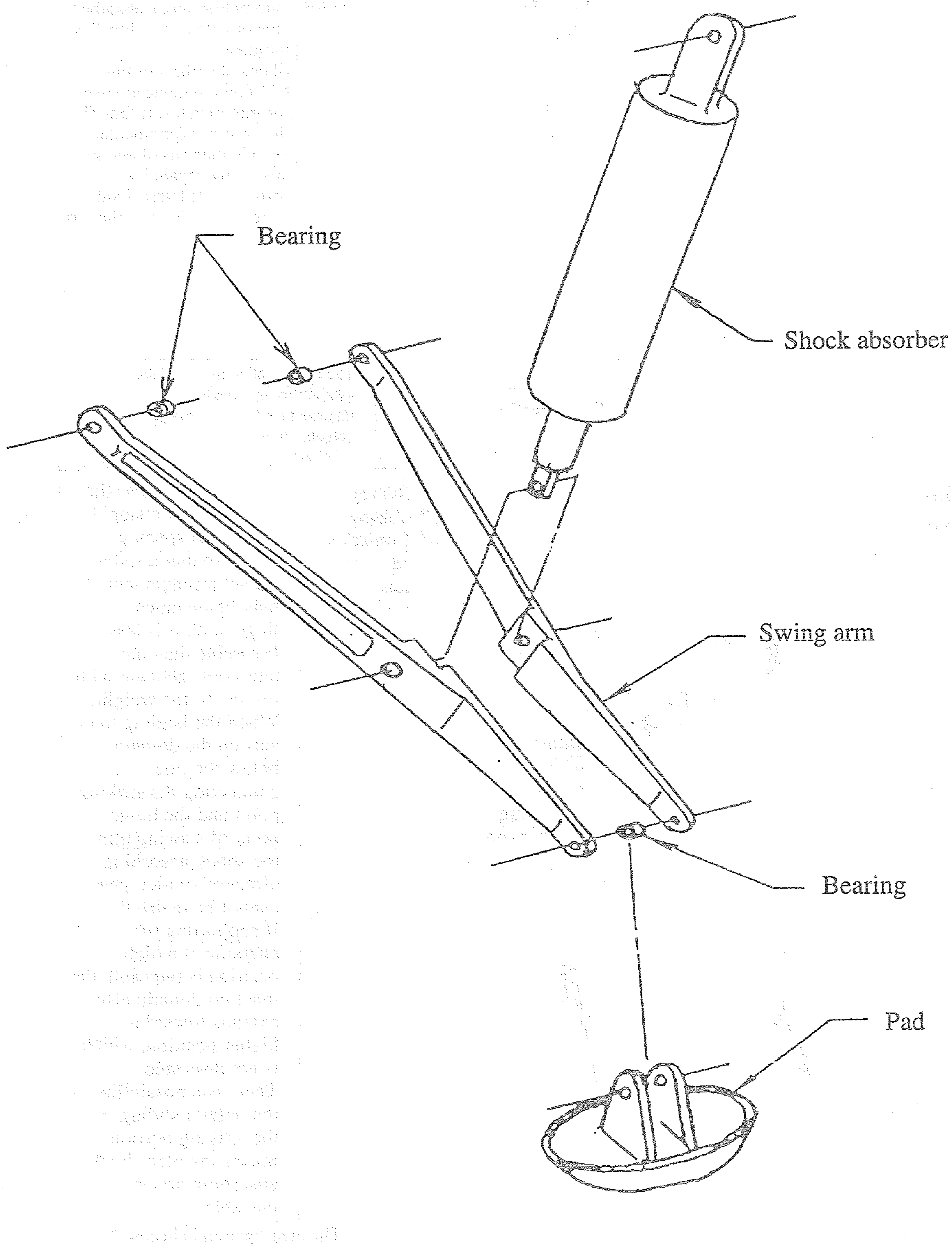


Fig.5.2.2-4 Second candidate leg set option (swing arm scheme)(2/2)

Table 5.2.2-3 Outline of the trade-off between various leg set options (1/2)

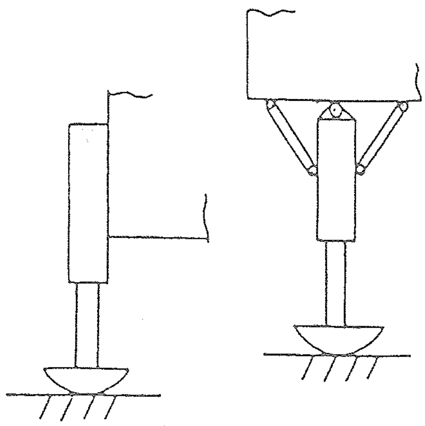
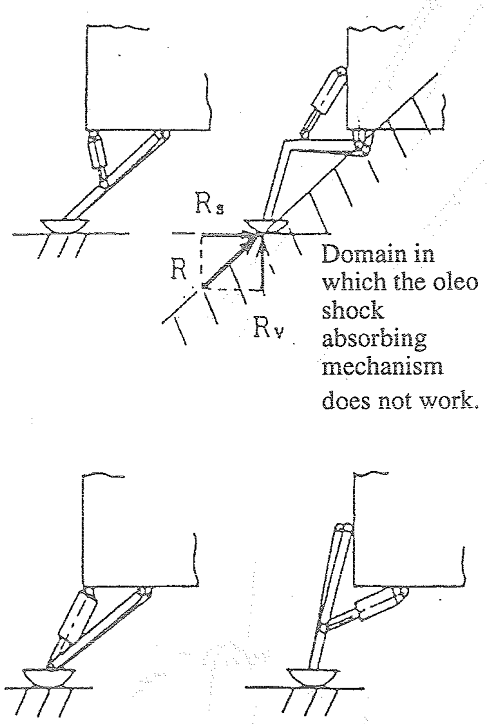
	Type	Outline	Applications	Features	Evalu-ation
Basic scheme	Telescopic	 To be the first candidate option ⇐ [The investigation indicates the availability of a ready-made product that can be adapted to the planned vehicle. (however, modification is required.)]	* DC-X * Aircraft in general * Front suspension for motorcycles	* Although the leg set is simple and lightweight, the oleo gear undergoes a bending moment at the time of landing. Consumer shock absorbers in general (oil buffers and automobile shock absorbers) cannot withstand a bending moment. Shock absorbers of this telescopic scheme are used for motorcycles; If they fit the current experimental vehicle in terms of energy absorbing capability, resistance to lateral load, weight, and the like, there is a possibility that they are adopted.	△
	Swing arm (lever suspension)	 To be the second candidate option ⇐ [The investigation indicates the availability of a ready-made product that can be adapted to the planned vehicle. (however, modification is required.)]	* Surveyor * Viking * Lunokhod * Motorcycle rear suspension	* This scheme has the advantage of changing the leg set spacing freely so that a stable leg set arrangement may be obtained. In general, it is less favorable than the telescopic scheme with respect to the weight. When the landing load acts on the domain below the line connecting the striking point and the hinge point of a swing arm, the shock absorbing effect of an oleo gear cannot be realized. If supporting the airframe at a high position is required, the inaction domain also extends toward a higher position, which is not desirable. There is a possibility that lateral sliding of the striking portion makes the oleo shock absorbing action unstable.	△

Table 5.2.2-3 Outline of the trade-off between various leg set options (2/2)

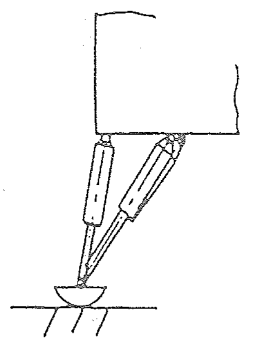
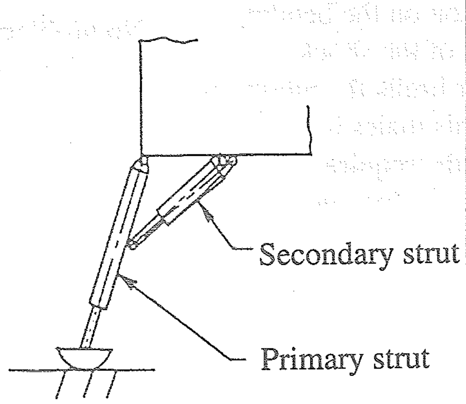
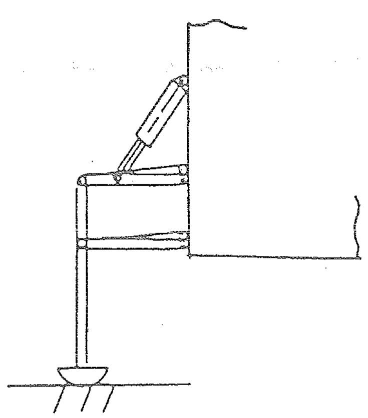
	Type	Outline	Applications	Features	Evaluation
Basic scheme	Tripod	 All three legs are equipped with an oleo gear.	* Vehicles used at the early stage of the Project Apollo	* Improved swing arm scheme. The shock absorbing action is effective to load in all directions. In addition, the absence of members to which the bending moment is applied makes it lightweight; however, a large number of oleo gears makes dynamic analysis difficult. This scheme is considered promising in the future for landers that perform high energy landing with a greatly inclined attitude or on greatly unlevelled or inclined surfaces.	×
Derived type	Cantilever Telescopic + tripod		Apollo	* Telescopic shock absorber provided with a shock absorbing function against lateral load; like the basic type, fit for high energy landing. The disadvantage is that the bending moment acts on the primary strut.	×
	* Wishbone Double swing arm		Automobile	* The shock absorber of this type is capable of preventing the bending moment to the oleo gear, to which the telescopic scheme is vulnerable, from acting. In addition, it eliminates the inaction domain associated with the swing arm scheme. Using a parallelogram link makes it possible to prevent the lateral shift of the striking point associated with the stroke form taking place. Although it is disadvantageous with respect to the weight, it makes it possible to protect the oleo gear from being heated by placing the oleo gear away from the engine jet flame.	×

Table 5.2.2-4 Outline of the results of comparison between the telescopic and the swing arm scheme

Item		Telescopic scheme		Swing arm scheme	
Perfor-mance	Landing volocity (vertical)	○	・3.0 m/s MAX	△	・2.5 m/s MAX
	Limit landing load factor	○	・3.0 MAX	○	・3.0 MAX
	Stroke	○	About 300mm	△	About 550mm
	Shock absorbing action	○	* The phenomenon described in the box to the right does not pose so serious a problem.	△	* There is a possibility that the frictional force generated by lateral sliding of the striking portion makes the shock absorber action unstable.
	Strength to the lateral load	△	* Limitation on the bending strength of the shock absorber limits the mounting angle; this males it difficult to meet the requirement of a large leg set spacing.	○	* No problem because only the axial force acts on the shack absorber.
Weight kgf/leg set	Shock absorber		12.0 (10.5) *1		8.0
	Others (pad and the like)		2.0		5.0
Total		○	14.0 (12.5) *1	○	13.0
Cost			TBD		

Notes
*1. The figures in the parentheses show the weight when springs made of Ti alloy steel wires are used.

Overall evaluation	○	△

Table 5.2.2-5 Outline of the trade-off between various shock absorber options (1/4)

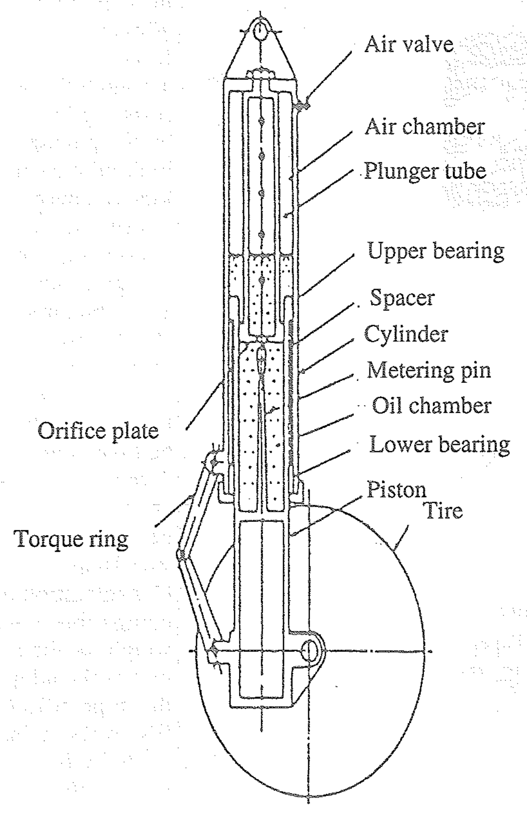
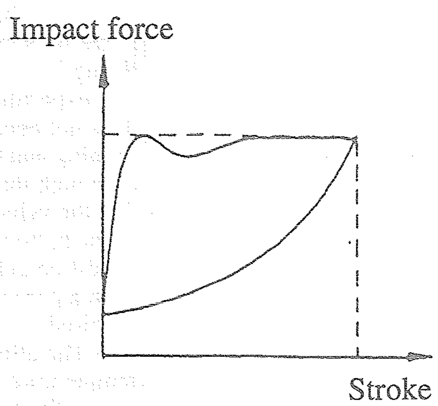
Type	Outline	Applications	Features	Evaluation
Oleo gear for aircraft	 [* The telescopic type is illustrated. * The damper for the lever suspension type has the same inner structure as the one illustrated by this figure.]  	Aircraft in general	* An oleo gear is filled with a high-pressure gas and a hydraulic oil; by controlling the orifice area with the metering pin, the optimum shock absorbing characteristics (realizing a high oleo gear efficiency) are obtained. The high-pressure gas is responsible for supporting the airframe weight empty, absorbing energy during low velocity shock absorbing, and preventing change in the attitude and bottom striking due to braking from taking place. To absorb shocks occurring at high velocity conditions such as landing and running, the viscous resistance of the hydraulic oil absorbs the energy. Aircraft use oleo gears are expensive because a high reliability and an ultimate reduction in weight are required.	×

Table 5.2.2-5 Outline of the trade-off between various shock absorber options (2/4)

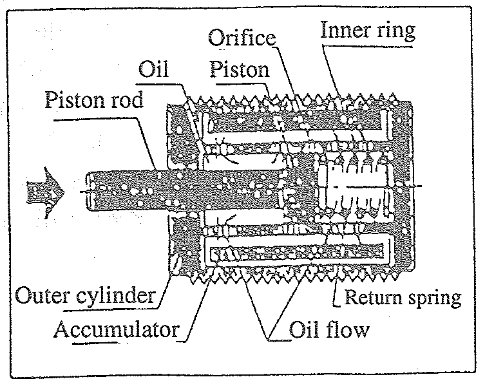
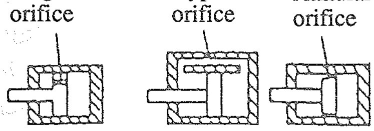
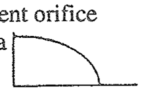
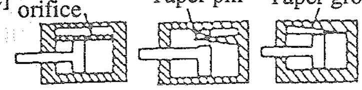
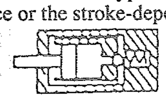
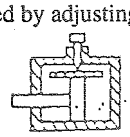
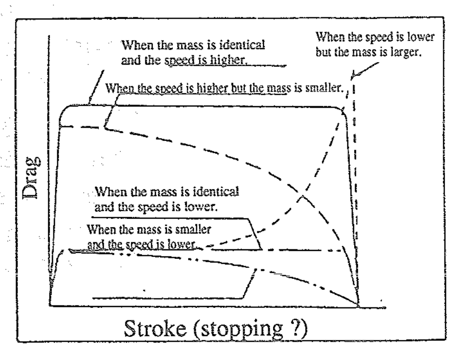
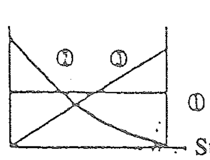
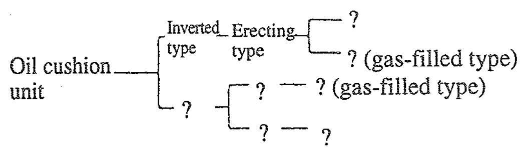
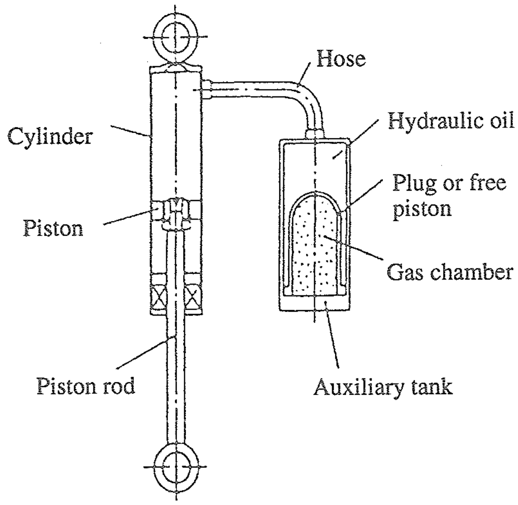
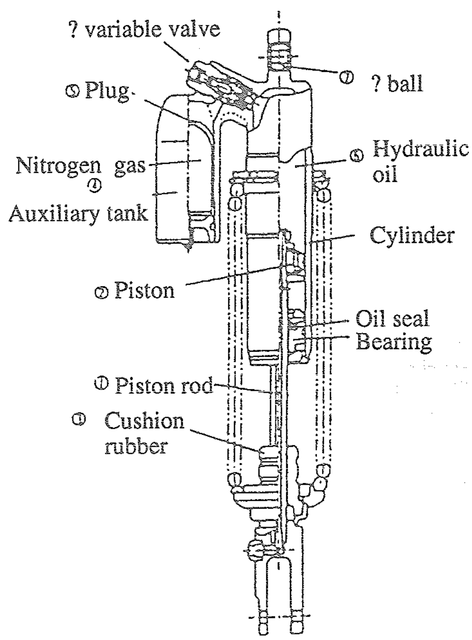
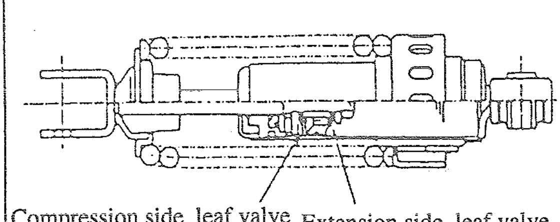
Type	Outline	Applications	Features	Evaluation
Industry use oil buffer	 1. Constant orifice type [Example] Single hole orifice Bypass orifice Annular orifice  2. Stroke-dependent orifice  [Example] Perforated orifice Taper pin Taper groove  3. Relief type Many of the orifices of the relief type are used along with those of the constant orifice or the stroke-dependent orifice type [Example]  4. Adjusting type The drag is adjusted by adjusting the orifice area. [Example]    (1) Stroke-dependent orifice type (eta = 100%) eta: Shock absorbing efficiency (2) Constant orifice type (3) Metal spring or the like	* Industrial machinery * Transfer facility * Crane * Elevator * Car coupler, collision prevention devices	* Device that absorbs the kinetic energy of a moving body and alleviates the impact force. Through the action of the resisting force generated by the passage of the hydraulic oil inside, the kinetic energy is converted into thermal energy and the kinetic energy is absorbed. <1> The direction of the shock absorbing action is limited to that of the piston stroke (the bending moment cannot be taken out. Adopting a leg set of the swing arm type is necessary. <2> Heavy. Consideration of the compatibility with the weight of the airframe makes the adoption of this type difficult. EX. A fixed damper made by Enidain, INC. weighs about 15 kgf/ unit. <3> The spring force is just enough only to return the piston to its original position; therefore, making a leg set freestanding requires a large-scale modification such as equipping the legs with powerful springs or gas filling. The experimental vehicle does not need ground running and braking. Although this suggests that the vehicle be used in the bottom striking condition , changes for such a purpose is required. <4> The allowable temperature is low (about 80 degrees C). It is possible to raise the allowable temperature up to about 120 degrees C by changing types of hydraulic oil and packings.	×

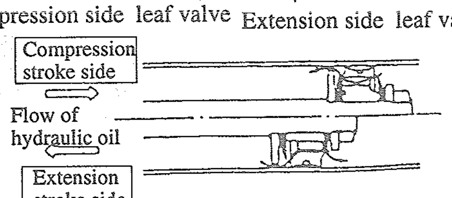
Table 5.2.2-5 Outline of the trade-off between various shock absorber options (3/4)

<table>
<tr><th>Type</th><th>Outline</th><th>Applications</th><th>Features</th><th>Evalu-
ation</th></tr>
<tr>
<td>Off-road motorcycle shock absorber</td>
<td>






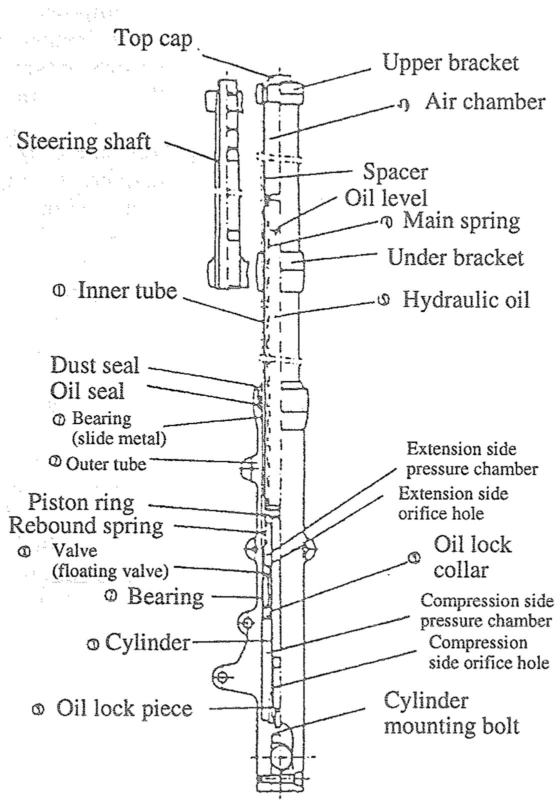


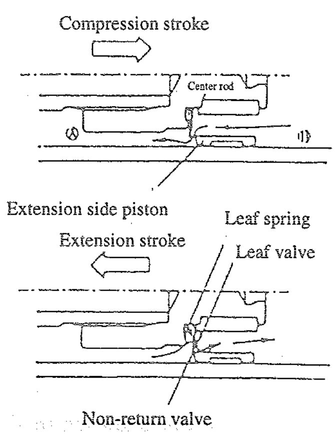


</td>
<td>* Motorcycle rear suspension For swing arm suspension</td>
<td>* Composed of a spring and a damper and absorbs vibration form the road surface.
The spring supports the airframe weight and converts the input vibration into elastic energy to alleviate vibration.
The damping force of the damper converts the elastic energy of the spring into thermal energy to alleviate vibration.
Since the spring supports the static load, a powerful spring is needed, which increases the weight; in addition, the progressive characteristics are difficult to obtain.

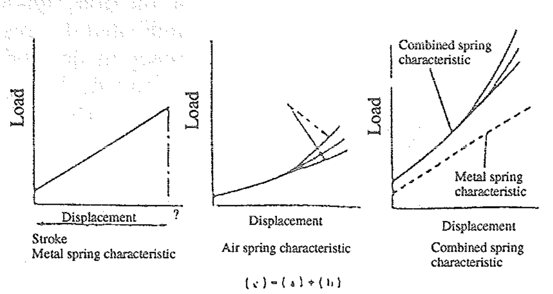
As is shown in Table 5.2.2-4, the investigation has indicated the availability of ready made products that almost fit the experimental vehicle. (however, modification is necessary.)
⇩
⇩
⇩
To be the second candidate option.</td>
<td>△</td>
</tr>
</table>

Table 5.2.2-5 Outline of the trade-off between various shock absorber options (4/4)

<table>
<tr><th>Type</th><th>Outline</th><th>Applications</th><th>Features</th><th>Evaluation</th></tr>
<tr>
<td>Off-road motorcycle shock absorber</td>
<td>




Action of the extension side piston when the compression stroke is on. When the damper is on the compression stroke, the non-return valve and leaf valve are lifted to supply the hydraulic oil from the pressure chamber <B> to the pressure chamber <A>.

Action of the extension side piston when the extension stroke is on. When the damper is on the extension stroke, the hydraulic oil flows from the pressure chamber <A>, passes the hole of the non-return valve, bends the outer periphery of the leaf valve supported by the inner periphery, passes the hole of the extension side piston, and reaches the pressure chamber <B>.


To be the first candidate option
⇩
For the outline of the candidate option, refer to Fig.5.2.2-5.
</td>
<td>* Motorcycle front suspension For the telescopic suspension use</td>
<td>The shock absorbing principle and structure is identical to that of the rear suspension shock absorber, except that the lateral strength is secured by increasing the lateral rigidity.

The investigation indicates the availability of a ready-made product that can be adapted to the planned vehicle. (however, modification is required.)</td>
<td>△</td>
</tr>
</table>

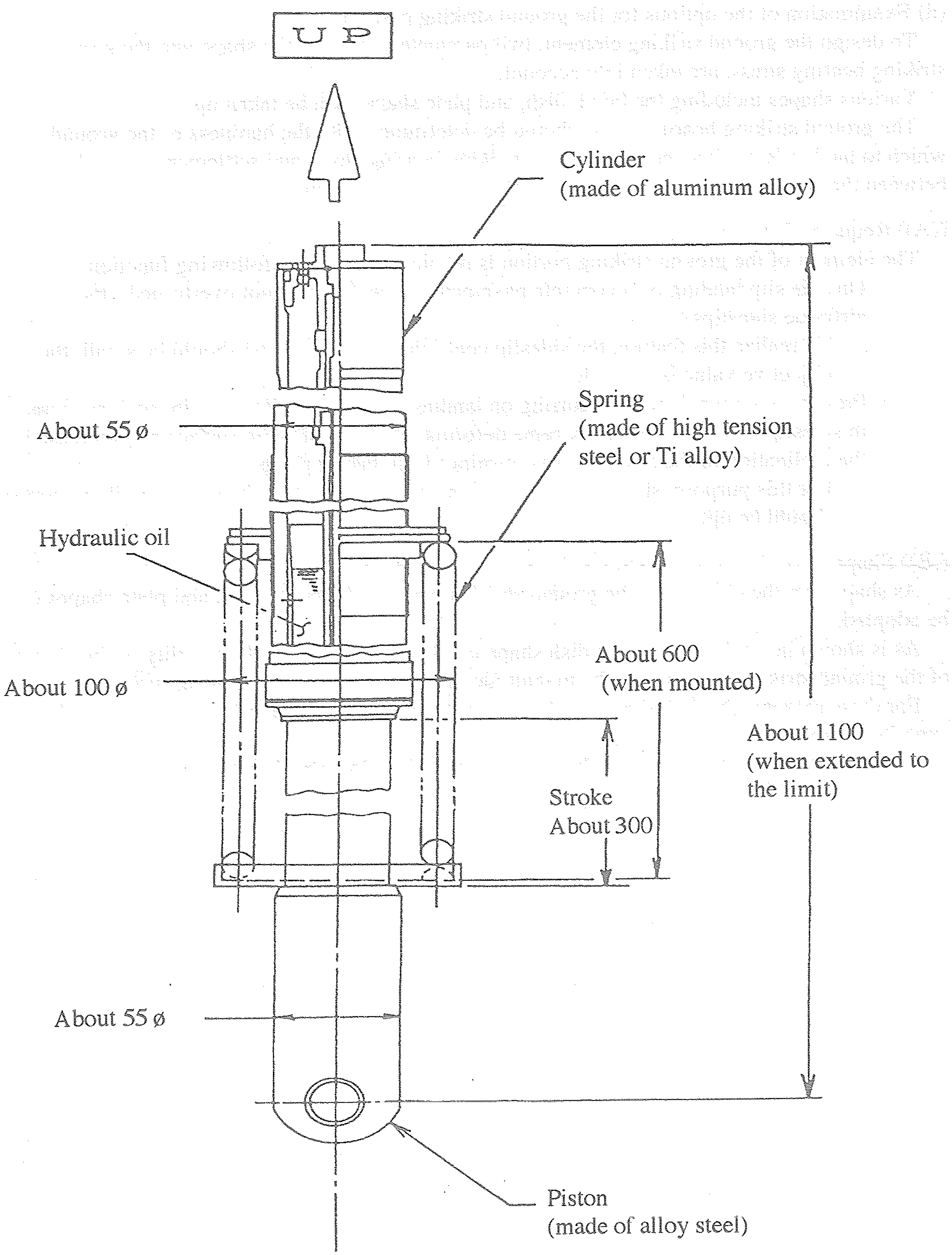


Fig.5.2.2-5 Outline of the candidate components for the shock absorber (telescopic type)

(d) Examination of the options for the ground striking portion

To design the ground striking element, two parameters, namely the shape and the ground striking bearing stress, are taken into account.

Various shapes including the bowl, dish, and plate shapes can be taken up.

The ground striking bearing stress should be determined with the hardness of the ground on which to land, relation between the ground striking bearing stress and settlement, and relation between the settlement and side slip drag taken into consideration.

((A)) Required functions

The element of the ground striking portion is required to have the following functions:

<1> On side slip landing on a concrete pavement, the airframe is not overturned. (the airframe sideslips.)

To realize this feature, the sideslip coefficient of friction (μ) should be small; the objective value is $\mu<=0.4$.

<2> Prevents inconveniences, occurring on landing on an unlevelled terrain, such as increase in sideslip resistance due to extreme deformation of the striking surface and increase in the inclination of the airframe (overturning) from taking place.

For this purpose, shapes and dimensions of the element of the ground striking portion should be optimized.

((B)) Shape

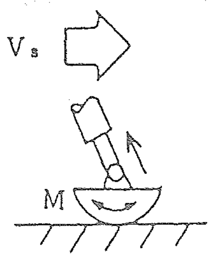
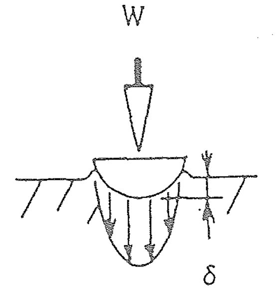
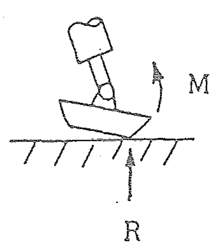
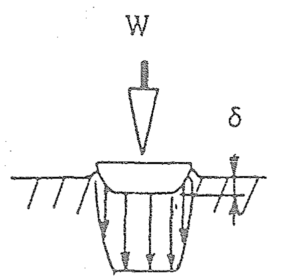
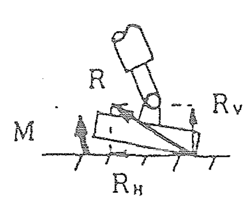
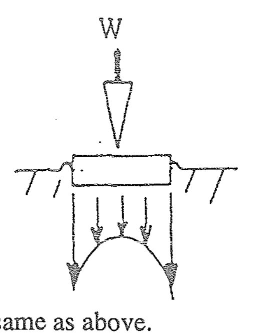
As shapes for the element of the ground striking portion, the bowl, dish, and plate shapes can be adopted.

As is shown in Table 5.2.2-6, the dish shape is superior in terms of the stability of the element of the ground striking portion and the magnitude of deformation of the striking surface.

For these reasons, the dish shape is adopted for the ground striking portion element of the experimental vehicle.

Most of other vehicles similar to the experimental one adopt the dish shape.

Table 5.2.2-6 Trade-off between shape of the element of the ground striking portion

No	Shape	Stability	Deformation of an unlevelled terrain	Evaluation
1	Bowl shape	 * Since the element of the ground striking portion tends to rotate when the shock absorber is actuated or the vehicle lands sideslipping, a mechanism for stopping this (such as stoppers) becomes necessary.	 * Because of the uneven bearing stress on the ground striking surface, delta becomes large.	×
2	Dish shape	 * Even if the element of the ground striking portion strikes the ground with some inclination, a moment that returns the element to a stable attitude (that is parallel to the striking surface) works, making the system stable.	 * The bearing stress distribution on the striking surface becomes almost uniform and the value of delta is small.	○
3	Plate shape	 * The same as above. However, as the sideslip resistance becomes larger, an unstable moment as shown in the figure above is generated and makes the action of the element unstable.	 * The same as above. * On sideslipping, soil removal becomes poor, causing the drag to increase; in particular, when an unstable moment described in the left box acts, the tip of the element of the ground striking portion digs into the soil, increasing the drag.	×

(e) Weight of the leg set

((A))Estimation of the weight of the leg set

The weight of the leg set depends greatly on the leg set arrangement, type of leg set, type of shock absorber used, and restrictions on cost.

On the first candidate option (leg set of the telescopic type; a front suspension shock absorber for off-road motorcycles is used as a shock absorber.)(refer to Figs.5.2.2-3 and 5.2.2-5), the weight is estimated.

The estimated weight for the leg set is 14.0kgf/leg set and 56.0kgf/vehicle.

The weight by component is shown in Table 5.2.2-7.

Table 5.2.2-7 Estimated leg set weight

Component designation	Number of pieces/leg set	Unit weight (kgf/unit)	Weight per leg set (kgf/leg set)
Shock absorber	1	12.0	48.0
Pad, fixing bolts, and others	1 set	2.0	8.0
Upper fixture	1	Included in the weight of the structure.	
Lower fixture	1		

Total weight of a leg set	14.0 kgf

Total weight of 4 leg sets	56.0 kgf

((B)) Weight reduction for the leg set

Following the request from the system consideration, weight reduction for the leg set was considered from the viewpoint of realizing the entire vehicle.

As a result, it was made clear that weight reduction leads to considerable increase in cost. Although the problem of the availability of materials requires further investigation, the material for the shock absorber spring is changed from a high tension wire to a Ti alloy wire (Ti-6Al-4).

Accordingly,

the weight of the leg set	= (14 - 1.5) kgf/unit x 4 units
	= 12.5kgf/unit x 4 units
	= 50 kgf/vehicle

(5) Outline specifications of the leg set

Table 5.2.2-8 shows the outline specifications for the leg set option.

Table 5.2.2-8 Outline specifications for the leg set option

Item		Draft specifications	Remarks
Leg set arrangement	Quantity	* The 4 legs required by the experimental vehicle are arranged on a regular square.	
	Spacing	* 1.36m (the length of a side of the regular square)	* The length of the diagonal of the regular square is 1.92m long.
	Overturning angle	* 67 ª	
	Ground clearance	* Stationary attitude on the ground: about 0.4m * With the legs compressed to the limit: about 0.3m	* Height of the engine jet nozzle from the ground
Type of the leg set		* Telescopic type	* Refer to Fig.5.2.2-3.
Type of the shock absorber		* Front suspension shock absorber for off-road motorcycles (telescopic tyep)	* Refer to Fig.5.2.2-5.
Composition of the leg set		* Composed of shock absorbers, pads, and other components.	* Refer to Fig.5.2.2-3.
Landing surface		* Normally, on a flat concrete pavement * Emergency landing on an unpaved level ground (sandy terrain) being taken into consideration.	
Maximum takeoff and landing weight		* 550kgf (landing with a single leg being considered)	
Landing velocity	Vertical	* 3 m/s MAX	
	Horizontal	* To be taken into consideration for normal landing on a concrete surface. * Not to be taken into consideration for emergency landing on sandy terrain and the like; however, cross-wind (50m/s) encountered at the time of landing is taken into consideration.	The sliding coefficient of friction is assumed to be <= 0.4.
Limit landing load factor		* 3.0 MAX	
Stroke		* About 300mm	
Weight (kgf/vehicle)		* 50.0 *2	* Excluding the weight of the fixtures
Heat control		* Heat is controlled as required in order to keep the temperature of the leg set within the allowable limit.	

Notes:
1. The system design conditions for the leg set (leg set design conditions) comply with the conditions shown in Table 5.2.2-1.
*2. Weight of the vehicle using the Ti alloy steel springs. The weight of the vehicle using high tension wire springs is 56.0kgf/vehicle.

5.2.3 Thermal Control

(1) Outline

With a view to summarizing the basic concept for the thermal control system (including the thermal protection system) that maintains the temperature of the onboard equipment and airframe of the vertical takeoff and landing experimental vehicle within an allowable temperature range during the entire flight experiment mission, examination was made on the thermal control system.

In the course of the examination, the requirements for the thermal control system were arranged and established, and with the thermal environment occurring before takeoff and during the flight and landing taken into consideration, the thermal control/thermal protection methods for different sections were drafted.

With the experimental vehicle, in particular, the thermal protection is important that protects the airframe bottom and landing gears from being heated by the engine plume (at the time of landing, in particular); therefore, emphasis was placed on the examination on this subject.

(2) Requirements

Possible requirements to be imposed on the thermal control system of the vertical takeoff and landing experimental vehicle were identified and arranged into the draft requirements, which are listed in Table 5.2.3-1.

(3) Design

Table 5.2.3-3 lists the outlines of the thermal control system design to meet the requirements described in the preceding paragraph (2).

In addition, Fig.5.2.3-2 shows the design flow for the thermal control system.

(4) Thermal analysis and its examination

In the following, the details of the thermal analysis and subsequent exam,inaction on it, conducted to determine the design specifications mentioned in the preceding paragraph (3), are given:

(a) Thermal control before takeoff

<1> Ground thermal environment

The weather condition in the vicinity of Obihiro in August is taken as the ground thermal environment. (experiments conducted during the summer when the outside air temperature is high provide the high temperature criteria.)

The draft ground thermal environment conditions are given in Table 5.2.3-5.

<2> Examination of the thermal control scheme

During the interval from installation on the catapult to takeoff, the vertical takeoff and landing experimental vehicle is exposed to heating by the following heat sources:

* Irradiation of the sunlight;
* Heat generated by the onboard equipment (electronic equipment, the battery, and others).

Table 5.2.3-1 Requirements for the thermal control system

Item	Details	Remarks
Functional requirement	To maintain the temperature of the onboard equipment and airframe of the vertical takeoff and landing experimental vehicle within an allowable temperature range during the entire flight experiment mission.	As shown in Table 5.2.3-2
Weight	Not more than 10kg	Weight allocated to the thermal control system
Electric power	TBD	For measurement of temperature
Reusability	To be capable of withstanding not less than 10 repeated flights (objective: to be capable of withstanding not less than 100 repeated flights accompanied by proper maintenance and repair) The flight time per mission is about 60 seconds.	
Environmental resistance	To be free from damage caused by the environmental conditions occurring in the course of flight and other operations. (sound, vibration, shock, and rainfall, in particular)	

Table 5.2.3-2 Design temperature conditions for the onboard equipment (draft)

Item		Allowable temperature range	Estimated design temperature range	Temperature margin
Hydrazine tank		5 ~ 40	10 ~ 35	5*
NTO tank		5 ~ 40	10 ~ 35	5*
GHe tank		5 ~ 40	10 ~ 35	5*
Airframe structure		~ 100	~ 85	15
Onboard electronic equipment		-10 ~ 55	5 ~ 40	15
Landing gear	Hydraulic section	~ 80	~ 65	15
	Spring section	~ 120	~ 105	15

Note) * The outer wall temperatures of the gas containing portions that affect the tank pressure were taken as the specified temperatures. Since it was impossible to allow sufficient design margins for the values, a tentative value of 5°C was adopted.

Table 5.2.3-3 Outlines of the thermal control system design

Item	Requirement	Details of design
Functional requirement	To maintain the temperature the onboard equipment and the entire airframe of the vertical takeoff and landing experimental vehicle within an allowable temperature range (Table 5.2.3-2) during the entire flight experiment mission	* To maintain the temperature of the onboard equipment and various sections of the airframe within an allowable temperature range by using the thermal control/thermal protection scheme shown in Fig.5.2.3-1.
Weight	Not more than 10kg	* Total weight of the thermal control system = 6.6kg (estimated value) * Details of weight by component are given in Table 5.2.3-4.
Electric power		* TBD(The details to be examined in the future)
Reusability	To be capable of withstanding not less than 10 repeated flights (objective: to be capable of withstanding not less than 100 repeated flights accompamied by proper maintenance and repair) The flight time per mission is about 60 seconds.	* To enhance the reusability, the following design approaches are applied to the airframe bottom and landing gears that are exposed to severe heating by the plume: 1) Airframe bottom To obtain a high durability, the surface of the thermal protecting material (flexible thermal insulating material) is covered with highly heat resisting stainless steel foil, the surface of which in turn is protected by heat resisting paint (allowable temperature up to 600°C). 2) Landing gear To obtain a high durability, highly heat resisting Ti alloy is adopted as the heat shield material, and the heat shield surface in turn is protected by heat resisting paint (allowable temperature up to 600°C)
Environmental resistance	To be free from damage caused by the environmental conditions occurring in the course of flight. (sound, vibration, shock, and rainfall, in particular)	1) Sound, vibration, and shock The thermal insulating materials are designed so as to withstand the sound, vibration and shock environments throughout the flight mission from takeoff via hovering to landing. (details to be examined in the future) 2) Rainfall The flexible thermal insulating material used in the airframe bottom is designed so as to be waterproof. (penetration of water may impair the thermal insulating effect.)

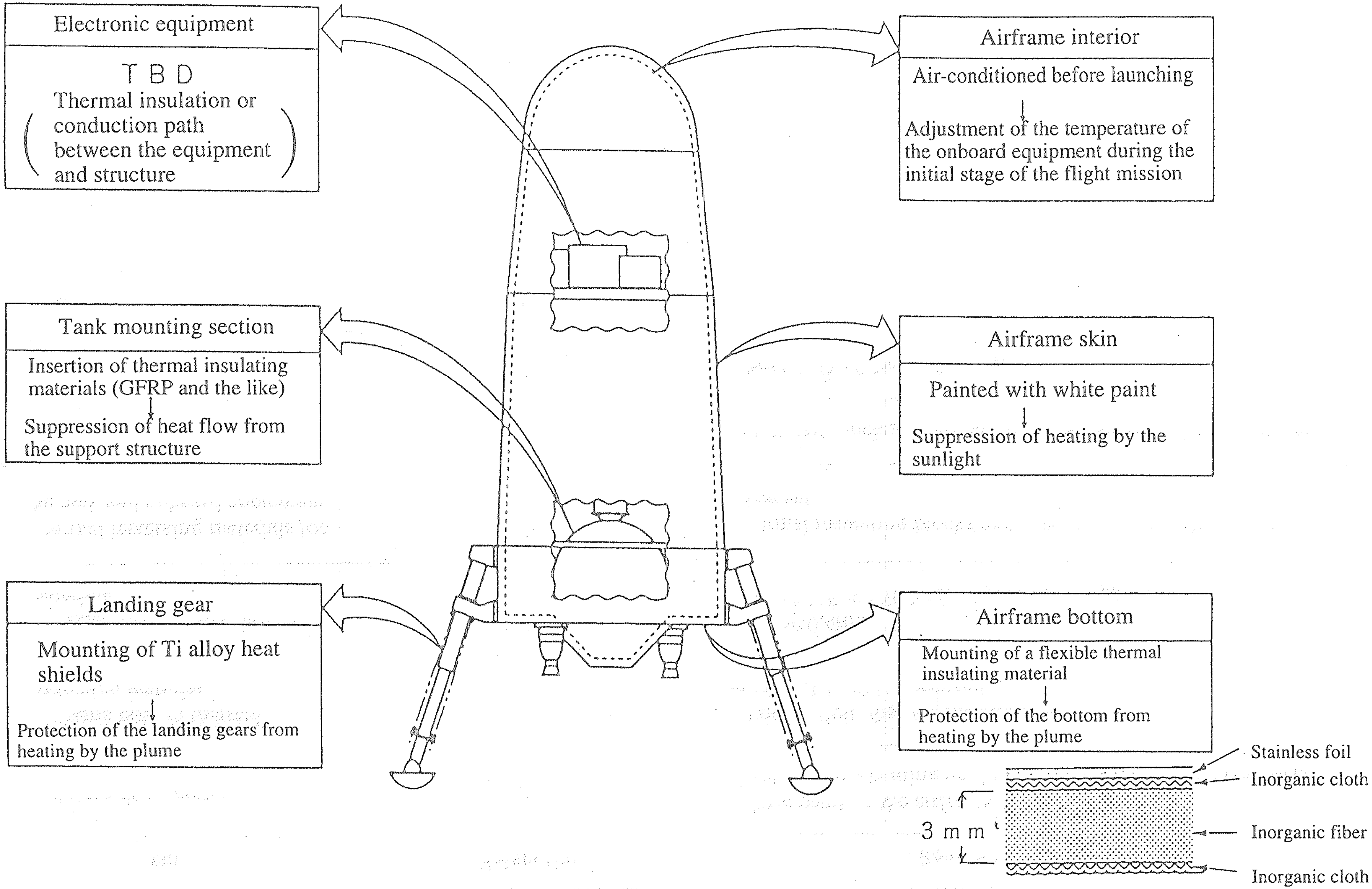


Fig.5.2.3-1 Thermal control/thermal protection scheme (draft)

Table 5.2.3-4 Weight of the thermal control system by component

Item	Weight (kg)	Remarks
Airframe skin painting	1.6	* White paint on the airframe skin * Heat resisting coating on the airframe bottom and landing gears
Airframe bottom thermal protecting material	1.8	* Stainless foil, thermal insulating materials (an area of 1 m2 is supposed.)
Landing gear thermal protection materials	1.6	* Ti alloy 0.4mm t A tube of 100 ø x 700mm (4 pieces) is supposed.
Thermal insulating materials for the tank and onboard equipment	0.5	* Thermal insulating blocks and spacers for the tank/onboard equipment
MSP	0.5	* Small parts such as thermal protecting material fixing fasteners
Margin	0.6	* About 10% of the total weight
Total	6.6	

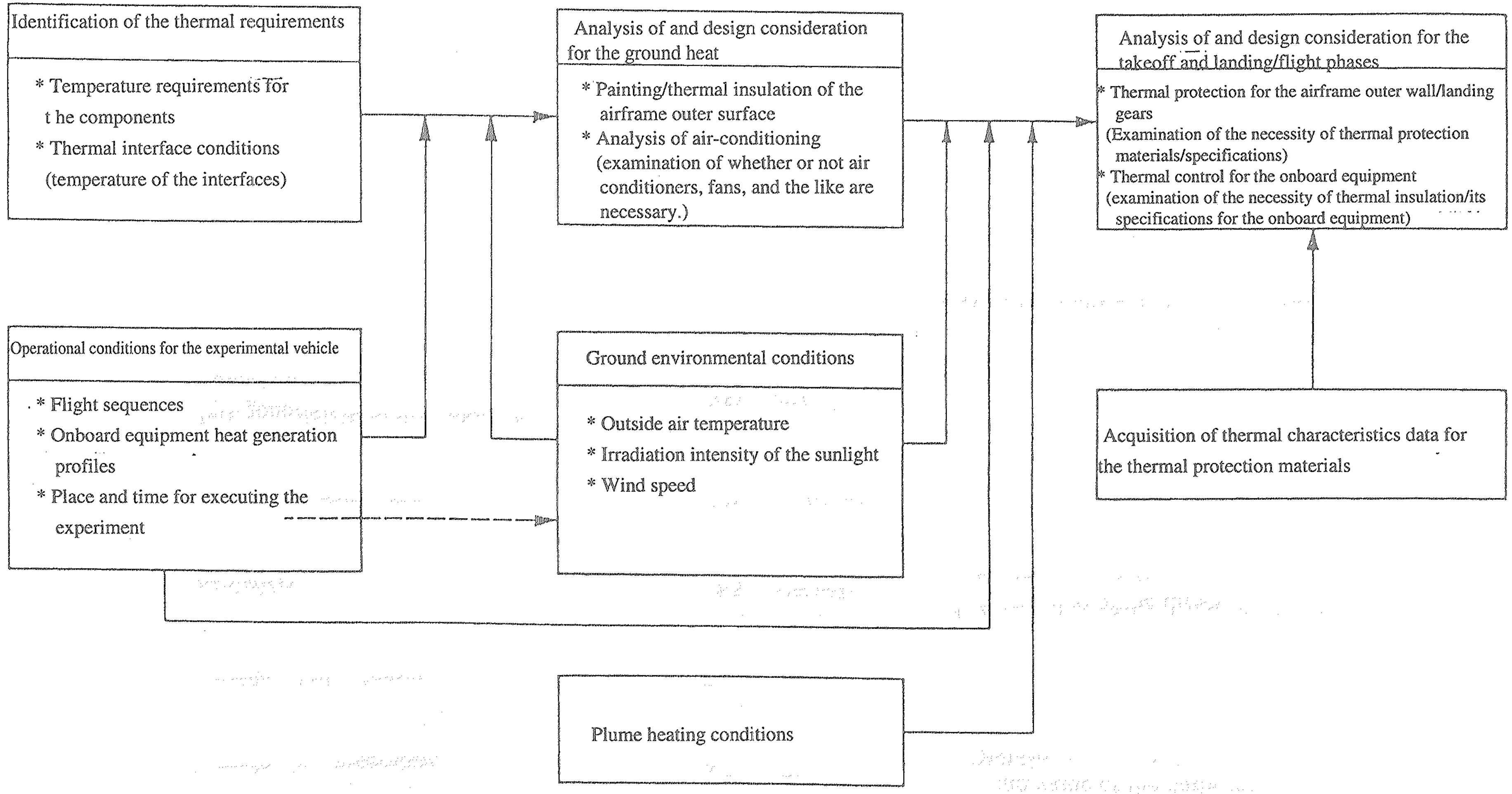


Fig.5.2.3-2 Thermal analysis and design flow for the vertical takeoff and landing experimental vehicle (draft)

Table 5.2.3-5 Ground environmental conditions (draft)

Item	Environmental conditions	Remarks
Outside air temperature	24.7 (°C)	* Mean value of the daily maximum temperature in Obihiro in August ✤
Outside wind velocity	TBD (m/s)	
Insolation	0.85 (kW/m^2)	* Mean value of the direct insolation in Nemuro in August ✤
Ground exposure time	TBD (Hours)	
Heat generation by the onboard equipment	TBD (kW)	

Note) ✤ According to the Rika nenpyo

For this reason, it is anticipated that maintaining the temperature inside the vehicle below the allowable temperatures for the onboard equipment (tanks, in particular) will become difficult unless thermal control is provided.

In addition, because the temperatures of different parts of the experimental vehicle rise during the flight mission, it is desirable to cool them down as much as possible before takeoff.

For these reasons, the thermal control schemes described below are considered effective for the experimental vehicle:

1) White painting on the outer surface of the airframe
   In order to minimize heating caused by the irradiation of the sunlight, the airframe outer surface is coated with low absorptance white paint (αs nearly equal to 0.2).
2) Air conditioning inside the airframe
   Temperature control of the onboard equipment is provided by the air conditioning unit (AGE).

<3> Estimation of the initial temperature at the time of takeoff

The initial temperature at the time of takeoff is approximately estimated in the following manner:

1) Parts which are not irradiated by the sunlight: 25°C
   (Members composing the airframe bottom, onboard equipment, tanks, and the like)
2) Parts which are irradiated by the sunlight: 40°C
   (Airframe skin, landing gears, and the like)

The initial temperatures discussed above will be examined more in detail when the conditions for the operating environment, air conditioning, and internal heat generation are determined.

(b) Thermal protection/thermal control during the flight mission

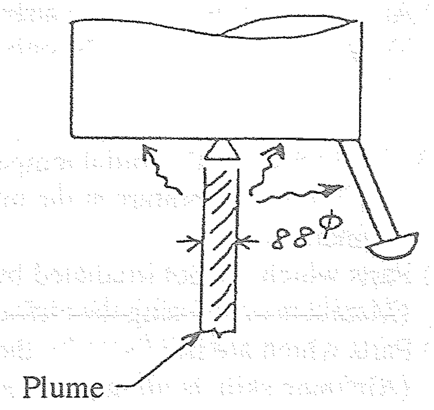
<1> Estimation of heating by plume

1) Heating by plume during the period from takeoff to hovering

The same plume radiation heating rate as that to be observed during the flight is supposed on the assumption that the plume generated during takeoff is discharged to the exhaust opening.

The plume radiation heating rate during flight is approximately calculated as follows:

* When a rocket engine is blasting into the atmosphere, it is anticipated that the plume hardly expands in the vicinity of the nozzle and takes a shape like a column. (refer to the figure to the right.)
Under this condition, the radiation heating rate is at about 5 (kW/m^2). (result of approximate calculation)



Conditions for the calculation

a) Plume temperature: 2000 (K) (substituted for by the temperature of the plume at the nozzle outlet.)
b) Plume shape: 88mm ø x 1m
c) Plume emissivity: 0.1 (substituted for by that of the flame of liquid fuel)

Since the actual plume temperature is expected to be lower than the temperature at the nozzle outlet, the calculation is considered to be on the safe side.

2) Heating by the plume during descent and landing (heating by flame)

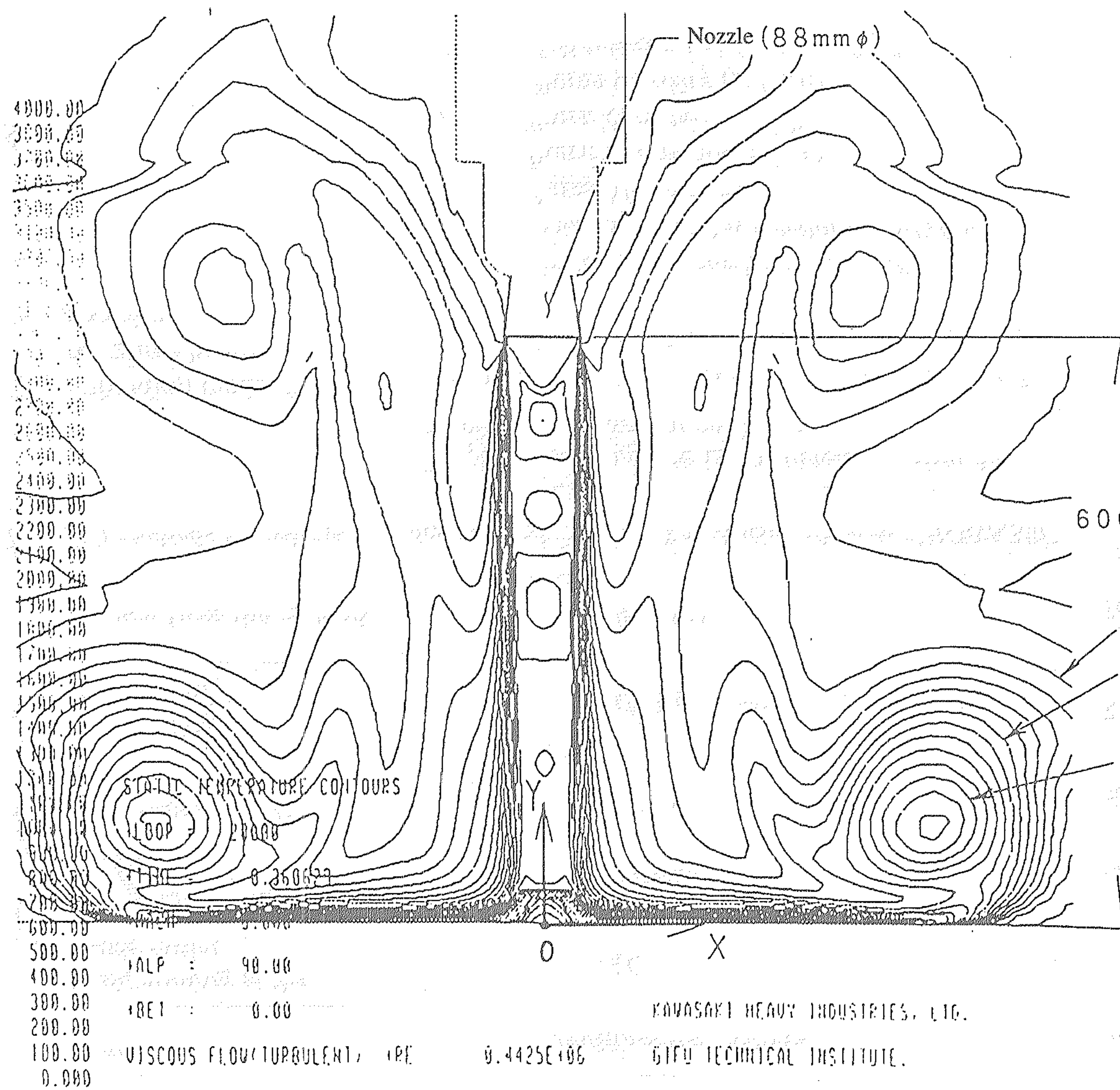
The estimation procedure flow for the plume heating rate during descent and landing (about 10 seconds including descent and landing) is as follows:

Engine characteristics data	⇒	Results of the CFD analysis (with the ground taken into consideration)	⇒	Estimation of the heating rate

Table 5.2.3-6 shows the conditions for analysis of heating by the plume during descent and landing; in addition, Figs.5.2.3-3 through 5.2.3-8 show the results of the CFD analysis.

Table 5.2.3-6 Conditions for analysis of heating by the plume

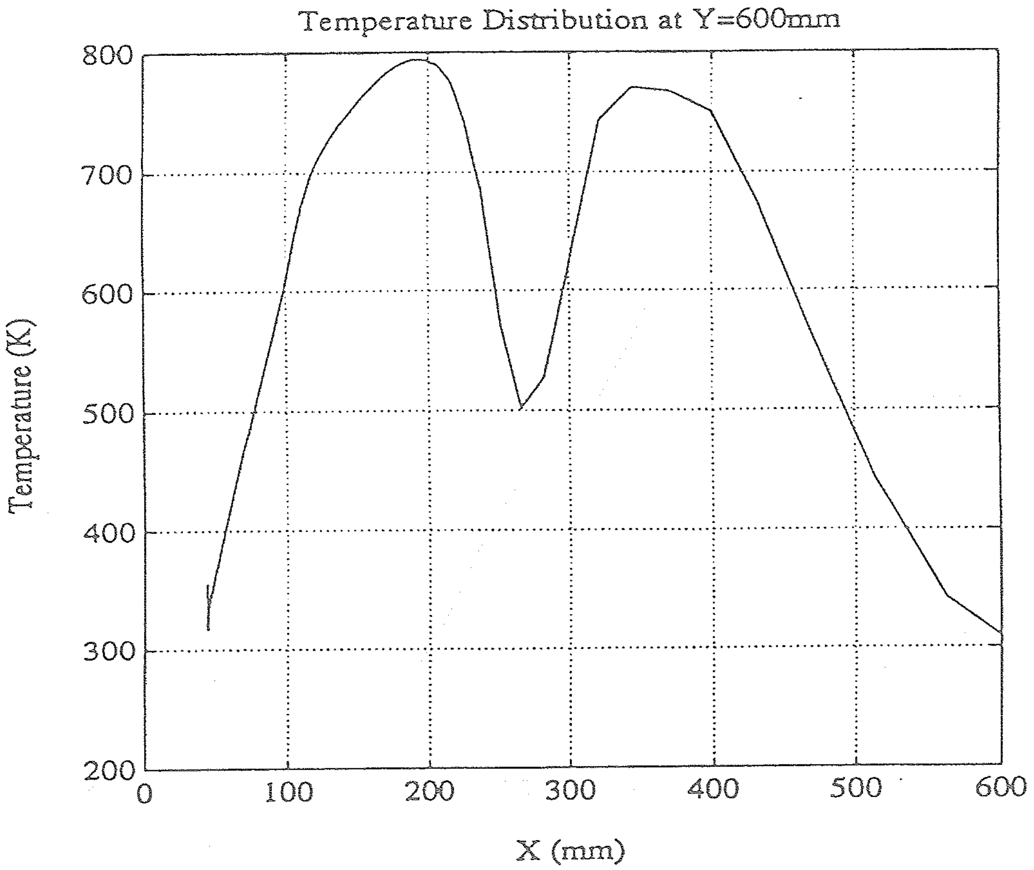
Item		Condition for analysis	Remarks
Engine characteristics	Mach number at the nozzle outlet	2.16	EVE engine characteristics
	Temperature of the nozzle outlet	2000 (K)	EVE engine characteristics
	Angle at the nozzle outlet	15 (°)	EVE engine characteristics
	Pressure at the nozzle outlet	0.83 (kg / cm^2)	EVE engine characteristics
	Distance from the ground	600 (mm)	Temporary condition
CFD	CFD analysis technique	According to the KHI analysis software "ENMA3D"	Fig.5.2.3-3 shows the results of the CFD analysis.
Heating intensity analysis	Estimation method for the plume heating intensity	The plume heating rate is estimated by using the following equation for analysis: $Q = 1.49 \times 10^{-9} / (\rho_{SL} \cdot X)^{0.5} \times [1 + 5/(0.85 \cdot M_1^2) \times (1 - T_W / T_1)] \times T_1^{0.383} \times \rho_1^{0.5} \times U_1^{2.39}$ ($BTU / ft^2 \cdot sec$) ρ_{SL} : Air density at sea level ($1b / ft^3$) X : Distance from the nozzle(ft) → 0.6m M_1 : Plume Mach number T_1 : Plume temperature (° R) ρ_1 : Plume density ($1b / ft^3$) U_1 : Plume velocity (ft / sec) T_W : Wall temperature(° R) →20(°C)	Source: J. SPACECRAFT Vol.6, No.3, March 1969 The results of analysis of the heating rate are as follows: * Airframe bottom: Fig.5.2.3-9 * Landing gears: Fig.5.2.3-11



Conditions for CFD analysis

Item	Condition for analysis
Mach number at the nozzle outlet	2. 16
Temperature at the nozzle outlet	2000 (K)
Angle at the nozzle outlet	15 (°)
Pressure at the nozzle outlet	0. 83 (kgf/cm²)
Height from the ground	600 (mm)

Fig.5.2.3-3 Results of the CFD analysis of the plume at the time of landing (Example)



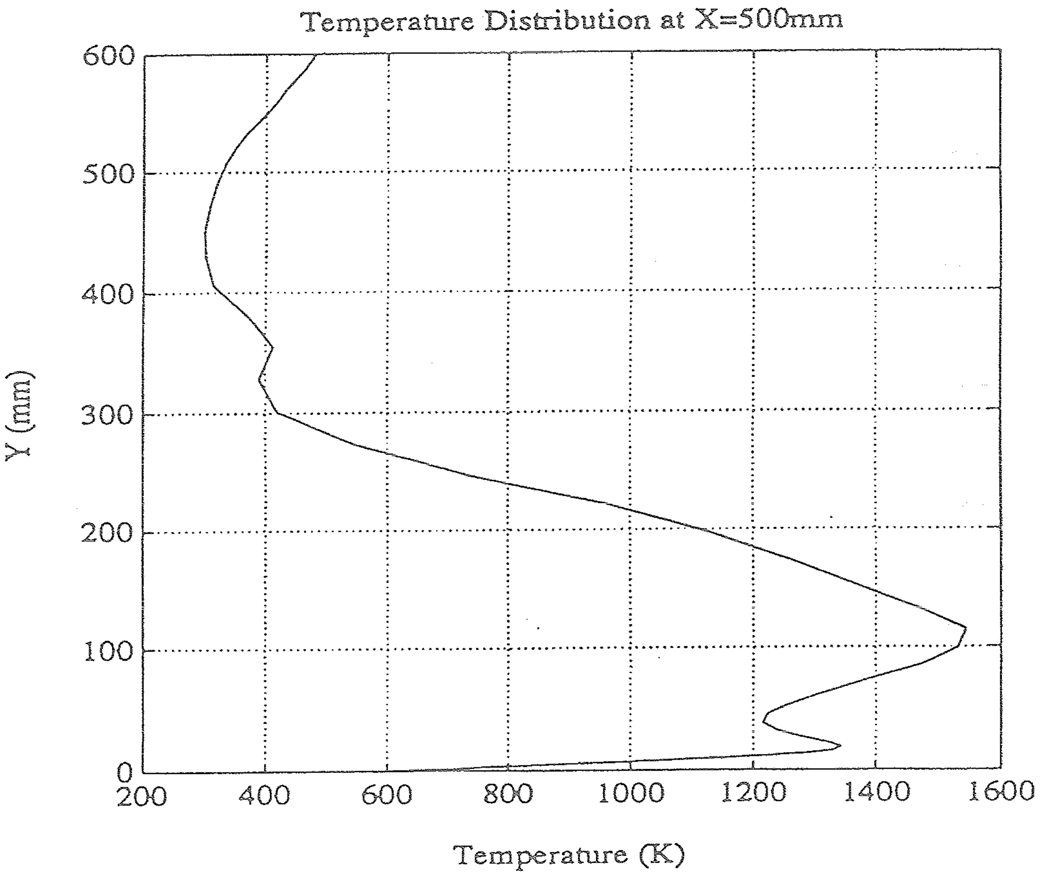
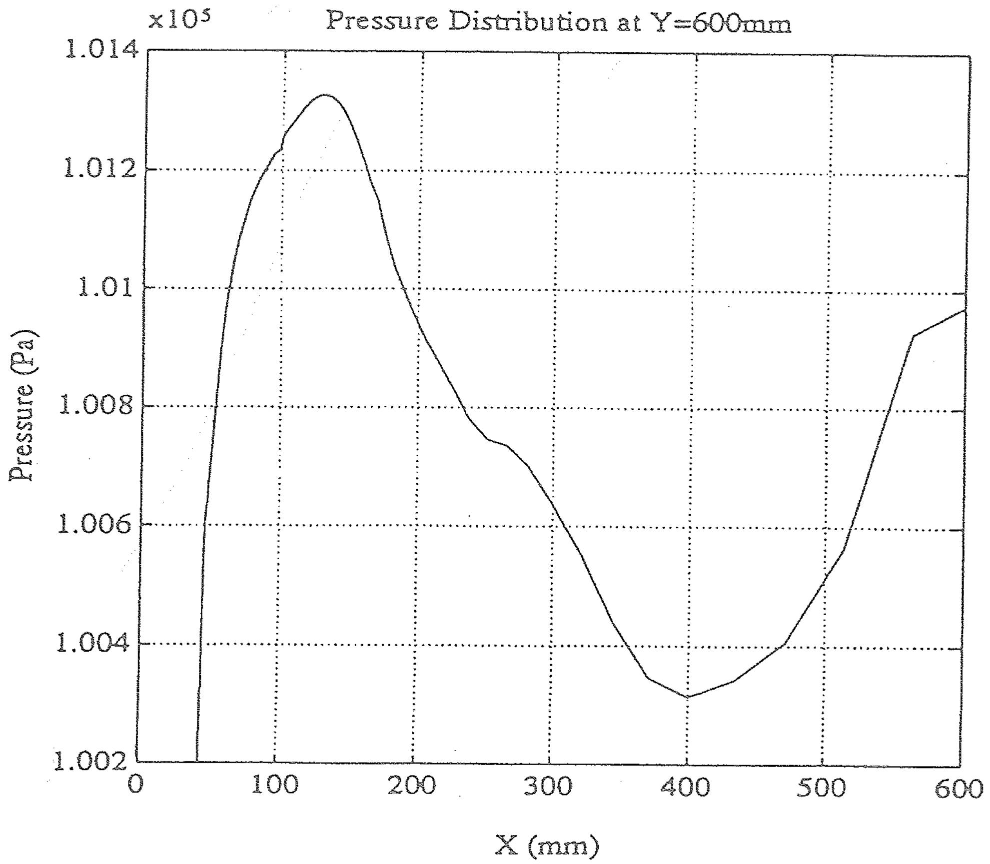


Fig.5.2.3-4 Result of the CFD analysis (temperature distribution)



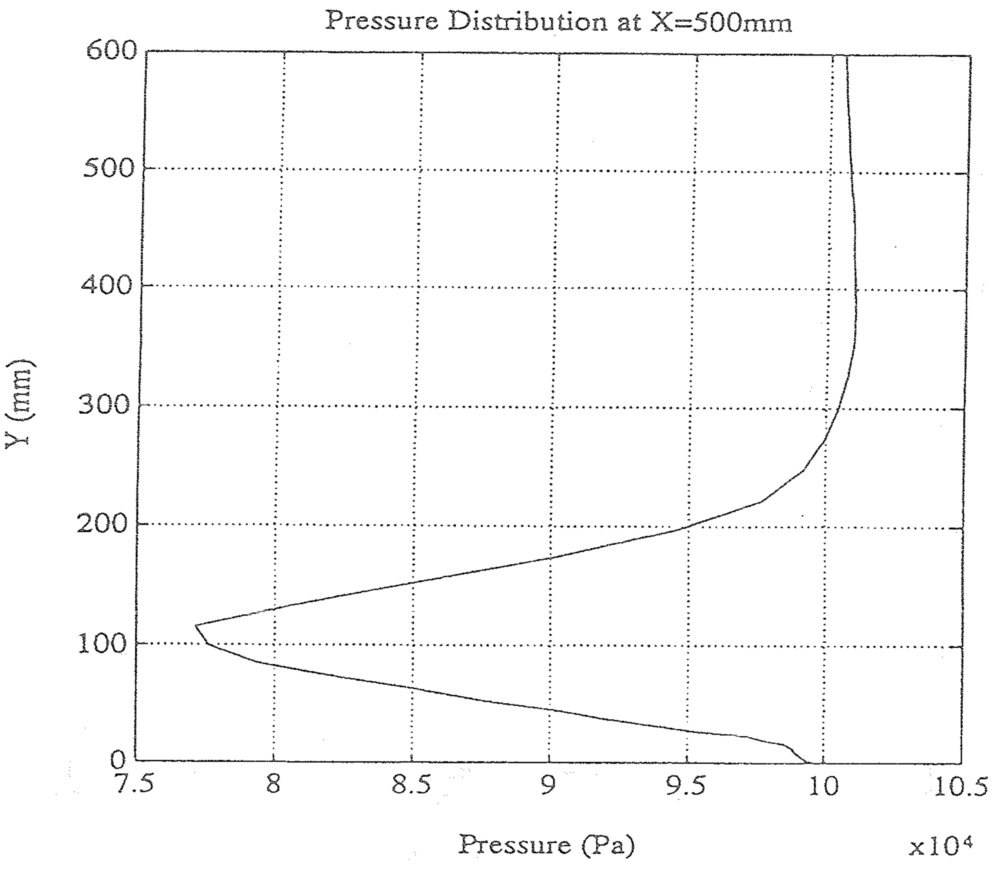
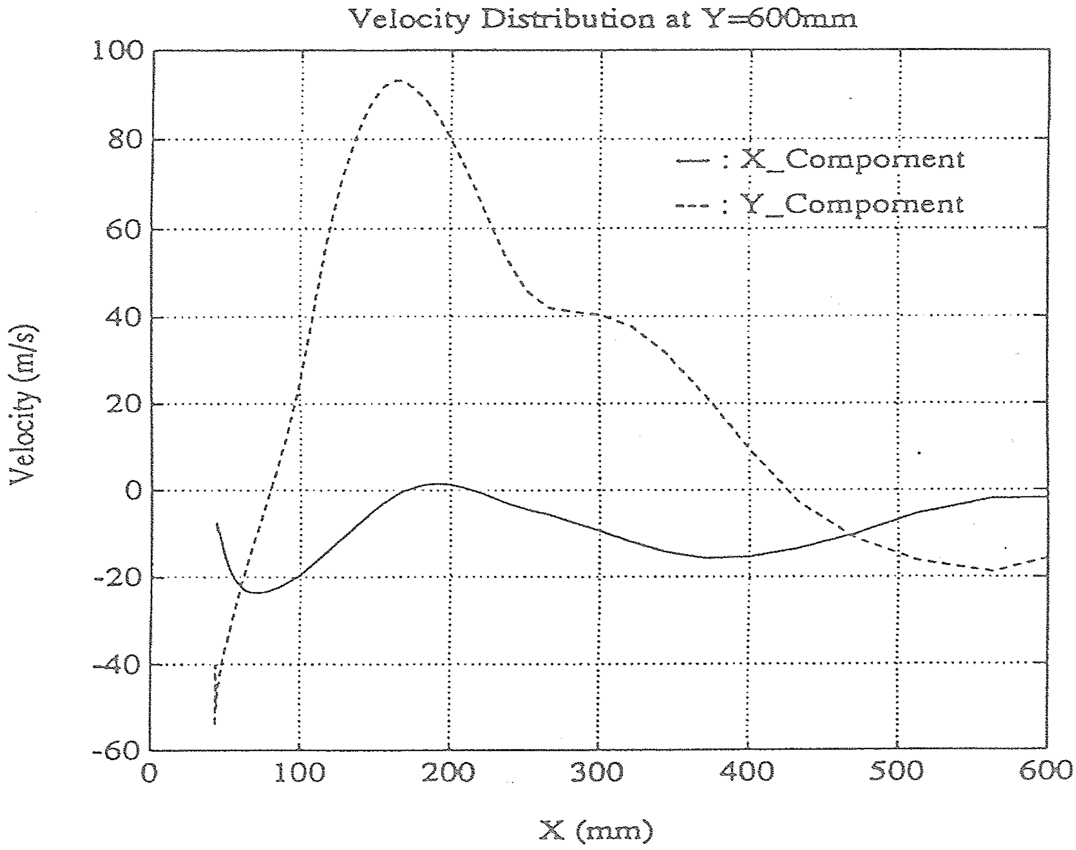


Fig.5.2.3-5 Result of the CFD analysis (pressure distribution)



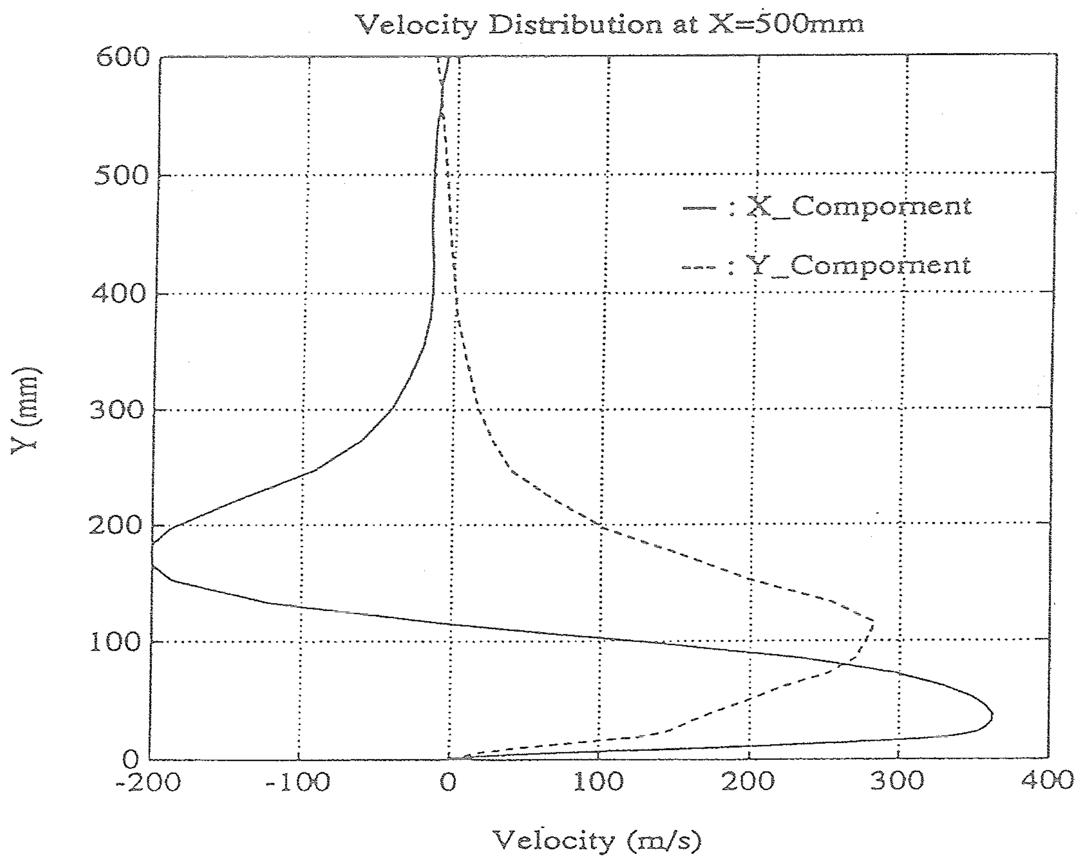
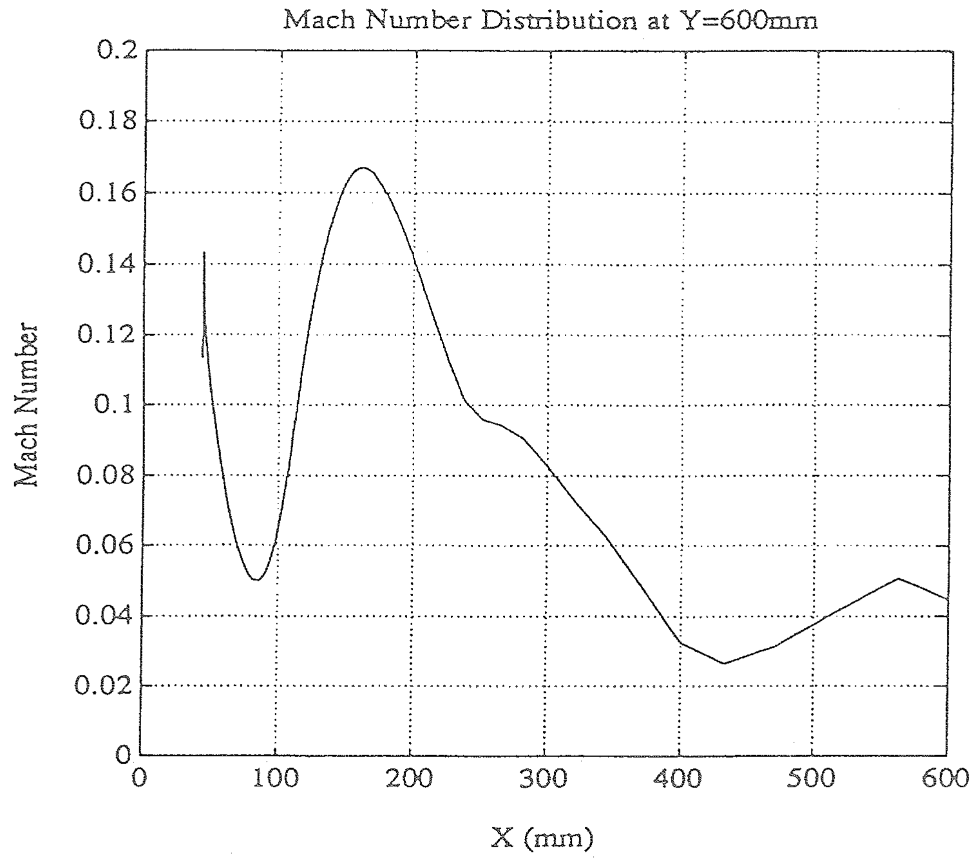


Fig.5.2.3-6 Result of the CFD analysis (velocity distribution)



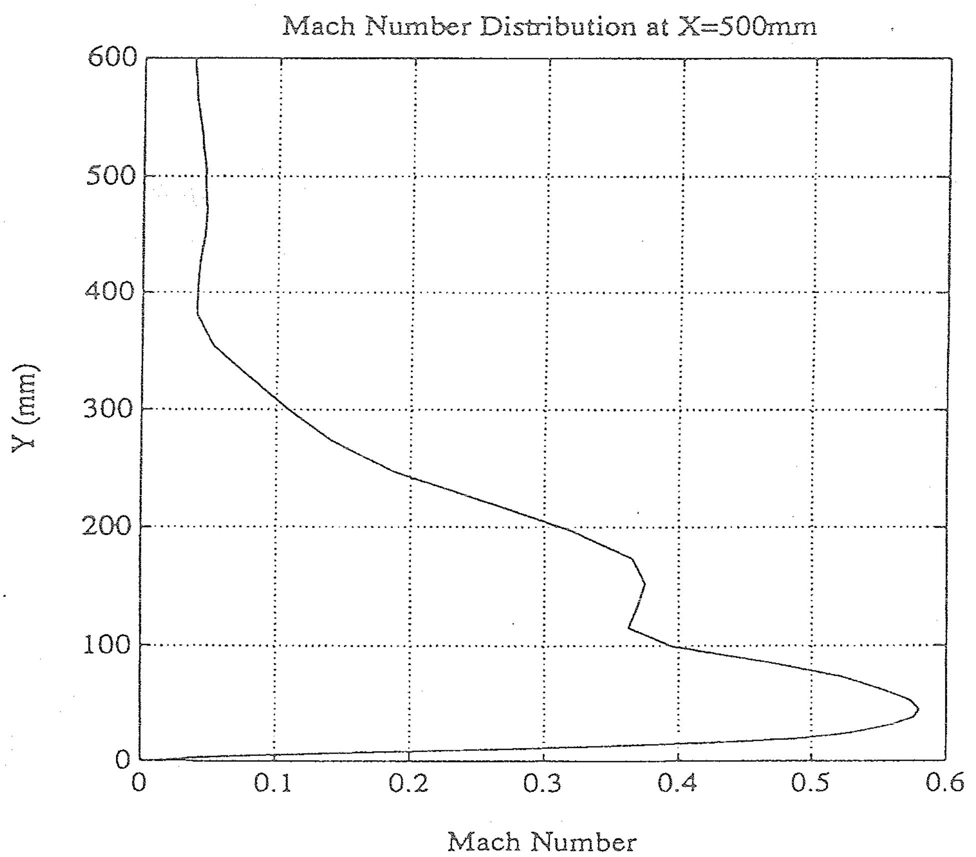
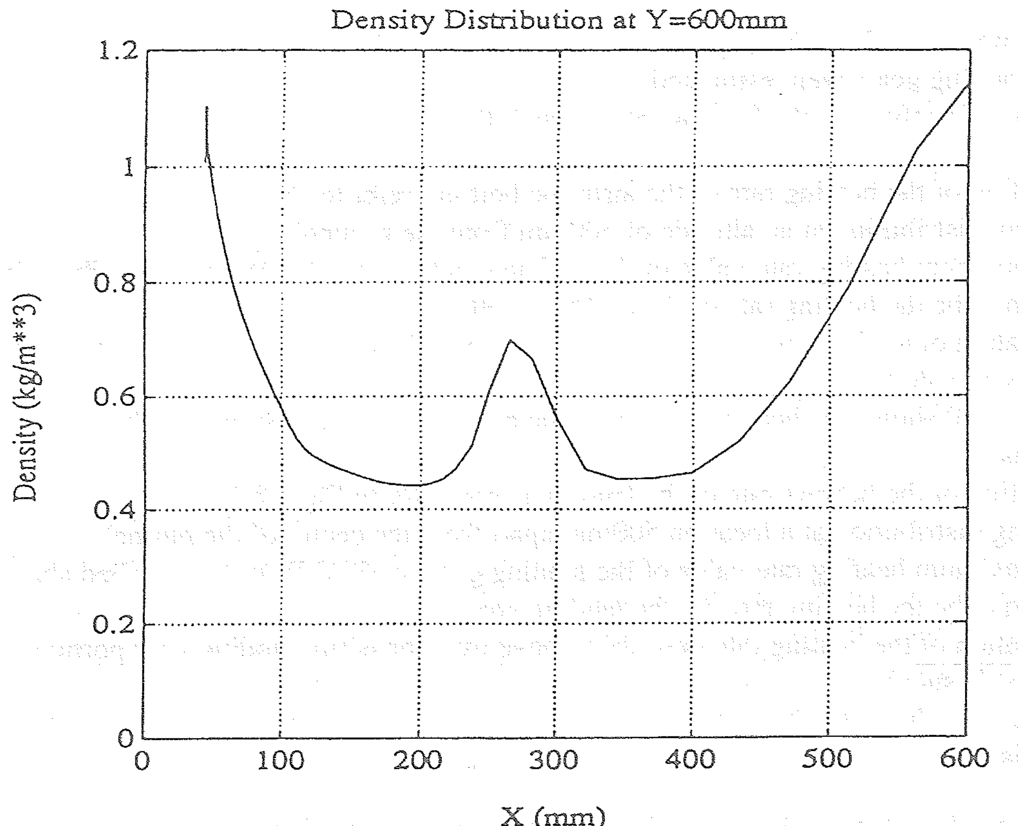


Fig.5.2.3-7 Result of the CFD analysis (Mach number)



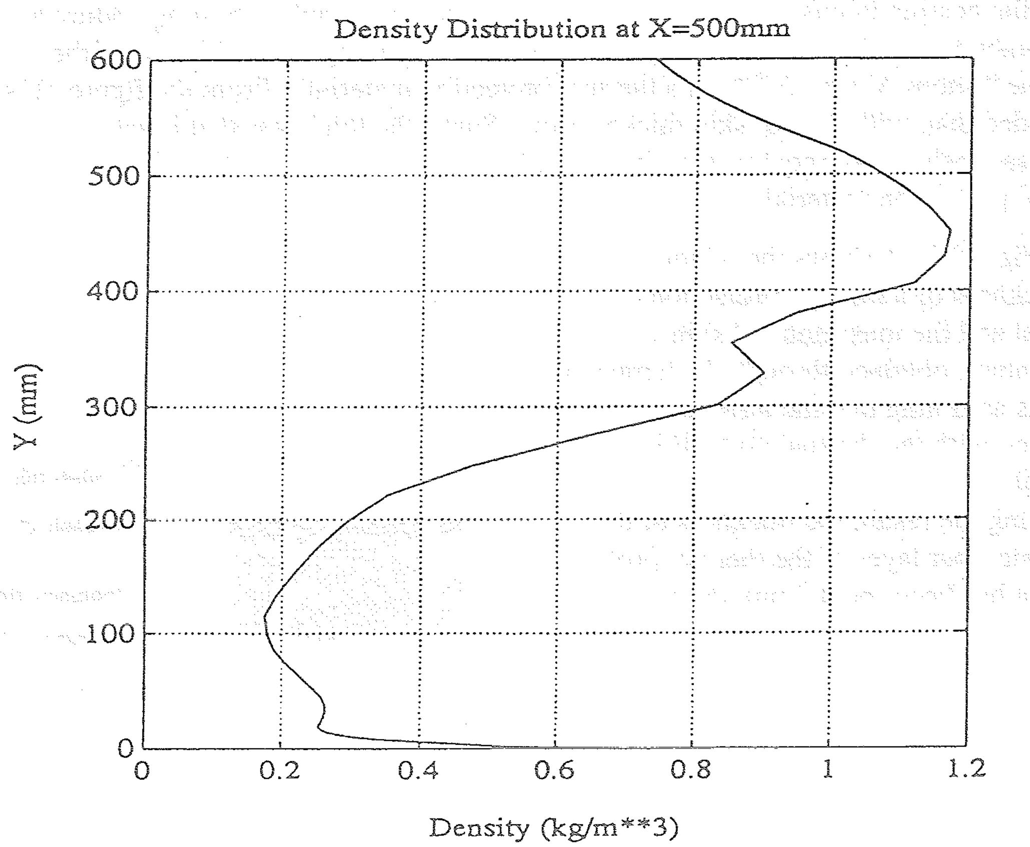


Fig.5.2.3-8 Result of the CFD analysis (density distribution)

Using the results of the CFD analysis described above, the heating rate for the airframe bottom and landing gears were estimated.

The results of analysis of the heating rate are as follows:

a) Estimation of the heating rate of the airframe bottom: refer to Fig. 5.2.3-9.
(Heating distribution at an altitude of 600mm from the ground)
The maximum heating rate value of the airframe bottom of 14 (kW/m^2), described above, is taken to be the heating rate of the airframe bottom.
(Utilization of the heating rate described above over the entire airframe bottom surface is on the safe side.)
Fig.5.2.3-10 shows the heating history of the airframe bottom assumed in the present analysis.

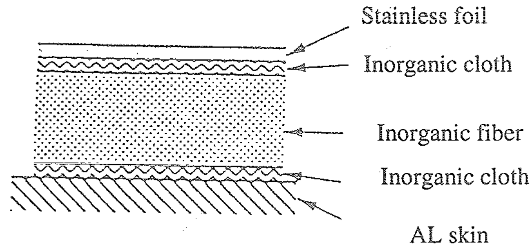
b) Estimation of the heating rate of the landing gears: refer to Fig.5.2.3-11.
(Heating distribution at a location 500mm apart from the center of the plume)
The maximum heating rate value of the landing gears of 52 (kW/m^2), described above is taken to be the heating rate for the landing gears.
(Utilization of the heating rate described above over the entire landing gear portions is on the safe side.)
Fig.5.2.3-12 shows the heating history of the airframe bottom assumed in the present analysis.

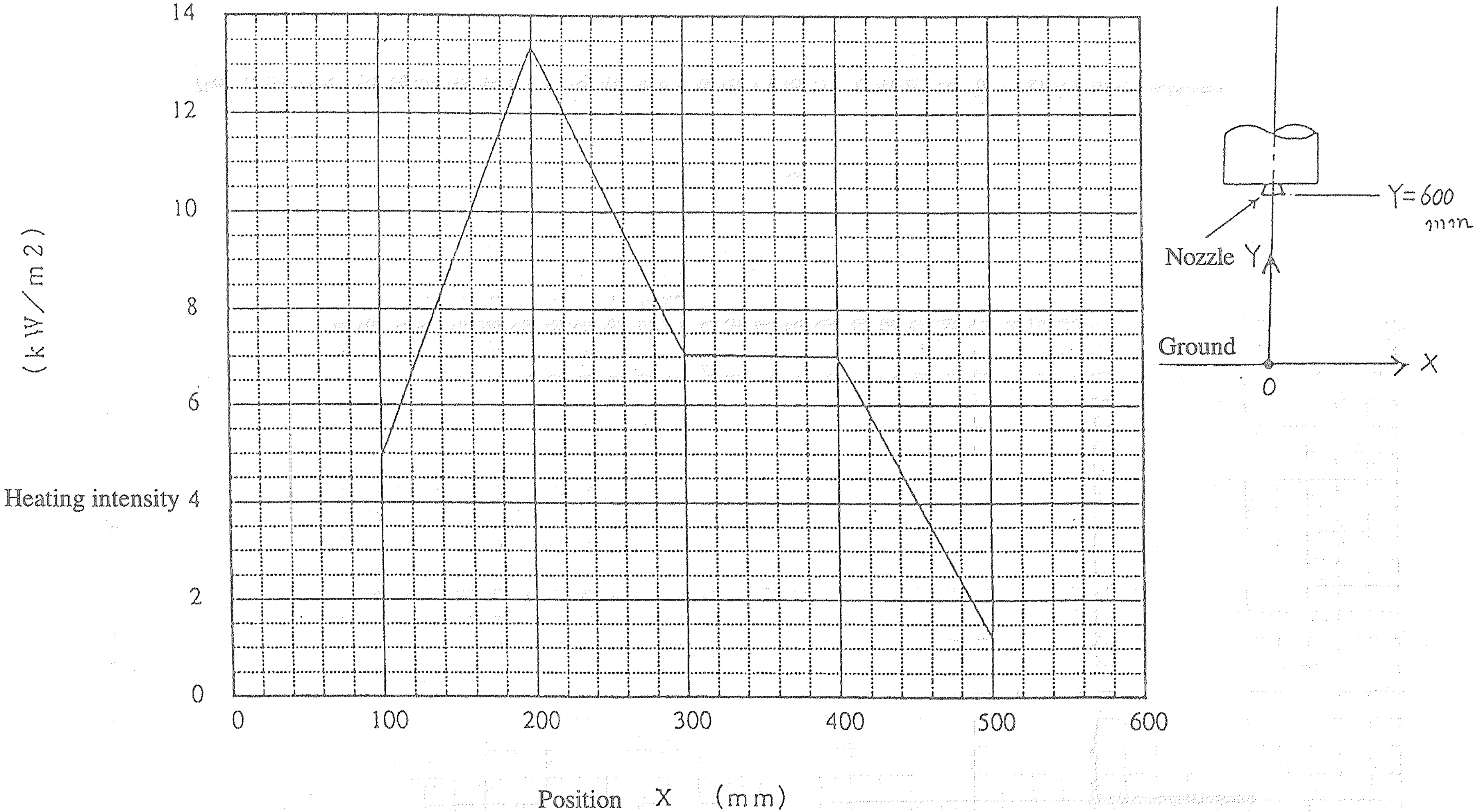
<2> Examination of the temperature analysis/thermal protection scheme

1) Temperature analysis/thermal protection of the airframe bottom

Using the heating intensity estimated in the preceding paragraph <1>, temperature analysis was conducted on the airframe . Fig.5.2.3-13 shows the temperature history of the airframe bottom Al skin (without a thermal protection material). From the figure, it is concluded that, with the AL skin thickness of 0.8mm (the thickness that is assumed in the structural design), the requirement (below 85°C) cannot be met without using a thermal protection material.

Next, Fig.5.2.3-14 shows the relation between the thickness of a thermal protection material and the maximum Al skin temperature, obtained through the temperature analysis assuming that the airframe bottom is furnished with the thermal protection material.
Following the result, the thickness of the inorganic fiber layer of the thermal protection material has been set at 3mm. (temporary value; see the figure to the right.)





(Note) Wall temperature=20 (° C)

Fig.5.2.3-9 Heating rate distribution at the height Y = 600mm (heating intensity distribution on the airframe bottom)

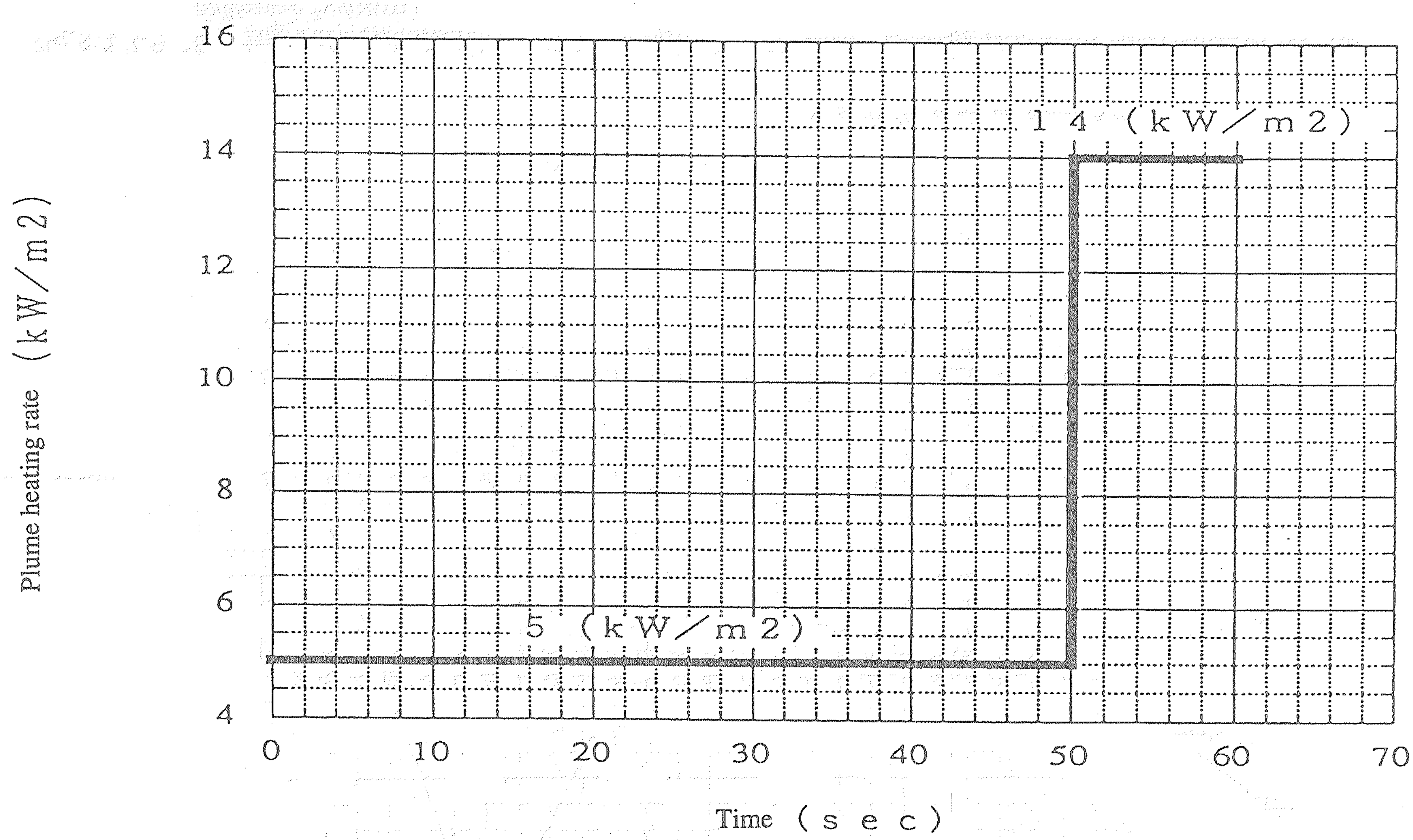
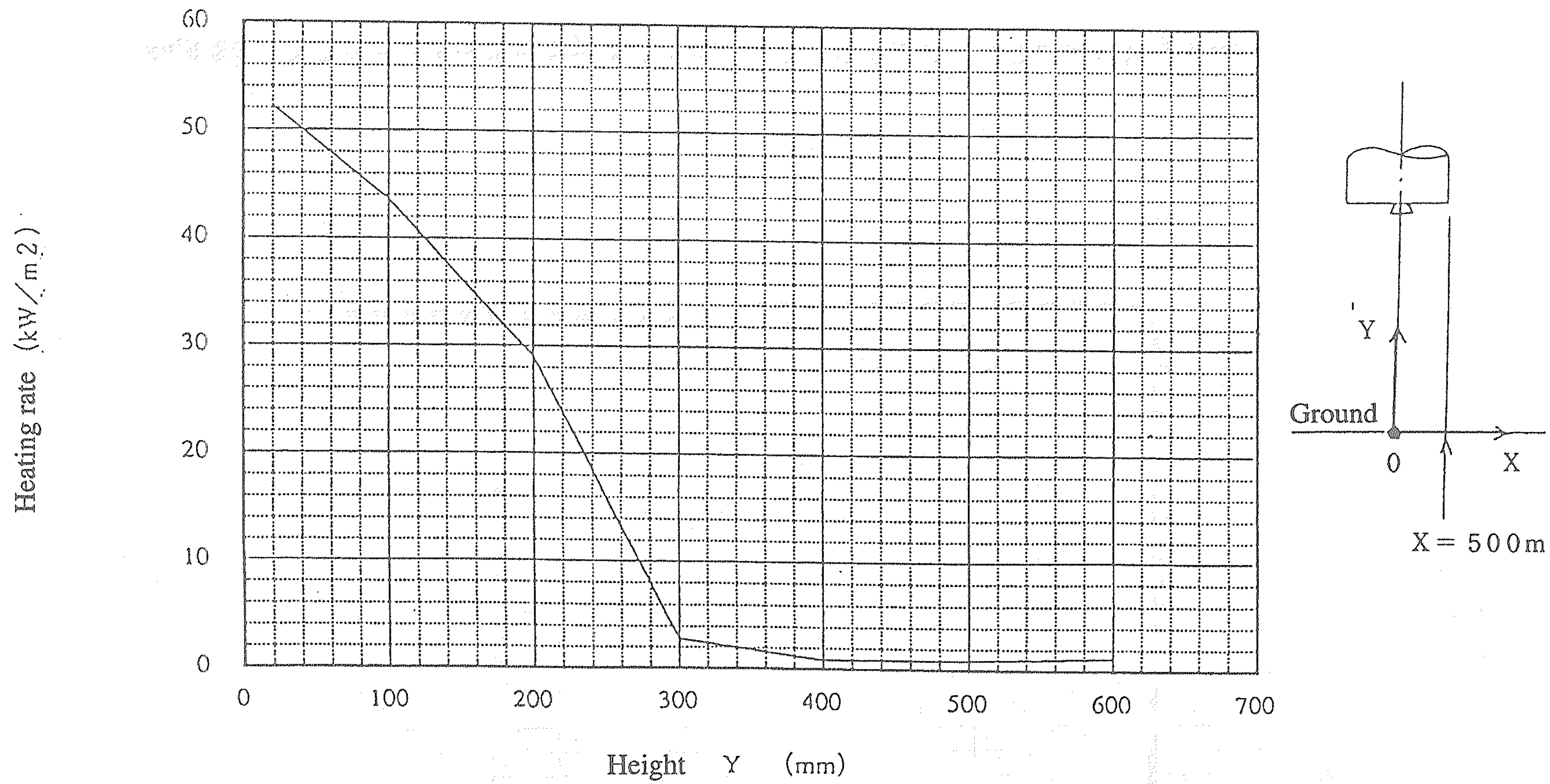


Fig.5.2.3-10 Short-term history of the plume heating rate on the airframe bottom



(Note) A wall temperature of 20 (° C) is supposed.

Fig.5.2.3-11 Heating rate distribution at the height X = 500mm (heating rate distribution on the landing gears)

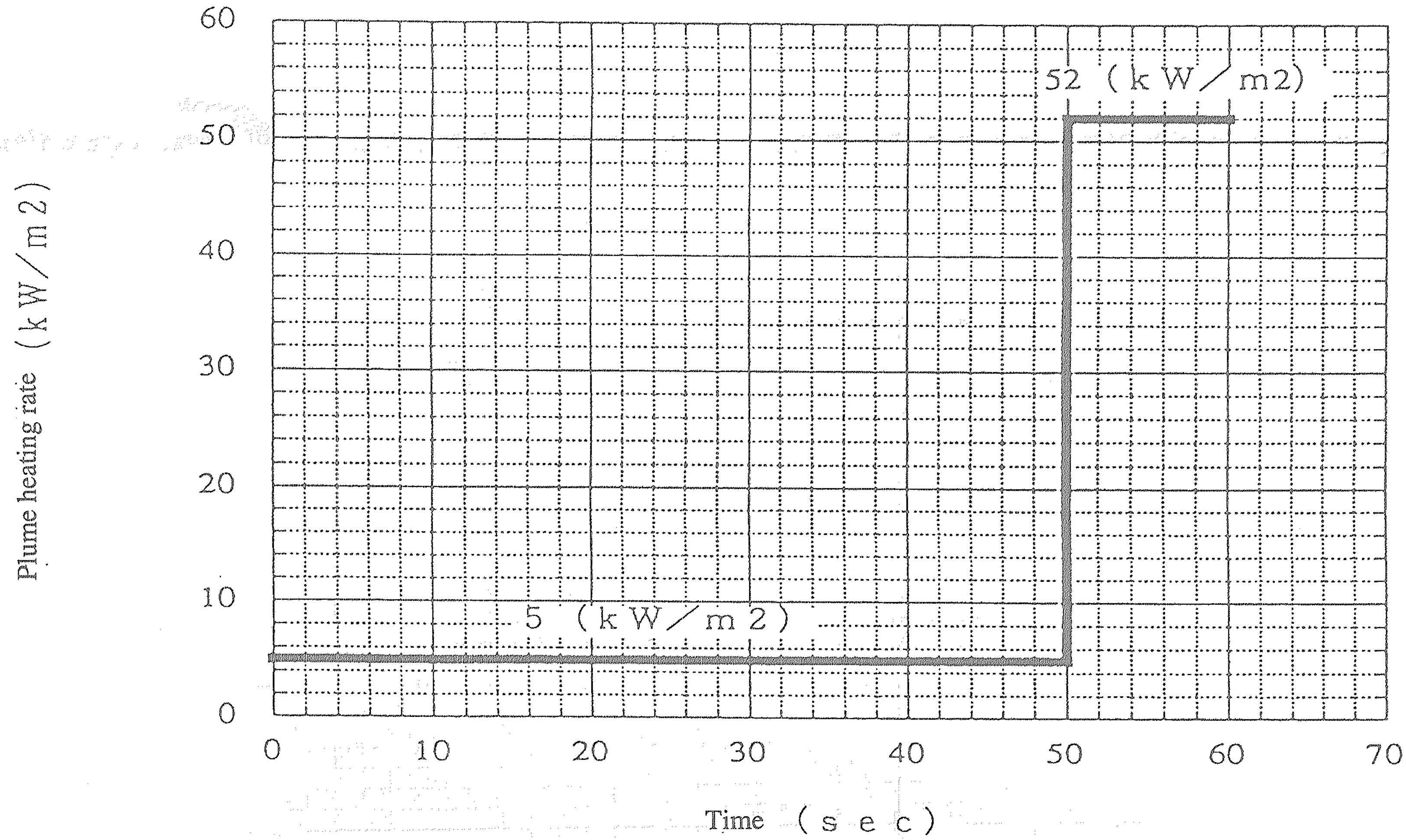


Fig.5.2.3-12 Short-term history of the plume heating rate on the landing gears

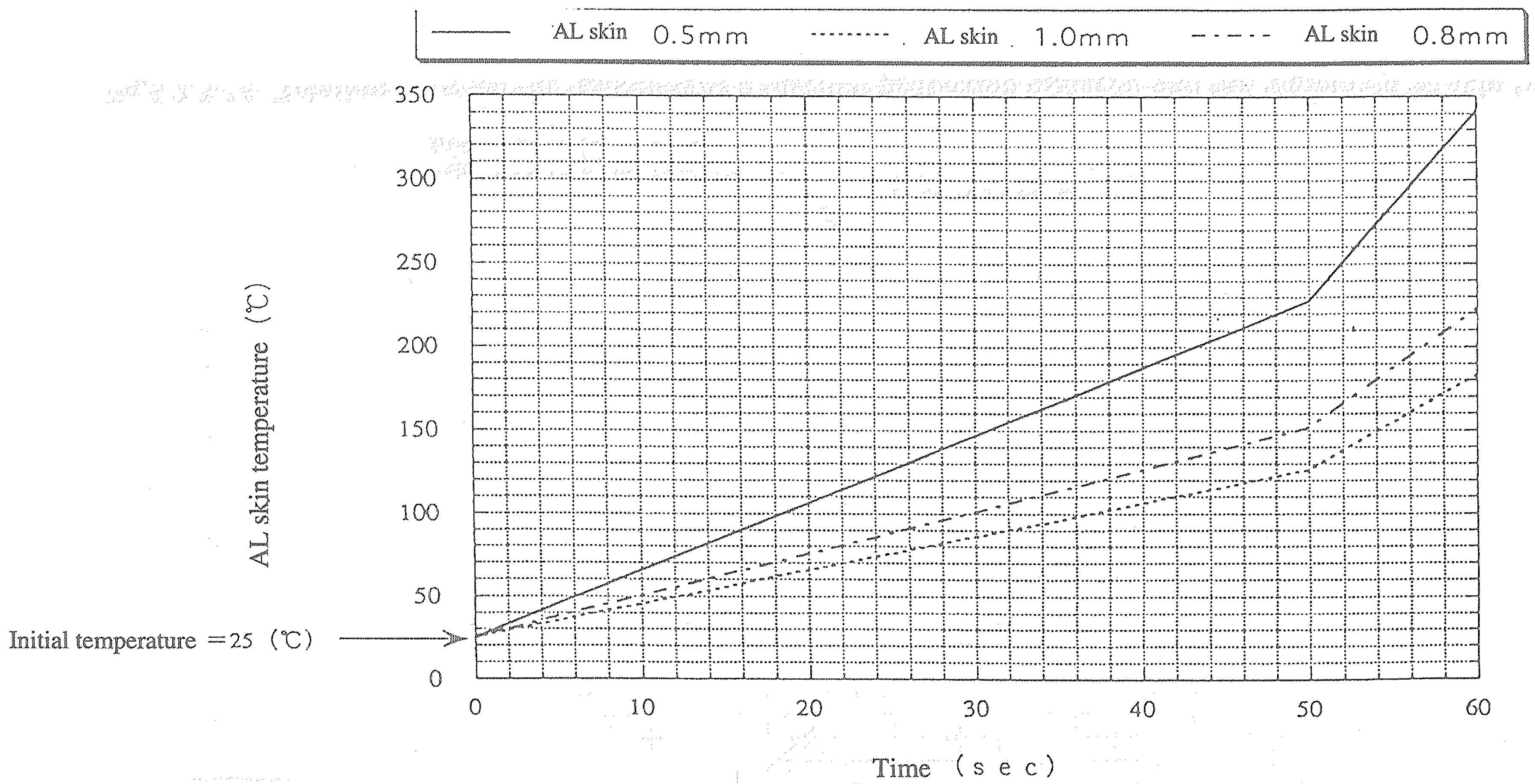


Fig.5.2.3-13 Temperature history of the airframe bottom Al skin (without a thermal protection material)

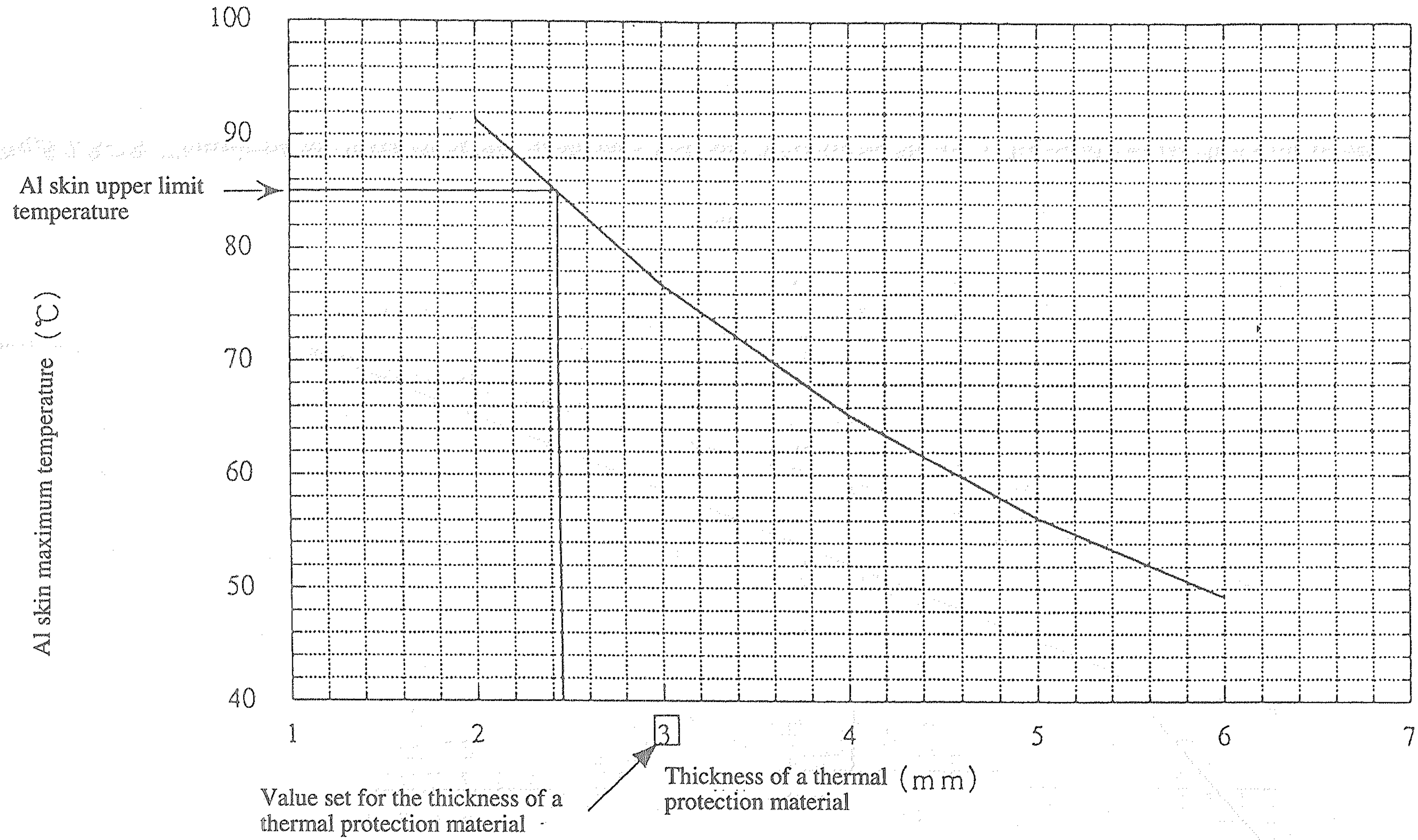


Fig.5.2.3-14 Relation between the thickness of a thermal protection material and the maximum Al skin temperature

Furthermore, Fig.5.2.3-15 shows the temperature history with a 3 mm thick thermal protection material.
Because it was predicted that the temperature of the tank outer wall near the AL skin on the airframe bottom would hardly rise throughout the period from takeoff to landing in this case, it was anticipated that installing a thermal insulation material on the tank outer wall would not be necessary.

Since, however, conduction of heat from the airframe bottom via the supporting structure to the tanks is likely to take place, the thermal control scheme to be applied to the tank mounting section will be examined when the design specifications for the support structure are decided on.

Fig.5.2.3-16 outlines the thermal analysis model assumed for the temperature analysis of the airframe bottom.
The initial temperature of different members was taken to be 25°C because the airframe bottom is not irradiated by the sunlight.

2) Temperature analysis/thermal protection for the landing gears

Temperature analysis was conducted on the landing gear using the heating intensity estimated in the preceding section <1>. Fig.5.2.3-17 outlines the specifications (temporary) of the landing gears assumed in the temperature analysis of the landing gears.
The landing gear consists of the following basic components:

a) Oil chamber (AL material with a thickness of 5mm)
b) Shaft (made of stainless steel, with a thickness of 1.5mm)
c) Spring (made of stainless steel, with a wire diameter of 1.5mm)

Among the components mentioned above, the oil chamber containing the hydraulic oil (temperature requirement: below 65°C) provides the high temperature criteria.

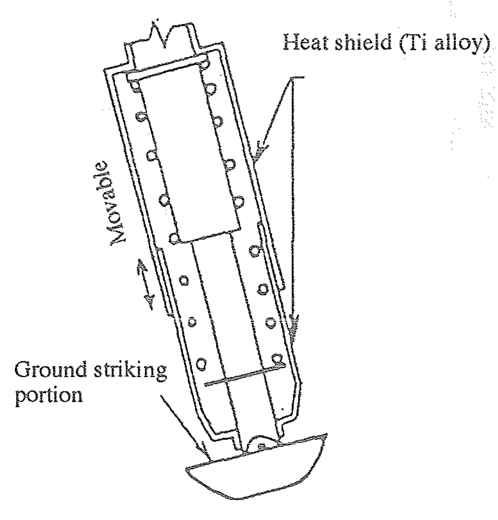
Fig.5.2.3-18 shows the temperature history of the landing gear components for the case in which the landing gears are not provided with a thermal protection material.

Examining the figure lead to the conclusion that, without a thermal protection material on the landing gears, the temperature requirements are not met in the oil chamber and shaft.

(no protection required for the spring)

A Ti alloy heat shield has been proposed for a thermal protection method for the oil chamber and shaft described above.

(refer to the figure to the right.)



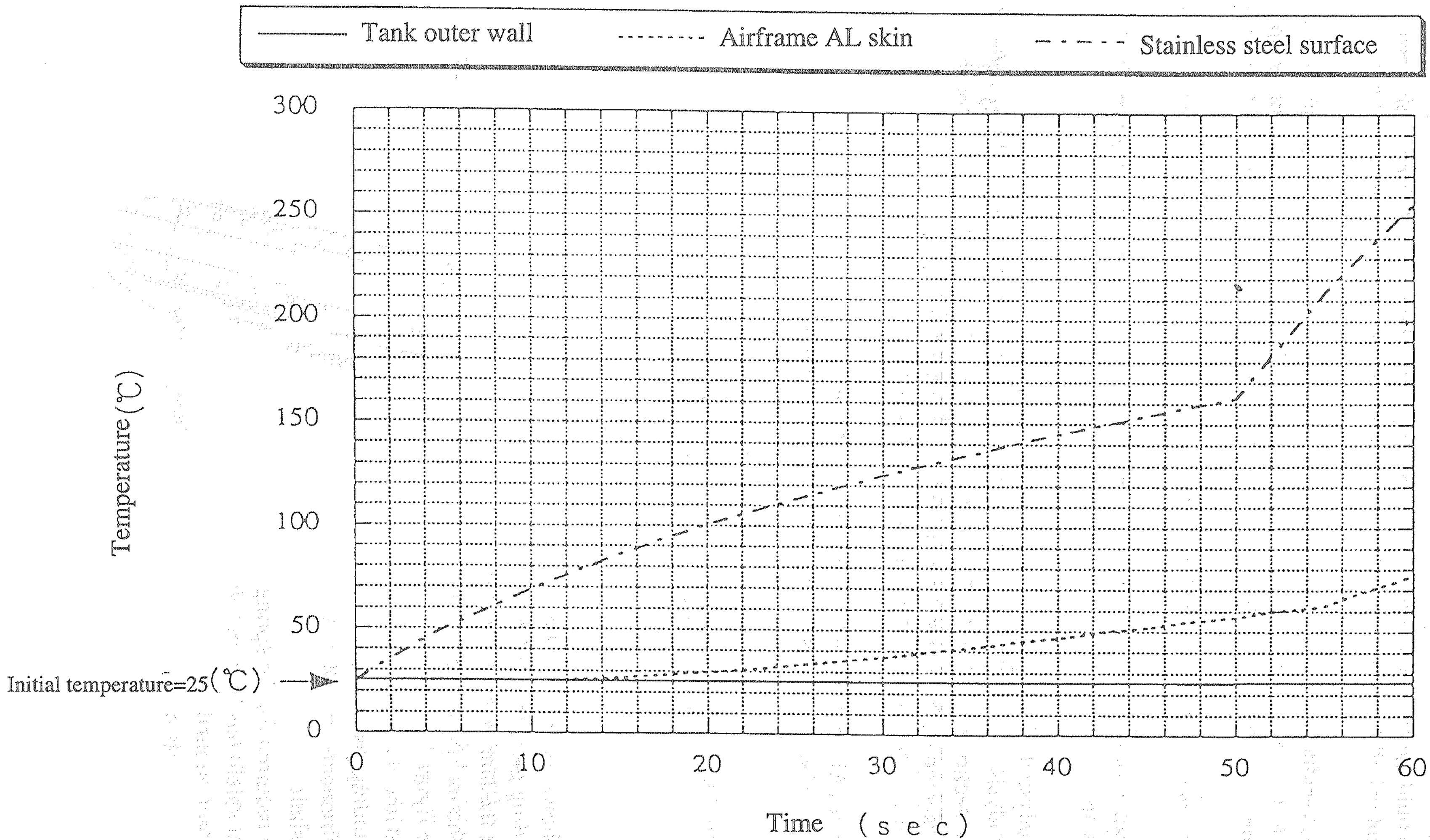


Fig.5.2.3-15 Temperature history with a 3 mm thick thermal protection material

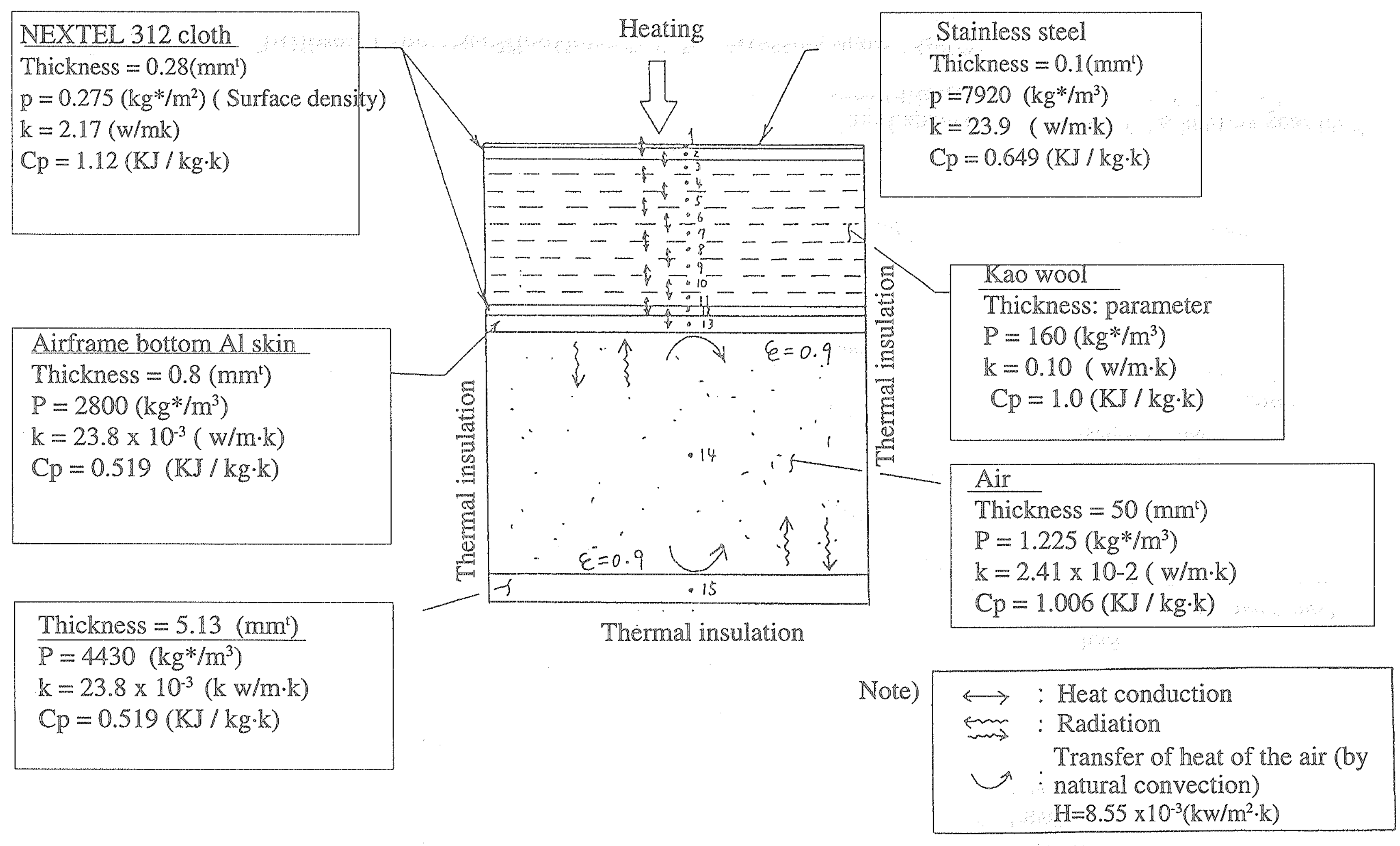


Fig.5.2.3-16 Thermal analysis model assumed for the temperature analysis of the airframe bottom

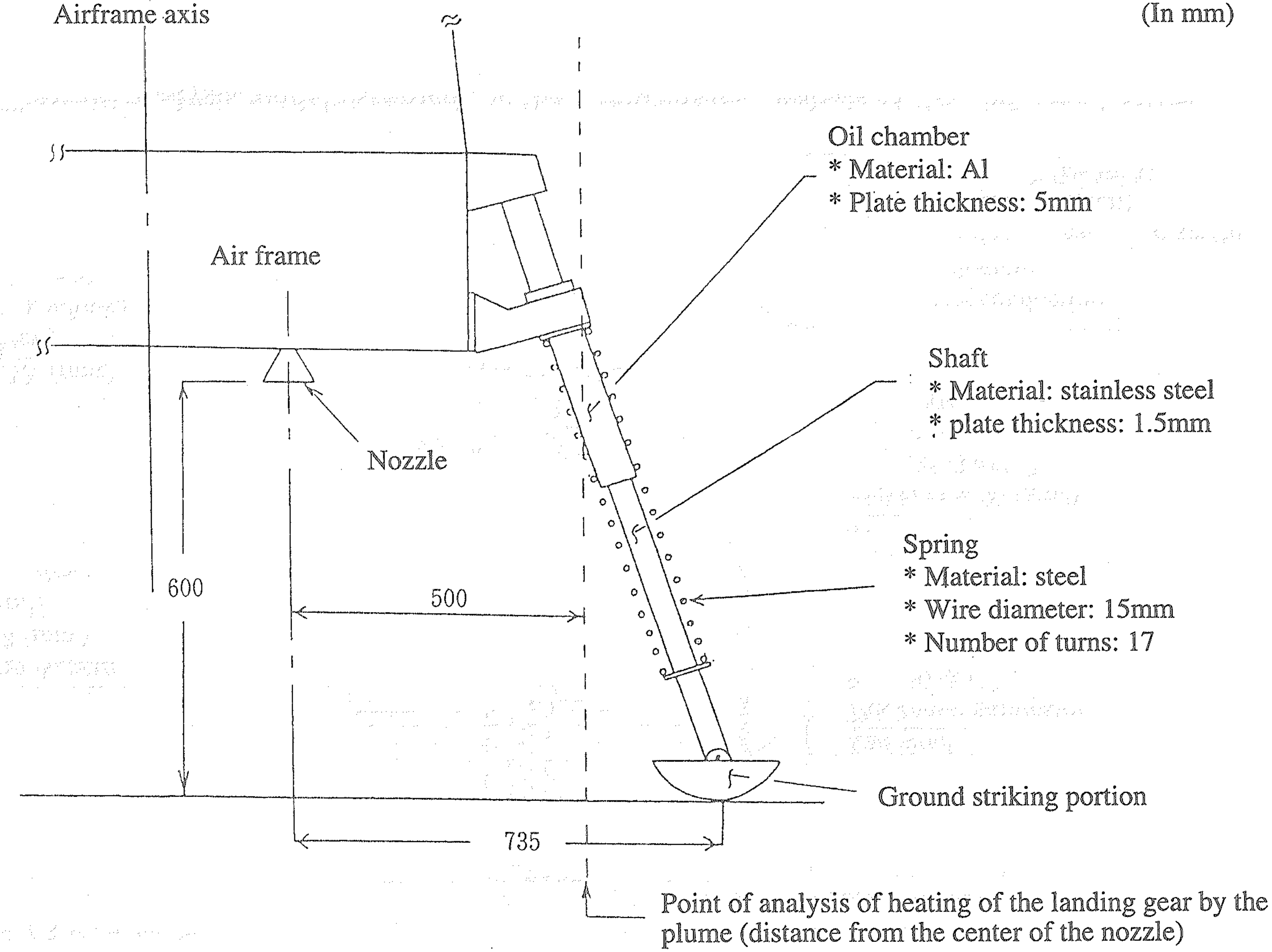


Fig.5.2.3-17 Outline of the specifications for the landing gear (draft)

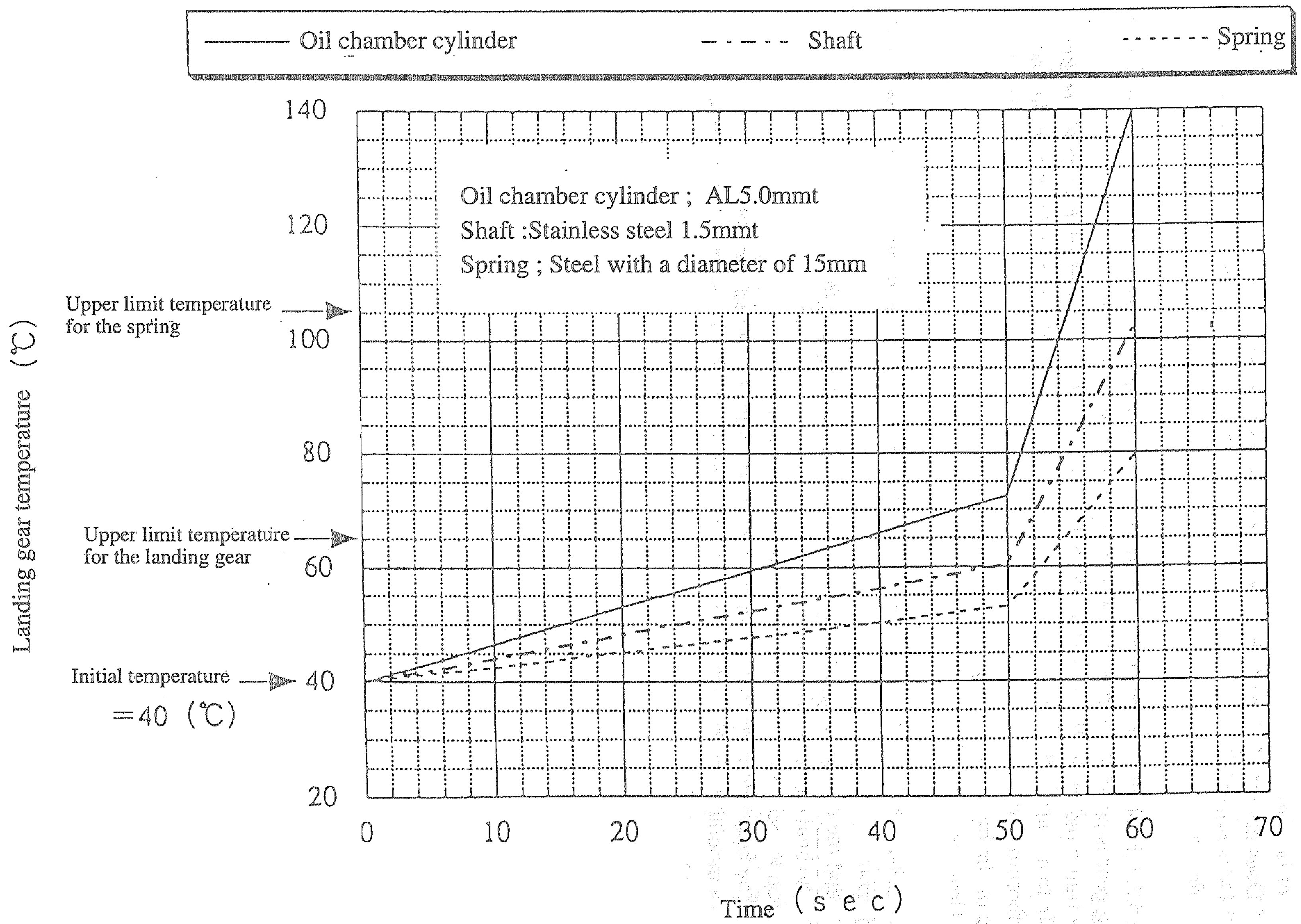


Fig.5.2.3-18 Temperature history of the landing gear components for the case in which the landing gears are not provided with a thermal protection material

However, the heat shield should meet the following requirements:

* Does not hinder the extension and compression actions of the landing gears at the time of landing;
* Does not interfere with the ground striking portion; and
* Does not exhibit interference, gripping, and the like due to vibration and thermal deformation.

Fig.5.2.3-19 shows the temperature history of the oil chamber and shaft in the case where the landing gear is provided with a Ti heat shield (plate thickness: 0.4mm).

The figure indicates that providing the landing gear with a 0.4mm thick heat shield makes it possible to maintain the temperature of the landing gear within the required temperature range.

Furthermore, the surface of the heat shield is coated with heat resistant paint to protect the surface of the heat shield at high temperatures.

(Heat resistant paint: white; heat resistant temperature: 600°C)

Fig.5.2.3-20 outlines the thermal analysis model assumed for the temperature analysis for the landing gear.

Note that the current analysis model is based on the assumption that:

* The heat capacity of the spring is not taken into consideration (on the safe side with respect to high temperatures); and
* The heat shield and the spring are thermally isolated from each other. (a mounting method which separates the heat shield and spring from each other is supposed.)

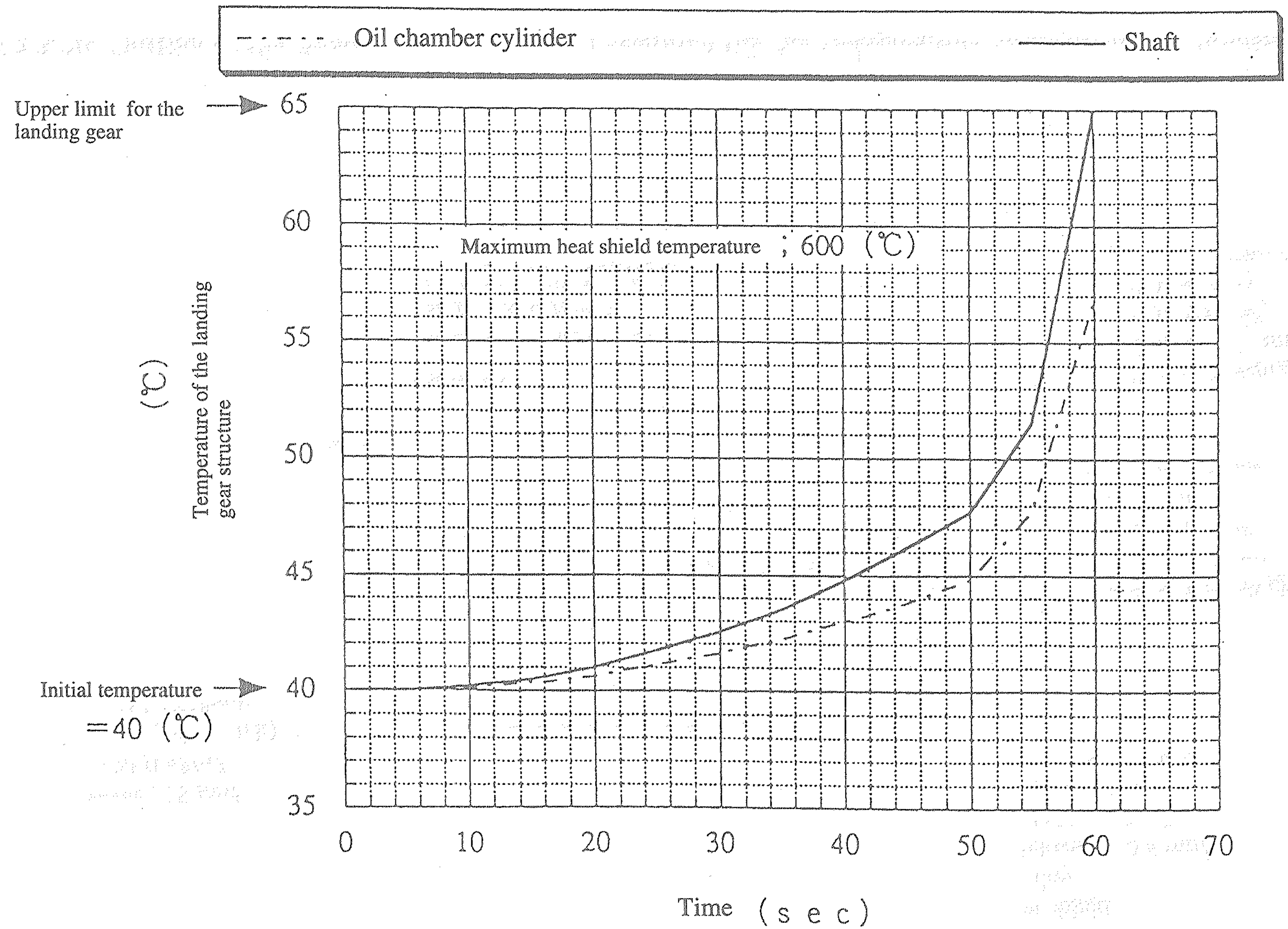


Fig.5.2.3-19 Temperature history of the landing gear provided with a Ti heat shield

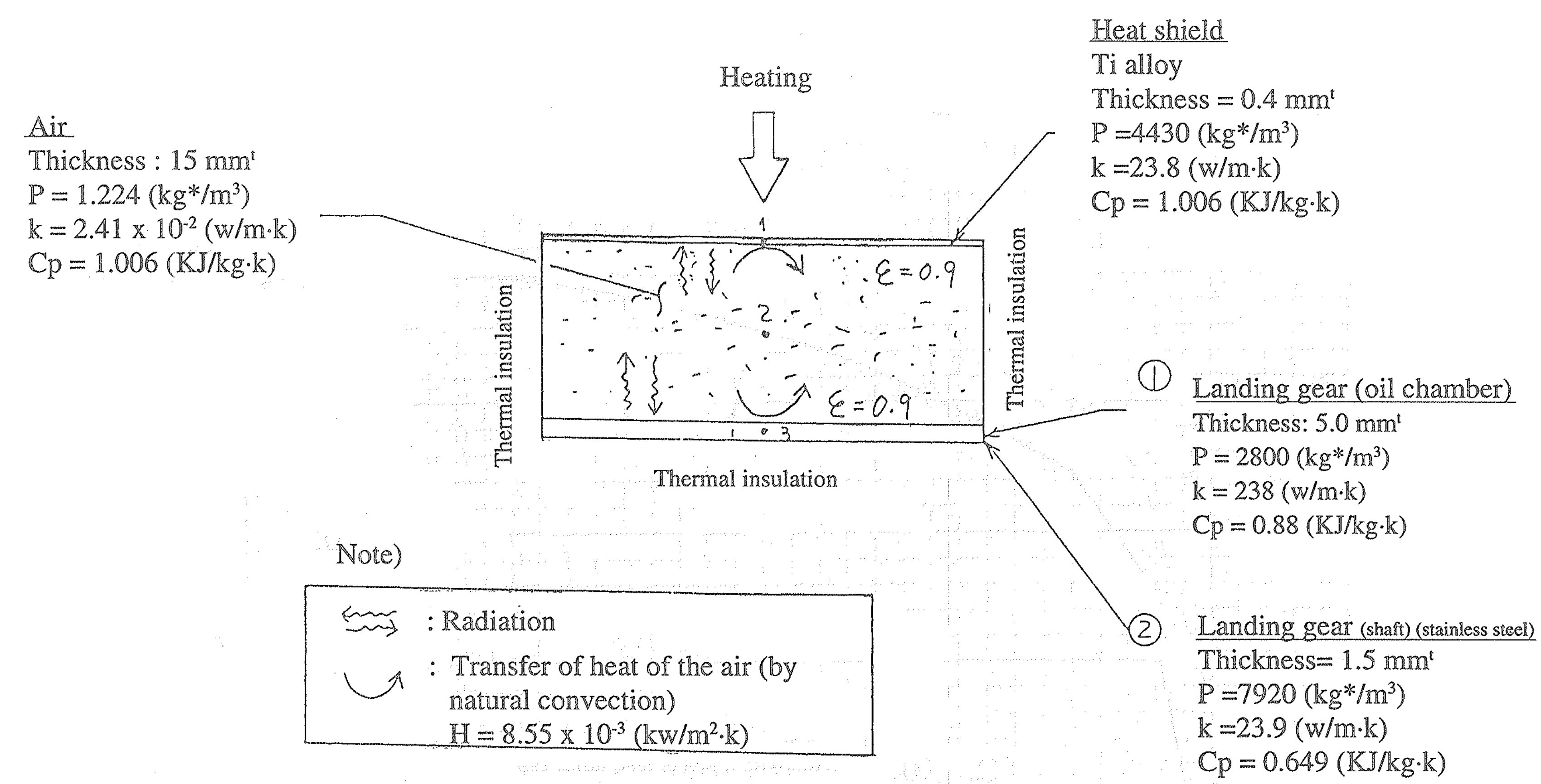


Fig.5.2.3-20 Outline of the thermal analysis model assumed for the temperature analysis for the landing gear.

3) Thermal control scheme for the onboard equipment

a) Thermal control scheme for the electronic equipment

The interior of the experimental vehicle is air-conditioned until just before takeoff and the time for the flight experiment is as short as about a minute; therefore, it is anticipated that the temperature rise due to heat dissipation by the electronic equipment (including the battery) will be limited to the allowable temperature range by its heat capacity.

Therefore, no specific thermal control will be necessary; however, if the temperature of the airframe structure rises, influx of heat from the equipment supporting structure through the conduction mechanism is possible. (in addition, heat influx through convection and radiation is possible.)

If the quantity of heat from these origins is large, it becomes necessary to insert thermal insulating spacers between the equipment and supporting structure.

These subjects will be examined when the heat generation profile and arrangements of equipment, and specifications for the supporting structure are established.

b) Thermal control scheme for the tanks

Although the tanks do not generate heat, the heat influx from the airframe structure is possible as in the case of the electronic equipment.

Particular attention should be directed to the NTO tank which is installed close to the airframe bottom. Since this arrangement allows heat to be conducted from the structure heated by the plume, via the supporting structure, to the tank, insertion of thermal insulating spacers may become necessary.

Examination on these subjects will be conducted when the specifications for the tank and supporting structure are established.

(5) Plan for Temperature Measurement

Data to be used for designing a reusable rocket is acquired through measuring the temperature and heat flux of the airframe and onboard equipment while the vehicle is air-conditioned on the ground, is in flight, and is at rest after landing.
The sensor output signal is sent to the ground station by telemetry via the signal conditioner in the communications system.

(a) Number of the points of measurement/items (draft)

<1> Temperature measurement

Airframe temperature measurement

* Outer surface of the thermal protection material on the airframe bottom:	5 points
* Outer plate on the airframe bottom:	5 points
* Inner plate on the airframe bottom:	5 points
* Oleo gear section of the landing gears:	4points
* Landing gear shaft:	4 points
* Landing gear heat shield:	4 points
Subtotal:	27 points

Onboard equipment temperature measurement

* Hydrazine tank:	2 points
* NTO tank outer wall:	2 points
* GHe tank outer wall:	2 points
* Electronic equipment:	3 points
Subtotal:	9 points
Total:	36 points

<2> Heat flux measurement

* Airframe bottom:	4 points
* Landing gears:	4 points
Total:	8 points

(b) Sensor

<1> Temperature sensor

A temperature measuring resistor requires a biasing voltage source; therefore, use of a thermocouple, which does not need a power supply for biasing, has been decided on.

<2> Heat flux sensor

A sensor is used which enables both radiation heat flux and convection heat flux to be measured.

5.3 Guidance and Control

Requirements for the guidance and control subsystem, which form the basis for the task "Examination of the Guidance and Control Technologies for Reusable Spacecraft" to be executed under a separate contract are summarized in Table 5.3-1.

Details of the subsystem blocks and others are examined in the following Chapter 5.4.

Table 5.3-1 Summary of the requirements for the guidance and control system (1/3)

1. Development Policies
(1) Items to compose the guidance and control equipment are basically furnished by NASDA; as the development environment, the existing NASDA facilities (or the manufacturer's equivalent facilities) are used as a rule; (2) By using the existing facilities mentioned above, the development of software and its verification are efficiently conducted , contributing to minimizing the development cost; and (3) By using a high-level language (C language, for example) for software, the development work is facilitated.

Table 5.3-1 Summary of the requirements for the guidance and control system (2/3)

II. Software system configuration

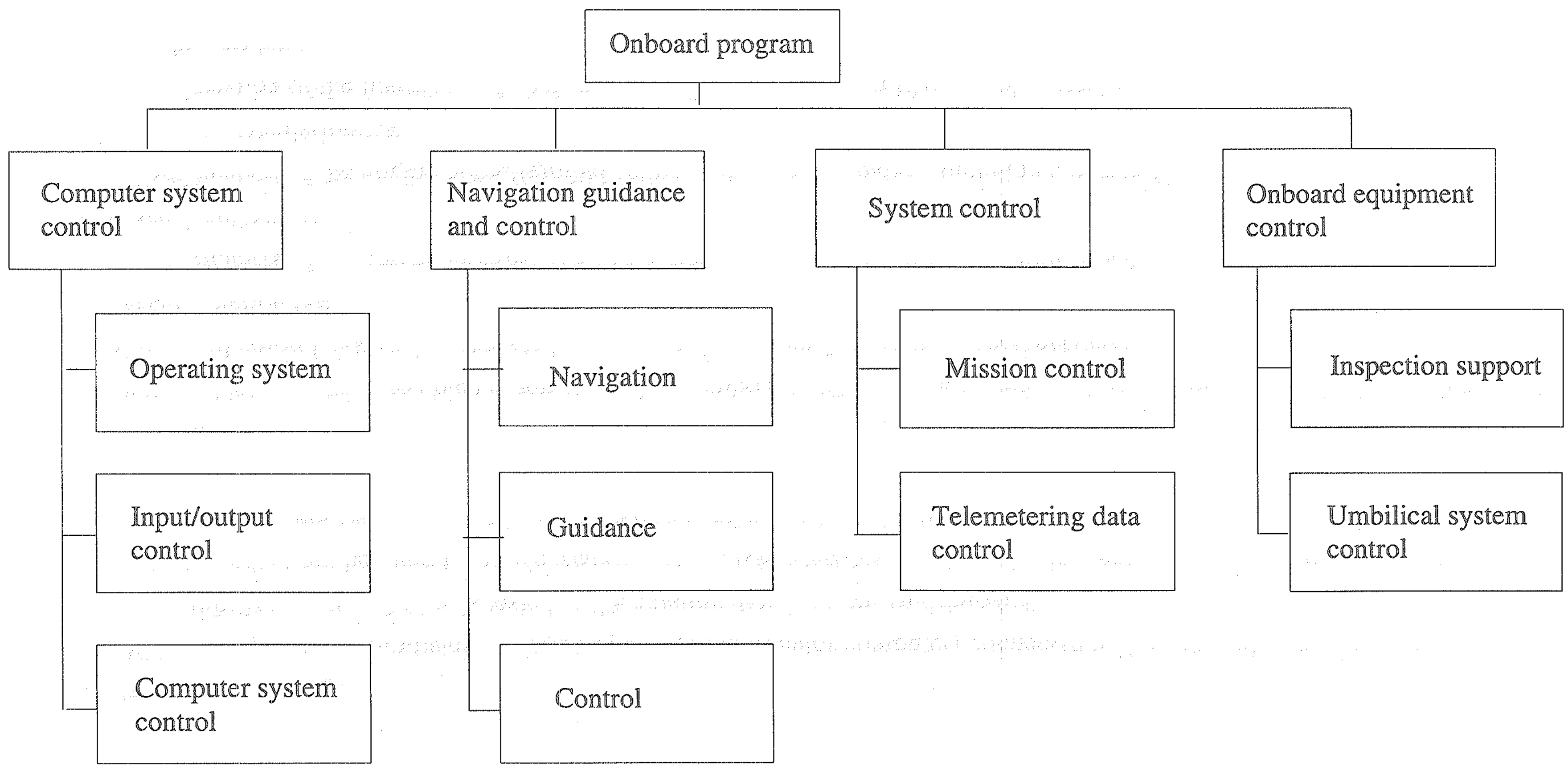


Table 5.3-1 Summary of the requirements for the guidance and control system (3/3)

III. Plan for Testing
(1) Verification of design a. With a computer that simulates the input and output (simulator computer) connected to the onboard computer, the compatibility of the software is tested in the existing development environment. b. Before conducting the overall vehicle system test, a system test for which the basic pieces of the navigation guidance and control equipment are connected is conducted to confirm and verify the design ultimately. (2) System test With the vehicle wholly assembled complete with the onboard software, the closed loop simulation test for the navigation guidance and control system is conducted to confirm the system functions comprehensively. <> Sensor function test * Confirmation of the computer operation and output under sensor input simulating signals. <> Engine function test * Confirmation of the engine output/gimbal action under the computer simulation command. <> Closed loop simulation test * Confirmation of the function and performance of the navigation guidance and control. * Flight simulation.

5.4 Electric System

Described below are the development policies and required specifications for the electric system of the vertical takeoff and landing experimental vehicle, together with the results of the examination of the vehicle design:

(1) Development policies

In order to simplify the system and minimize the weight and cost to the extent that it can meet the required specifications, the following policies should be followed:

<1> To divert the pieces of equipment developed for ALFLEX and other, and avoid developing new equipment;

<2> To have NASDA, as extensively as possible, furnish the pieces of equipment to be diverted from other applications;

<3> To substitute common consumer products (commercially available products) for pieces of equipment that must be otherwise newly developed;

<4> In order to alleviate the burden on pieces of equipment for which their resistance to the range environment is not verified (such as items under study by NASDA and consumer products), shock mounts or other similar methods will be used to mount such pieces to the vehicle, with the exception of inertial measuring units of which precision of mounting alignment is required.

<5> Except equipment and systems for the emergency stop, the equipment/ system are not redundantly configured.

(2) Required specifications

Table 5.4-1 shows the required specifications for the electric system.

(3) Examination of the design

The design specifications and other related conditions for the configuration, functions, and components of the electric system, required to meet the above required specifications, are described below.

(a) Equipment configuration

Fig.5.4-1 shows the equipment configuration block diagram, Fig.5.4-2 the functional block diagram, and Table 5.4-2 the dimensions and other data of the onboard equipment. In the following, details of the individual systems are given.

(b) Navigation guidance and control system

* Is equipped with an inertial measuring unit (IMU) and a radio altimeter (RA) as sensors for the navigation guidance and control system.
* Arithmetic operations for the navigation guidance and control are processed by the onboard computer, which incorporates the onboard software shown in Fig.5.4-3.
* For the sequence control for the flight experiment, a landing gear position detecting switch unit is installed to detect takeoff and landing.
* In order to provide an interface between the thrust magnitude controller, which is the controller for the propulsion system, and the onboard computer, the vehicle is equipped with a CIU interface.
* For analyzing experimental data, sensor data from the navigation guidance and control system and status data are sent to the ground facility via the communications system.

Table 5.4-1 Required specifications for the electric system

Item	Details of the requirements
1. Functional requirement	
<1> Navigation guidance and control function	* Is provided with a function for measuring the acceleration and angular acceleration signals required to control the orbit and attitude; * Is provided with a function for controlling the engine thrust and gimbal actuator; and * Allows more than one flight pattern to be selected and is capable of controlling the pattern selected.
<2> Telemetry command function	* Is capable of measuring the in-flight experiment data and transferring it to the ground facilities. The data is to be recorded by the ground facilities, not by the onboard equipment; and * Receives commands from the ground and executes pieces of processing in accordance with the commands received.
<3> Electric power supply function	* Supplies electric power to the onboard equipment during the flight; and * Is capable of being supplied with electric power by the ground facilities for maintaining and checking the onboard equipment before the in-flight experiment.
2. Serviceability requirement	* Takes into consideration the serviceability to enable the in-flight experiment to be repeated in short time intervals.
3. Safety requirement	* Takes dangerous conditions arising from failures during the flight into consideration and is provided with sufficient safety measures.
4. Electric power requirement	Nominal: not more than 1200 W Maximum: not more than 2000 W
5. Weight requirement	Not more than 90 kg

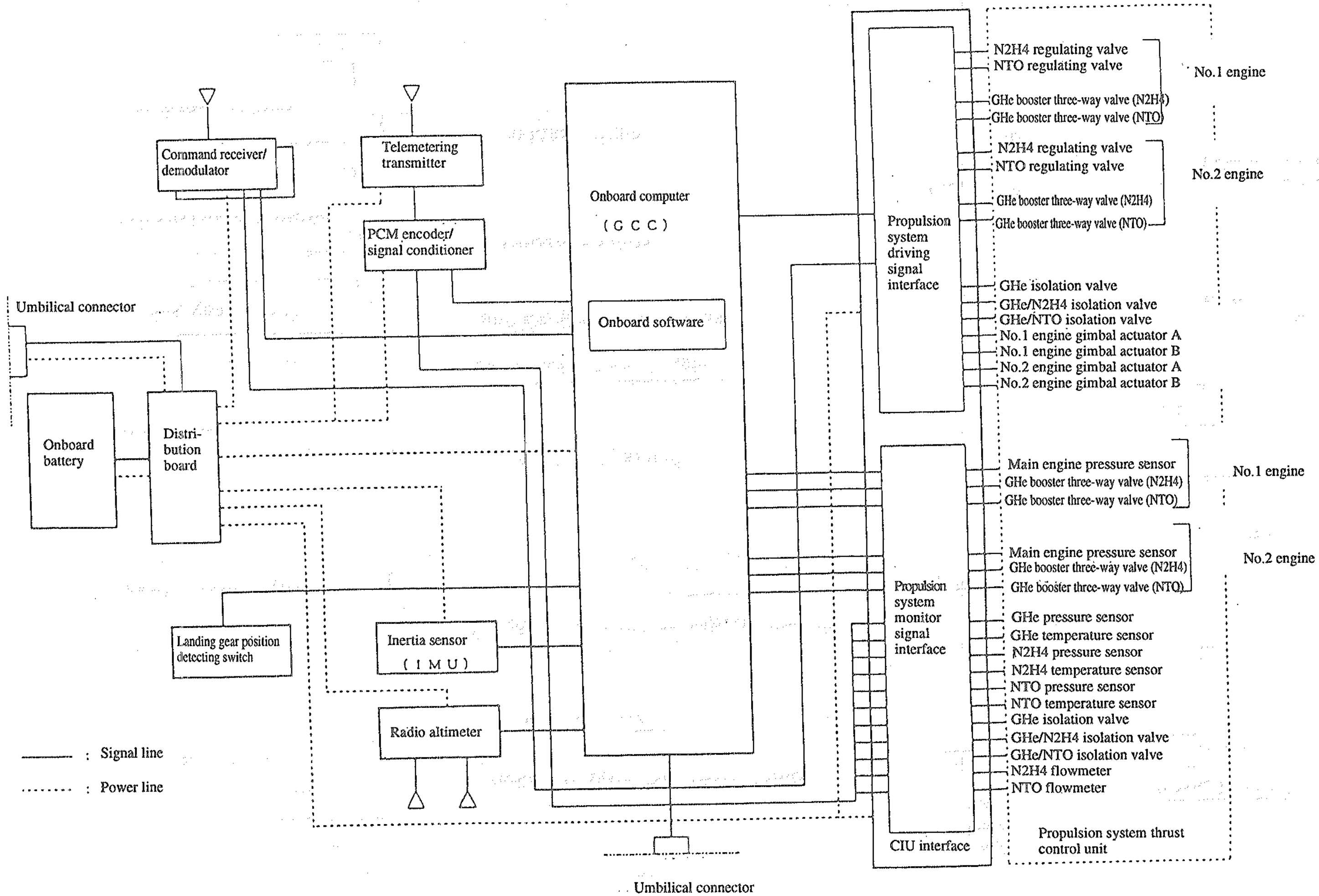


Fig.5.4-1 Configurational block diagram for the experimental vehicle onboard equipment

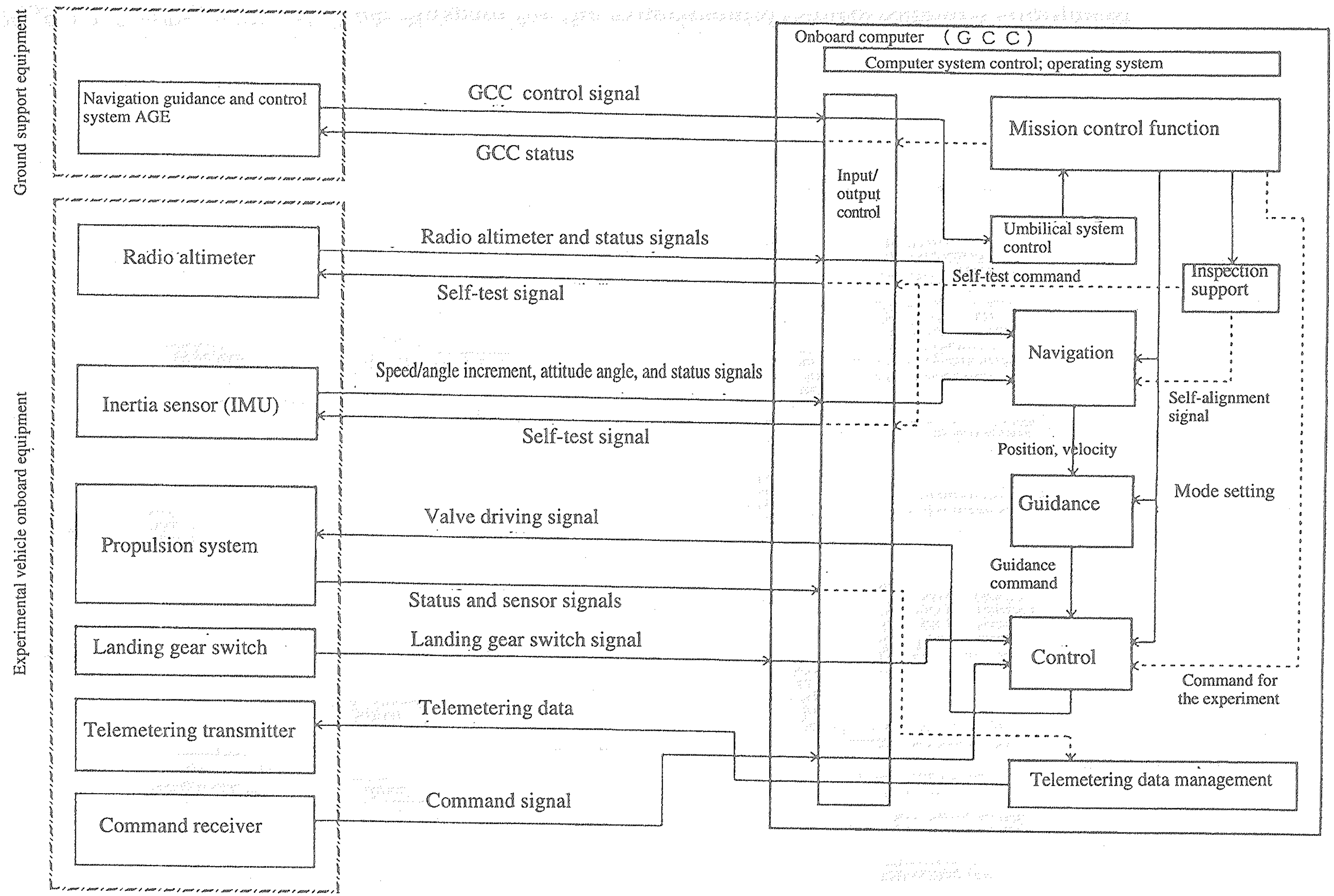


Fig.5.4-2 Functional block diagram for the experimental vehicle onboard systems

Table 5.4-2 Experimental vehicle onboard electric and electronic equipment

* Per unit

System	Unit designation	Quantity	Power consumption (w) *		Weight (kg)	Outer dimensions (W x H x Dmm)	Remarks
			Nominal	Peak			
Propulsion system	High-pressure sensor	1	8	8	(Included in the propulsion system)	(Included in the propulsion system)	
	High-pressure isolation valve	1	0	47			Latch valve
	Booster isolation valve	2	0	9			Latch valve
Propellant feed system	Low-pressure pressure sensor	2	8	8			
	Flowmeter	2	5	5			
	Regulating valve	4	80	160			
Engine system	Three-way valve	4	37	37			
Gimbal system	Actuator	4	78	156			
	CIU	1	100	100			
	Heater						Necessity of a heater under study
Navigation guidance and control system	Inertia sensor (IMU)	1	55	55	10	236 x 188 x 232	
	Computer (GCC)	1	60	60	18	368 x 159 x 512	
	Radio altimeter (RA)	1	25	25	1.4	109 x 84 x 129	
	RA antenna	1	-	-	0.06	91 x 28 x 91	
	CIU interface	1	10	10	5	150 x 200 x 100	Wire harnesses and others
	Others	1	-	-	3	-	
Communications system	Telemetering transmitter	1	30	30	5	150 x 120 x 190	
	PCM encoder	1	35	35	8	330 x 120 x 190	
	Command receiver	1	20	20	3.5	155 x 141 x 165	Including a demodulator
	Telemetering antenna	1	-	-	0.1	ϕ 17 x 250	The quantity TBD
	Command receiver antenna	1	-	-	0.05	ϕ 17 x 250	The quantity TBD
	Others	1	-	-	3	-	Wire harnesses and others
Power supply system	Battery	1	-	-	6.1	220 x 100 x 210	
	Distribution board	1	-	-	7	440 x 140 x 330	
	Others	1	-	-	3	-	Wire harnesses and others

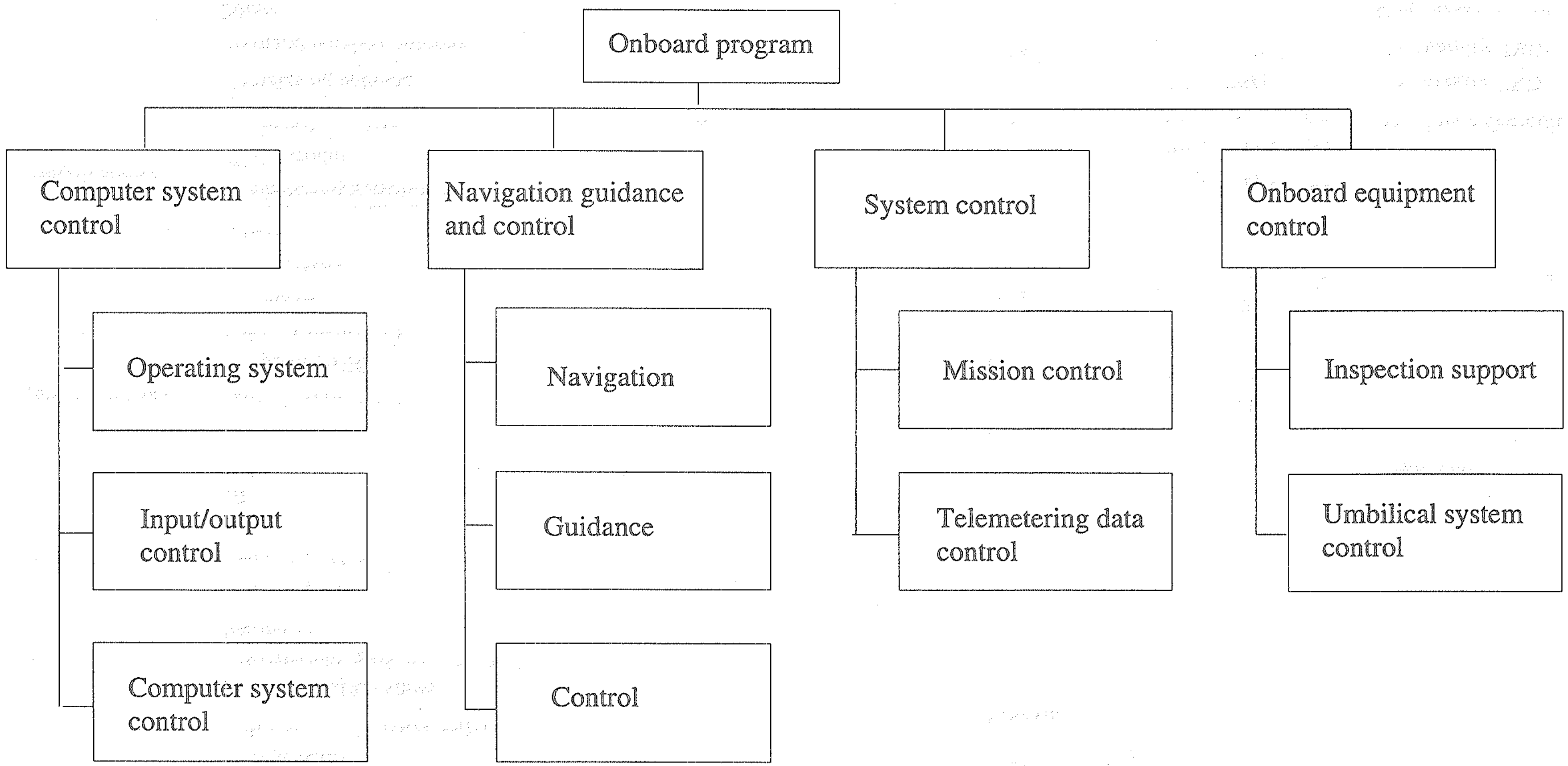


Fig.5.4-3 Hierarchical diagram for the basic functions of the onboard software

(c) Communications system

* Is equipped with a telemetering transmitter and an encoder for transmitting onboard sensor signals to the ground facility.
* Is equipped with a command receiver to receive commands from the ground facility; as few commands as possible are to be used and only emergency stop signals are used.

(d) Power supply system

* Fig.5.4-4 shows the configuration of the power supply system.
* Each piece of onboard equipment is supplied with electric power by the onboard battery during the flight experiment.
* While on the ground for inspection and adjustment, the onboard equipment is supplied with electric power by the external power supply.
* For the reason of reusability, a rechargeable NiCd battery is used as the onboard battery.
* Is equipped with a distribution board to change over between the external and the internal power supply, distribute electric power to different pieces of equipment, and charge the onboard battery.
* Fig.5.4-5 shows the electric power profile, required energy, and battery capacity.

(e) Umbilical system

* Umbilical lines are installed to connect the external power supply and check the functions of the onboard computer.
* The umbilical lines are disconnected by the quick disconnect type connectors immediately before starting the flight experiment. The connectors are disconnected by pulling the lanyard with a linear actuator installed on the service tower.

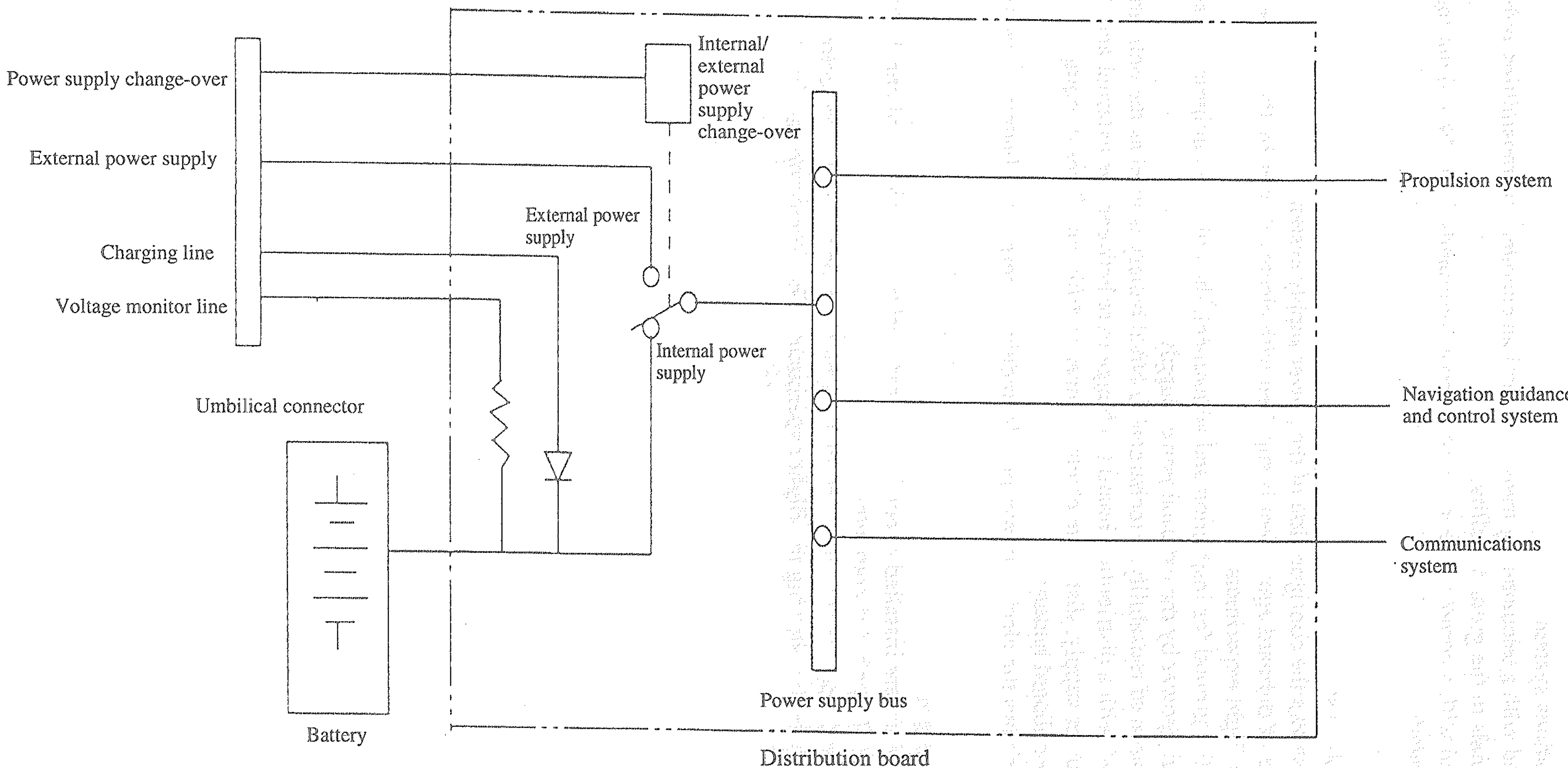
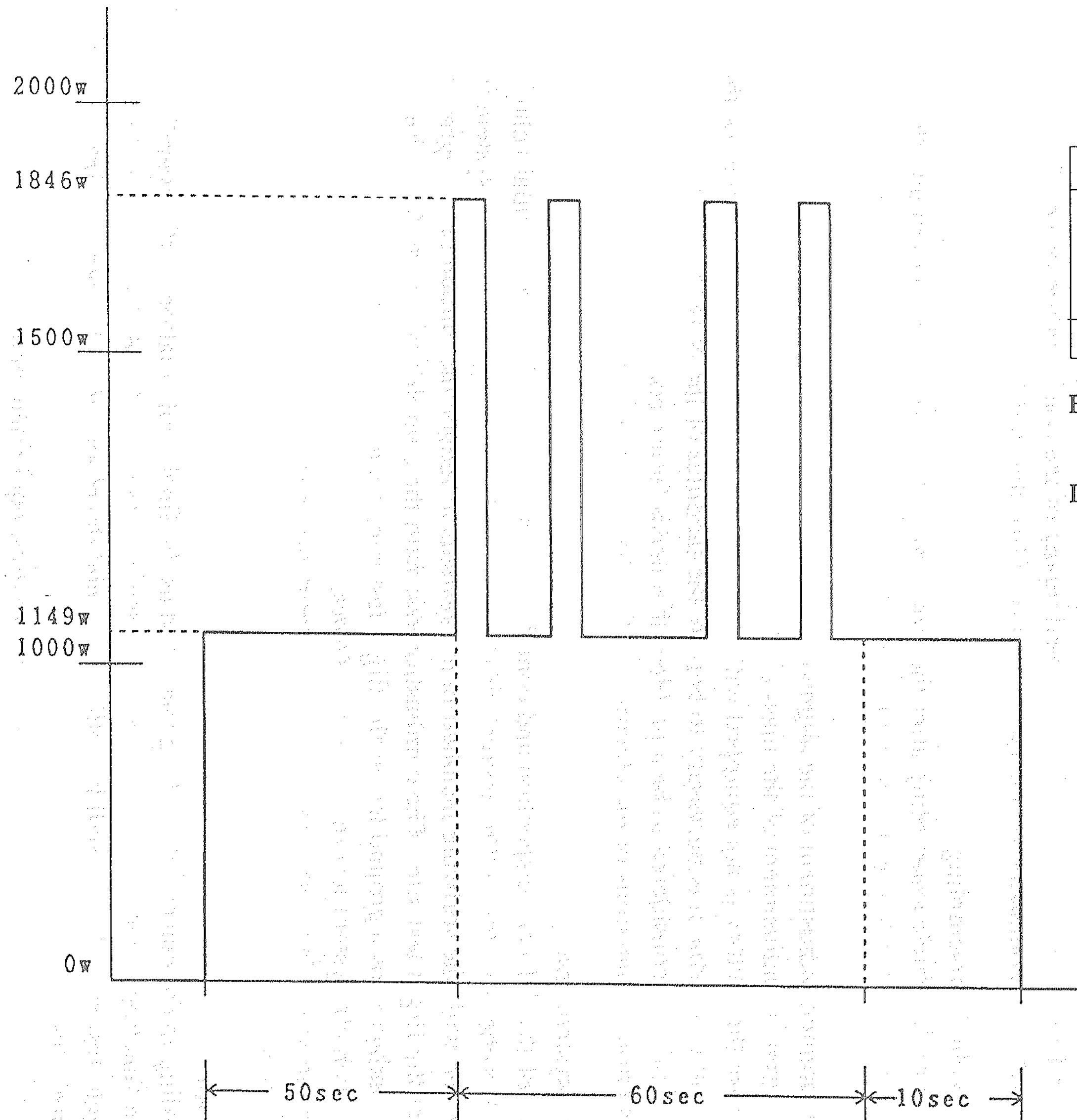


Fig.5.4-4 Power supply system



	Nominal power	Peak power
Propulsion system	914w	1611w
Navigation guidance and control system	150w	150w
Communications system	85w	85w
Total	1149w	1846w

Electric energy consumption=(nominal power+(peak power-nominal power)xduty)x"Power On" time

Duty: Ratio of the time during which the valves in the propulsion system are ON to the time during which the power is ON.

Duty	Electric energy consumption
At 50%	49.9 wh
At 25%	44.1 wh
At 10%	40.6 wh

49.9WH = 28V(Cell voltage 1.2V*24) x 1.78Ah

Therefore, NiCd battery cells N-4000DR (4Ah) are adopted; 27 cells are used in series connection to ensure a voltage margin.

Fig.5.4-5 Power consumption

5.5 Vehicle Assembling

In the course of the system integration and examination of the operations plan, the assembling/operation plan, including the vehicle assembling processes, overall vehicle system test, and free flight test, was discussed in outline as part of the system design of the vertical takeoff and landing experimental vehicle. Through this discussion, the items for the system test, details of ground support system, and those of the test at the flight test site were made clear. The scale of the flight test planned here extends to the wide area free flight which is the ultimate goal of the flight experiment.

(1) Flight experiment work breakdown structure (WBS)

The work breakdown structure for the design/fabrication of the experimental vehicle, fabrication of the ground support facilities, examination of the operations plan is shown in Fig.5.5-1.

(2) Assembling/testing plan

On the basis of the experiment concept work breakdown structure, the flow of the major processes of assembling/testing were defined as shown in Fig.5.5-2. The assembling, inspection, and testing of the experimental vehicle are limited to a minimum of work at a factory such as the confirmation of the basic functions, verification of the software/onboard equipment interface, and measurement and adjustment of airframe physical properties; at the same time, efforts are put into avoiding the overlapping of the confirmation or verification activities with tests conducted on the equipment or subsystem level.

a) Overall vehicle assembling

In the overall vehicle assembling after the outfitting of the propulsion system and the mounting of the landing gear and equipment, the following activities are performed at the factory:

* Measurement/adjustment of the alignment
* Measurement/adjustment of the mass characteristics
  Because the factory is not equipped with a test unit for measurement/adjustment of the moment of inertia, it is necessary to borrow the facilities of the NASDA Tsukuba Space Center; this is considered to be a problem that needs future coordination.
* Inspection of functions of the electric and the propulsion system

b) Overall system test

As part of the delivery inspection and completion inspection of the experimental vehicle, the comprehensive test of the navigation guidance and control and the electric system is conducted with the software installed in the systems to reduce the burden of retrogressive work at the flight test site. The combustion test with the completely assembled vehicle is to be conducted as a ground test at the flight test site, not at the factory or the like. The overall vehicle system test items are as follows:

* Navigation guidance and control system simulation test, and
* Electric system integrated test.

c) Flight test

According to the current plan, the ground test as the final confirmation of the overall system functions (combustion test for the propulsion test and the like) will be conducted at the flight test site; the test will be followed by and developed into the captive flight test and free flight test.

Table 5.5-1 summarizes the outline of the activities, equipment and jigs used, and site of execution for the major activities listed in the assembling/testing flow.

- 000 Takeoff and landing flight experiment
  - 100 Project management
  - 200 Development and design
    - 210 Structure system (including the thermal control)
    - 220 Propulsion system
    - 230 Avionics
    - 280 Integration
    - 290 Operations plan
  - 300 Fabrication of the experimental vehicle
    - 310 Structure system (including the thermal control)
      - 311 Overall structure
      - 312 Landing gear
      - 313 Thrust support structure
      - 314 Skin structure
      - 315 Flame protection system
      - 362 Others
    - 320 Propulsion system
      - 321 Engine system
      - 322 Thrust control system
      - 323 Gimbal system
      - 324 Tank system (including pipe line and valves)
      - 325 Pressuring system
      - 326 Auxiliary propulsion system (gas jet)
      - 327 Others (controller system CIU)
    - 330 Avionics
      - 331 Onboard software
      - 332 Navigation sensor (inertial measuring unit,RA) *
      - 333 Onboard computer *
      - 334 Control interface unit (CIU)
      - 335 Others
      - 341 Telemetering system *
      - 342 Command system
      - 343 Antenna system
      - 344 Ground equipment (the same as that listed under the classification number 420 through 450) *
      - 345 others
      - 351 Onboard battery
      - 352 Distribution board
      - 353 Others
    - 380 Overall vehicle assembling
      - 381 Propulsion system outfitting
      - 382 Airframe outfitting
      - 383 Overall vehicle assembling and adjusting
      - 384 Overall vehicle system test
      - 385 Others
  - 400 Fabrication of the ground facilities
    - 410 Range system
    - 420 Control system *
    - 430 Experiment data processing system *
    - 440 Ground communications system *
    - 450 Ground measurement system *
    - 460 Arrangement of the flight test site
  - 500 Flight test
    - 510 Vehicle transportation
    - 520 Ground test
    - 530 Flight maintenance activities
    - 540 Flight test
    - 550 Others

Note) The items marked with an asterisk are to be furnished by NASDA.

Fig.5.5-1 Takeoff and landing flight experiment work breakdown structure

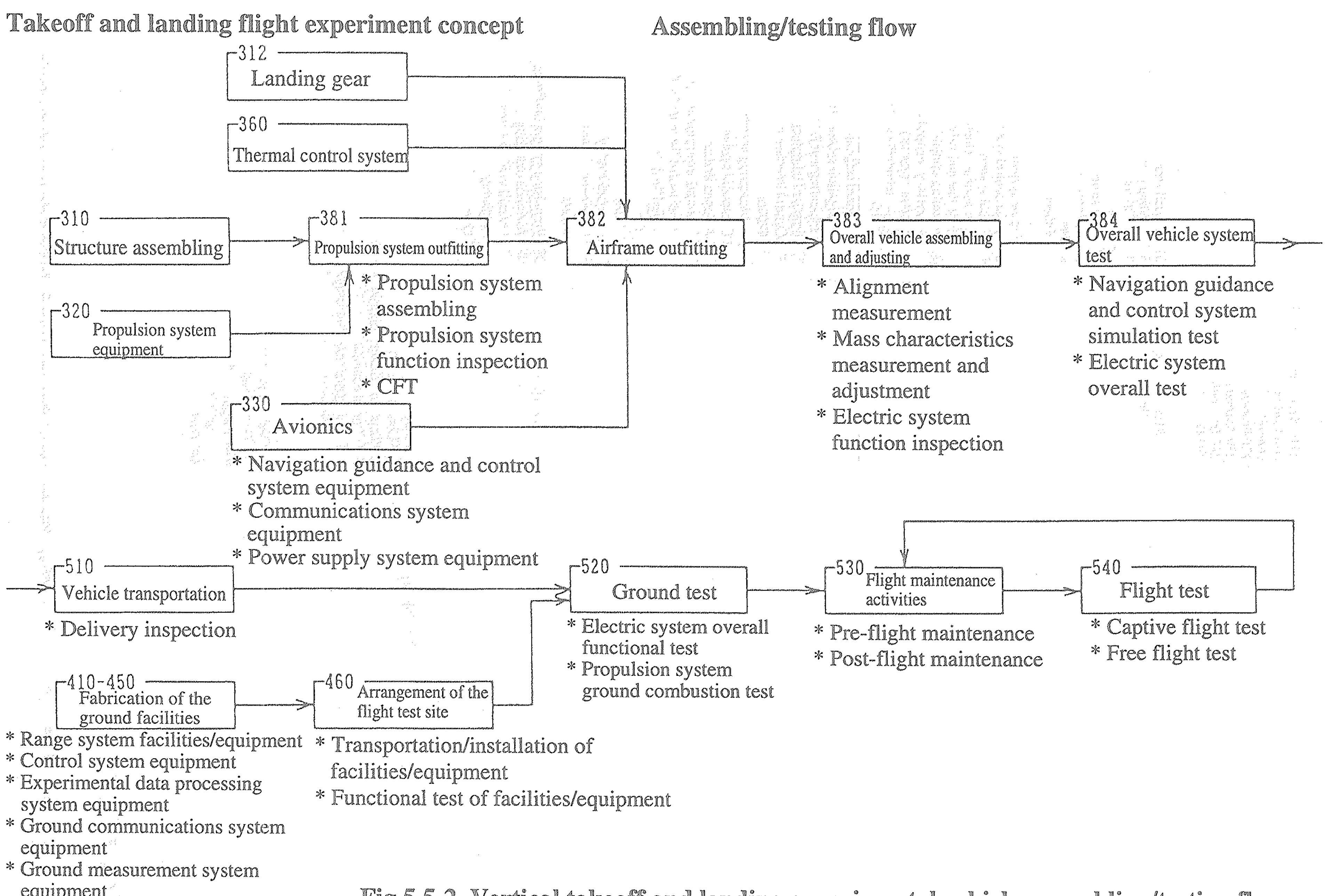


Fig.5.5-2 Vertical takeoff and landing experimental vehicle assembling/testing flow

Table 5.5-1 Outline of assembling/testing (1/5)

Outline of averall vehicle assembling/system test/flight test [Division 4] Integrate and operations (1/5)

WBS Item	Outline of activities	Equipment, facilities, jigs, and the like	Place of execution	Time of execution	Remarks
380 Overall vehicle assembling					
381 Propulsion system outfitting	The propulsion system equipment (including pipe line) is mounted to the airframe assembly, the functions are inspected, and CFT is conducted. (1) Assembling of the propulsion system * Mounting of the engine system, propulsion control system, gimbal system, pressurizing system andothers (2) Inspection of the propulsion system functions * System airtightness leak check (by a snoop solution or a pressure decay method) * Component function check (valves, sensors, and actuators) (3) CFT (captive flight test) * With the structure of the propulsion system assembled	Test equipment for the propulsion system (existing/diverted equipment) High-pressure test facilities (existing)		Fiscal 1996/ latter half	
382 Airframe outfitting	Functional components (including wiring) are mounted to the airframe assembly. * Equipment of the navigation guidance and control system, communications system, and power supply system * Landing gears, equipment/components of the thermal control system	―――		Fiscal 1996/ latter half	
383 Overall vehicle assembling and adjusting	With the entire airframe assembled, the physical characteristics are measured and adjusted and the electric system functions are inspected; these activities conclude the fabrication of the airframe. (1) Alignment measurement * Setting of the reference axis * Alignment measurement for IMU and RA * Alignment measurement and adjustment for the main engine and gimbal systems * Alignment measurement and adjustment of the propellant tank and landing gears (2) Measurement and adjustment of the mass characteristics (dry) * Measurement of weight * Measurement and adjustment of the position of the center of gravity (in the three directions) * Measurement and alignment of the moment of inertia (T.B.D.... no equipment available)	Alignment measuring unit (existing) Small load cell (existing)		Fiscal 1996/ latter half	

Table 5.5-1 Outline of assembling/testing (2/5)

Outline of averall vehicle assembling/system test/flight test [Division 4] Integrate and operations (2/5)

WBS Item	Outline of activities	Equipment, facilities, jigs, and the like	Place of execution	Time of execution	Remarks
383 Overall vehicle assembling and adjusting (continued)	(3) Electric system function inspection * Checking of the functions of the onboard computer and input/output signals * Checking of the functions of the onboard equipment (IMU,RA, and others) * Checking of the function of the power supply system * Checking of the functions of the external umbilical lines	GCC checkout unit (to be newly fabricated) Various inspection units/ equipment (existing)			
384 Overall vehicle system test	With the vehicle wholly assembled complete with the onboard software, the closed loop simulation test for the navigation guidance and control system and functional test for the communications and the electric power supply system are conducted to confirm the systems functions. (1) Navigation guidance and control system simulation test a) Sensor function test * Confirmation of sensor input simulating signals/computer output signals. b) Engine function test * Confirmation of computer simulated command/engine side output signals c) Closed loop simulation test * Confirmation of the function and performance of the navigation guidance and control system d) Gimbal system function test * Confirmation of static operations of the gimbal actuator (2) Electric system overall test * Confirmation of functions of all the systems by means of the flight sequence simulation	GCC checkout unit Propulsion system checkout unit Navigation guidance and control system AGE (to be newly fabricated)		Fiscal 1996/ latter half	

Table 5.5-1 Outline of assembling/testing (3/5)

Outline of averall vehicle assembling/system test/flight test [Division 4] Integrate and operations (3/5)

WBS Item	Outline of activities	Equipment, facilities, jigs, and the like	Place of execution	Time of execution	Remarks
400 Fabrication of the ground facilities					
410 Range system	With respect to the following facilities/equipment required to operate the experimental vehicle launch site (to execute the flight test) and their interfaces with the control system equipment (connecting harnesses), the specifications are defined and the procurement (fabrication/purchase) is executed: (1) Range facilities (2) Range inspection and maintenance equipment (3) Safety/ environmental preservation related equipment (5) Others	—		Fiscal 1995/ latter half through the fiscal 1996/ latter half	
420 Control system 430 Experiment data processing system 440 Ground communications system 450 Ground measurement system	The specifications, functions, performance, and interface of the NASDA furnished items are confirmed, and the design integration for various ground-base systems is conducted.	—		Fiscal 1995/ latter half through the fiscal 1996/ latter half	
460 Arrangement of the flight test site	The ground support facilities are installed, adjusted, and subjected to function checking to conclude the preparation for the range operations for the experimental vehicle. (1) Transportation/installation of the facilities/equipment * Transportation of facilities/equipment * Site construction work for the facilities/equipment, including installation work (including the mounting and adjusting of the pipe line/ equipment) (2) Inspection of functions of the facilities/equipment * Inspection of functions of the facilities/equipment associated with the electric system propellant, and others.	— Various inspection units/ equipment (existing)	Flight test site	Fiscal 1996/ latter half through the fiscal 1997/ former half	The place of execution is a remote site home.

Table 5.5-1 Outline of assembling/testing (4/5)

Outline of averall vehicle assembling/system test/flight test [Division 4] Integrate and operations (4/5)

WBS Item	Outline of activities	Equipment, facilities, jigs, and the like	Place of execution	Time of execution	Remarks
500 Flight test 510 Vehicle transportation	After the delivery inspection, the entire airframe is properly packed and transported to the flight test site by a trailer, a ship, or other means of transportation.	Transportation jigs (existing)	———	Fiscal 1997/ former half	The site of execution is a remote site home.
520 Ground test	The functions/performance of the airframe systems and their interface with the ground facilities are finally confirmed before the flight test is executed. (1) Electric system overall functional test * Confirmation of the functions/performance of the navigation guidance and control system * Confirmation of the communications and electric power supply systems * Confirmation of the electric interface between the experimental vehicle/ground facilities * Confirmation of the functions of the air conditioning systems for the experimental vehicle/ground facilities (2)Propulsion system ground combustion test * Sequence check (without propellant) * Propellant volume check test * Gimbaling check test (both with/without the propellant) * Flight simulation test (confirmation of the compatibility with the propulsion system)	Facilities/equipment of the range system Ground equipment such as the control system and others Navigation guidance and control system AGE Various inspection units/ equipment (existing)	Flight test site	Fiscal 1997/ former half	Procurement of the propellant/ pressurizing gas included
530 Flight maintenance activities	Maintenance of the airframe before and after the flight test and provision of the safety measures and environmental preservation (1) Pre-flight maintenance * Pre-flight check * Loading/pressurizing of the propellant (including the leak checking and other inspection) (2) Post-flight check * Post-flight check * Propulsion system contamination preventing unit (leak checking, pressure reduction, and propellant discharge)	Facilities/equipment of the range system Ground equipment such as the control system and others	Flight test site	Fiscal 1997/ former half	

Table 5.5-1 Outline of assembling/testing (5/5)

Outline of averall vehicle assembling/system test/flight test   [Division 4] Integrate and operations (5/5)

WBS Item	Outline of activities	Equipment, facilities, jigs, and the like	Place of execution	Time of execution	Remarks
540 Flight test	The captive flight test and free flight test are conducted: in the captive flight test, part of the flight freedom is subjected to restrictions; in the free flight test, the capability of withstanding multiple flights is demonstrated in the course of escalating the flight pattern from simple to complex ones. (1) Captive flight test * TN0 Flight with limited flight altitude/attitude Takeoff and landing with the vehicle skimming the ground Airframe bound by a rope or the like to prevent it from being tilted beyond the overturning angle. Confirmation of the performance of the propulsion system and navigation guidance and control system (2) Free flight test * TN1 - TN2 Vertical jump (50m - 100m) Thrust control, attitude stability, soft landing, flame protection, and so forth * TN3 - TN4 Hovering horizontal movement (one-way, round trip) Hovering stability,dynamic stability, guidance and control, and the like * TN5 - TN6 Drop braking (low velocity, high velocity) Deep throttling, engine rapid acceleration and high velocity braking * TN7 - TN8 Parabolic drop braking (low velocity, high velocity) Simultaneous horizontal/vertical braking, guidance and control, high velocity braking, and the like	Facilities/equipment of the range system Ground equipment of the control system and others	Flight test site	Fiscal 1997/ former half The testing period is estimated at two months.	Procurement of the propellant/ pressurizing gas included

5.6 Flight Test Concept

With respect to the experimental vehicle system operations, the items and details of the test to be conducted at the flight test site, safety measure for abnormal flights, and effects of disturbances on the flight range were examined in outline.

The scale of the test considered in the examination is one which enables the "Wide Area Free Flight," the ultimate goal of the present flight experiment, to be conducted; the experiment, therefore, supposes a domestic remote site (Hokkaido area or similar area) which permits a vast expanse of a test security zone to be secured.

(1) Flight test plan

Table 5.6-1 shows the draft concept for the flight test including the ground test at the flight test site.

The test items are as follows:

* Ground test (electric system overall functional test, propulsion system ground combustion test)
* Flight test (captive flight test, free flight test)

Table 5.6-2 shows the draft test schedule including the maintenance activities before and after the flight test. This draft schedule is based on the range maintenance work flow shown in Fig. 5-3; according to the draft schedule, a test cycle takes about 10 days for a test period.

(2) Safety measures against abnormalities in the flight

It is necessary to secure safety of the surrounding environment and facilities by one method or another when an abnormality occurs. To address control abnormalities during flight, in particular, the airframe should be equipped with an emergency command system so that, on receiving the engine cutoff command, the airframe may be made to drop safely and exactly inside the predetermined test security area.

The results of the examination of the safety measures examined are outlined in the following:

a) Detection of a flight abnormality

To judge whether a flight is normal or not, the flight position of the experimental vehicle is measured by the following methods:

<u>Methods</u>

1) By the optical tracking (to be confirmed by the observation on the ground)
2) By the IMU navigation data (to be self-contained onboard)
3) By the IMU navigation data (to be confirmed through the ground telemetry)

<u>Criteria</u>

a) Whether the vehicle is following the flight plan;
b) Whether the vehicle is within the test flight domain; and
c) Whether the vehicle is within the engine cutoff command issue boundary.

b) Method of making the vehicle drop

Shutting down the propellant flow to the engine of the experimental vehicle makes it unable to fly and it drops. The following two shutdown methods are available:

<u>Valve actuation</u>

1) By closing the the pilot valve (a three-way valve) for the propellant valve
2) By closing the thrust control valve (a regulating valve)

Signal path

1) From the command receiver/demodulator through GCC to the propulsion system controller.

2) From the command receiver/demodulator directly to the propulsion system controller.

Transmission of command

1) Judgement made on the ground/transmission of the command (optical detection, telemetry data)

2) Self-contained onboard (programmed actions)

(3) Effects of disturbances on the flight range

Assuming that the flight test is conducted in a steady wind with a limit wind velocity of 5m/sec (up to 10m/sec for gust)* the deviation in the horizontal direction was calculated using a simplified calculation method.

* Assuming that the airframe characteristics are based on the KHI concept option, the deviation to be caused by a wind with this much velocity is 1m at most; this allows one to judge that there are no problems with the test security domain and test safety. However, errors in the equipment such as IMU in the navigation guidance and control should be examined along with errors in the control system in the future.

With respect to the effect of the disturbance (wind) on the attitude control and safety, no major problem occurs provided that the control requirements given in paragraph 5.1.(3) are met.

Table 5.6-1 Flight test plan (draft)

Takeoff and landing flight experiment concept

(1) Test site: a remote site home (Hokkaido or Tanegashima Island area)
(2) Test period: Summer through Fall, 1997
(3) Flight test period: The planned actual flight test period is for about 2 months (3 months at most). (the periods for flight test site maintenance, preparation for the test, and decommissioning are excluded.)
(4) Test item:
  A) Ground test
    <a> Electric system overall functional test
    <b> Propulsion system ground combustion test (including the maintenance activities before and after the test)
  B) Flight test
    <a> Captive flight test (including the maintenance activities before and after the test)
    <b> Free flight test (including the maintenance activities before and after the test)
(5) Test procedure: The test starts with the ground combustion test and proceeds with the test pattern evolving from simple to complex ones; for the detailed test procedure, refer to Paragraph 5.1.
(6) Test frequency: About 10 days /test cycle, 5 to 6 times in total/2 months (on the assumption that test activities are carried out on workdays with good weather that allow a full day's work)
(7) Test personnel: 12 persons on the average

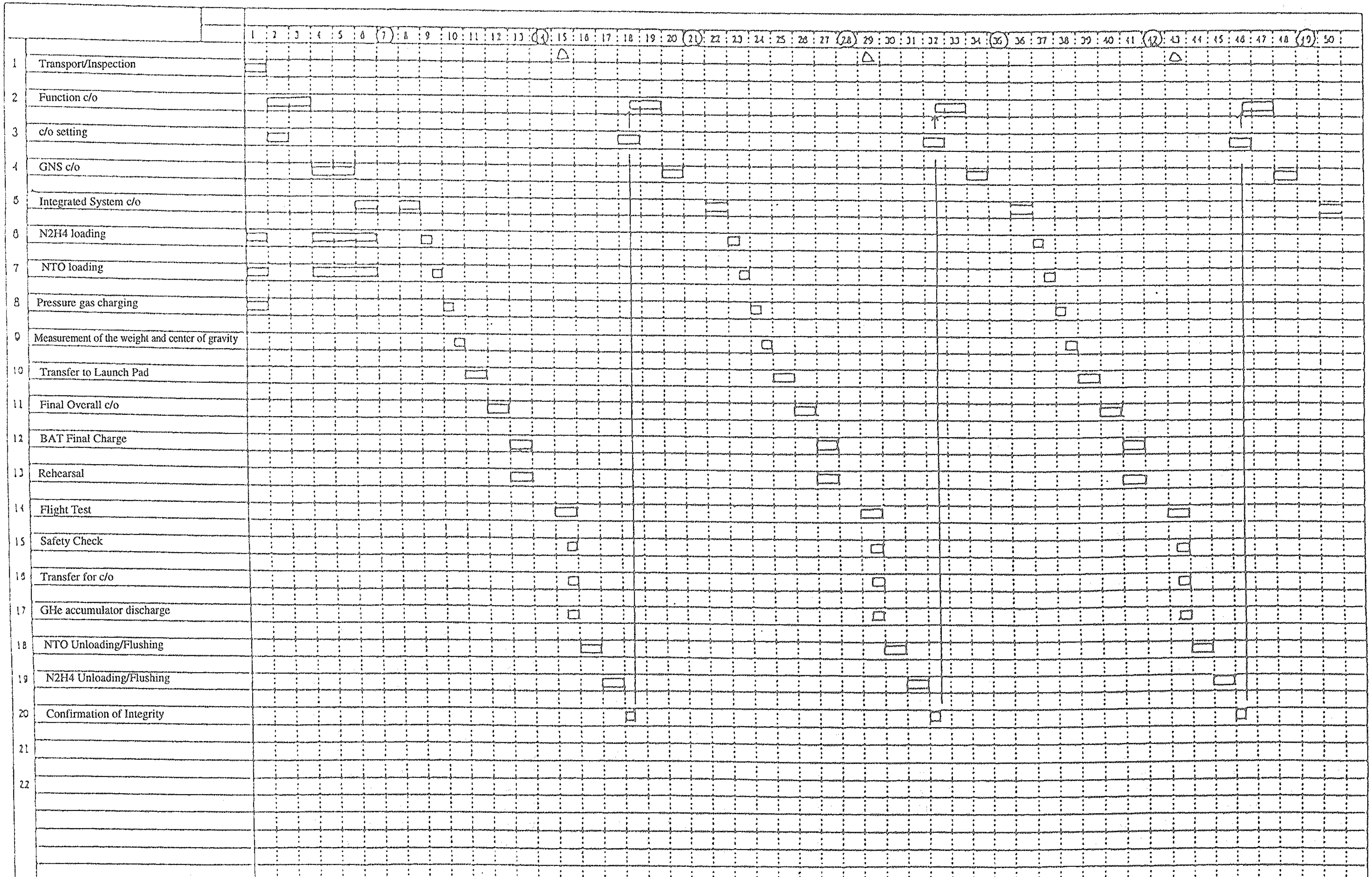
Number of persons by job

* Test management (principal/assistant)		2
* Airframe technology	Structure/landing gears/heat	1
	Propulsion system	2
	Navigation guidance and control/electric	2
	Fabrication/inspection technology	1
* Range facilities	Facilities (mechanical and electrical)	2
	Propulsion system	2

* Flight operations	Control	1 (job concurrently held by a person)
	Communications	2 (job concurrently held by a person)
	Measurement	2 (job concurrently held by a person)
	Safety	1 (job concurrently held by a person)

(8) Procurement of propellant and other materials: the propellant and pressure gas are procured/purchased according to the number of tests (combustion tests). The maximum amount of fuel or oxidant consumed in a test (combustion test) is about 75 liters.
(9) Other matters:
  <1> In the present plan, the initial condition is defined as one in which the arrangement of the flight test site/AGE and the like has been finished and a test can be started at any time.
  <2> In compliance with relevant laws and regulations and from the viewpoint of safe operations of the project, activities affecting safety (gas pressurizing/discharge, propellant fill/drain) are performed by qualified specialist operators.
  <3> Activities and jigs (containers and the like) required by transporting the airframe to the site (from the factory to the flight test site) are taken into consideration.
  <4> It is assumed that the equipment/devices that are not newly fabricated/purchased are to be borrowed and that utility services such as water supply and waste liquid disposal are provided.

Table 5.6-2 Range maintenance activities schedule (draft)



6. Ground Equipment

6.1. Basic Policies

Because the reusability is given priority, the site for takeoff and landing should be examined along with the examination of the airframe. The selection of the test site is considered from the following viewpoints:

(A) Sufficient area for the flight test (security distance);
(B) Convenience of carrying in materials and constructing facilities;
(C) Absence of high buildings;
(D) Absence of houses;
(E) Less difficulties and costs in borrowing the land;
(F) No private ownership; and
(G) Domestic.

A site meeting the requirements listed above is to be found in Hokkaido.

The ground facilities consists of the following systems:

(A) Range system;
(B) Control system;
(C) Data processing system;
(D) Ground communications system;
(E) Ground measurement system.

Figs.6.1-1, 6.1-2, and 6.1-3.

6.2 Semi-captive Test

Fig.6.2-1

Before conducting the free flight test, the basic functions of the experimental vehicle is confirmed with the binding fixtures attached. Most of the items to be confirmed in the vertical jump can be demonstrated by this semi-captive test.

Thrust control performance
Gimbal performance
Attitude static stability
Altimeter performance
Shock absorbing landing gear design
Flame protection measures
Soft landing and airframe recovery
Reusability
Standard descent
Ground operability

6.3 Flight Test Domain (safety domain)

Since this experimental vehicle uses NTO/N2H4 and the airframe returns to the landing site, greater consideration for the environmental preservation is necessary than with a common rocket.

On the basis of the maximum flying capability (horizontal flight limit), the flight test safety distance was determined, on which the demand for a flight test suite was decided on.

(A) Maximum range capability

Without considering the experimental vehicle's return to and landing on the test site, the horizontal flight limit distance (inertial flight and free fall after fuel has run out.) was determined. This corresponds to the abnormal flight condition which incapacitates flight control through failures in the control system after launching.

The horizontal flight limit distance is about 5[km]. See Fig.6.3-1.

For reference, the vertical flight limit distance (maximum attainable altitude) is 1.4[km].

(B) Test safety control area

As described above, the horizontal flight limit is considerably great, making it difficult to provide a test safety control area as extensive as this distance. (from the viewpoint of securing an extensive site)

To solve this problem, a measure to narrow the flight security area is needed. For this purpose, a forced dropping boundary inclusive of the distance margin is determined on the basis of the expanse of the test area to be specified test by test; for an abnormal flight going beyond this boundary, the engine cutoff command is issued to make the vehicle drop safely, thus narrowing the required security area.

The engine cutoff command issue limit boundary is set at a distance of about 300[m] from the launch pad, with the test site safety distance set at a radial distance of 1.2[km] on the basis of the dropping flight domain of the vehicle and the safety distance required for the propellant. The area defined by this setting is still too large, requiring reconsideration in the future. Refer to Figs.6.3-2 and 6.3-3.

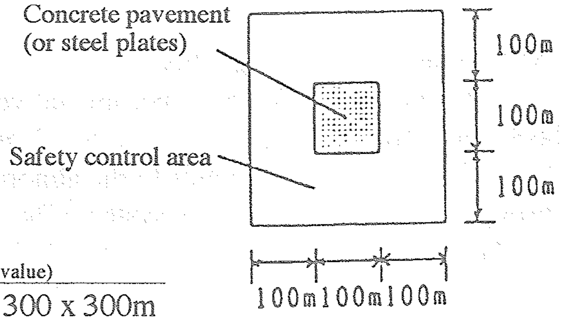
6.4. Launch Support System

* The takeoff and landing test site requires a location where a concrete paved area of about 100 x 100 m can be secured; if the available area is smaller than this requirement, the balance may be covered by laying steel plates. In addition, a flat safety control area that encloses the paved square lot at a distance of 100m or larger from each of its sides is provided.

* "Flight domain examination" (based on the original plan)

1) Safety distance: 90m
2) Horizontal distance over which the vehicle can move: 40m
3) Margin in the error in the navigation guidance and control system and others: 20m(temporary value)

150m phi→ 300 x 300m



* The experimental vehicle is installed on the launch pad by a crane.
* The ground support system is a portable one that can be self-propelled, towed, or lifted.

400 Fabrication of ground Equipment

410 Launch system

(1) Support Equipment

a) Flight test site maintenance 1 set (to be diverted): Provision of a concrete paved lot with a dimension of about 100 x 100 m and a flat security zone that encloses the paved square lot.

b) Movable flame deflector 1 (to be fabricated): Made of steel plates painted with a heat resistance material.

c) Maintenance and inspection car 1 set (to be fabricated):
* Unpacking of the experimental vehicle, inspection of the quantity and appearance.
* Function inspection (measurement and inspection of gimbal valve operation and airtightness leak).
* Measurement of the airframe weight and the weight of the loaded propellant.

d) Air conditioning car 1 (to be rented): * Air conditioning of the airframe and maintenance of the environment for propellant charging and discharging

e) Power supply car 1 (to be rented): * Supply of electric power to the airframe, AGE, facilities, and others

f) Crane car 1 (to be rented): * Hoisting of the airframe for measuring the weight of the airframe and loaded propellant.

(2) Equipment for airframe inspection and maintenance

a) GCC checkout unit 1 (to be rented): * The software is to be created.

b) Battery charging unit 1 (to be rented): *Charging of the airframe battery

c) Propulsion system checkout unit 1 (to be fabricated): * Inspection of the function of the propulsion system

(3) Propellant related equipment: * Loading of the propellant and pressurizing of the accumulator

a) Propellant loading unit 2 sets (to be fabricated): * Flushing of the propellant discharge line and propellant gas loading/unloading line

b) Propellant neutralizing unit 2 sets (to be fabricated):

c) Pressurizing gas charging unit 1 set (to be fabricated): * Treatment of waste liquid and exhaust gas

d) Propellant transportation vessel 2sets (to be borrowed): * Simplified flushing

e) IPA/water supply unit 1 set (to be fabricated):

(4) Safety/environmental preservation related equipment:

a) Fire engine 1 (to be borrowed)

b) Water truck 1 (to be rented)

c) Waste liquid tank 2 (to be fabricated)

d) NTO storage 1 (to be fabricated)

e) Hydrazine storage 1 (to be fabricated)

f) Shower/eye-bath 1 (to be fabricated)

g) Protective clothing/gas densitometer 1 set (to be borrowed)

h) Breathing air supply unit

(5) Umbilical

a) Umbilical 1 set (to be fabricated)

(6) Others

a) Operations building 1 (to be built): * For flight control, ground communications, ground measurement, processing of experiment data, waiting room for the experiment group

b) AGE storage 1 (to be fabricated)

c) Truck 1 (to be rented)

d) Tools 1 set (to be borrowed)

6.5. Control System

The ALFLEX facilities are to be diverted in principle.

Flight control console
Flight control processing unit

6.6. Experimental Data Processing System

The ALFLEX facilities are to be diverted in principle.
(T.B.D.)

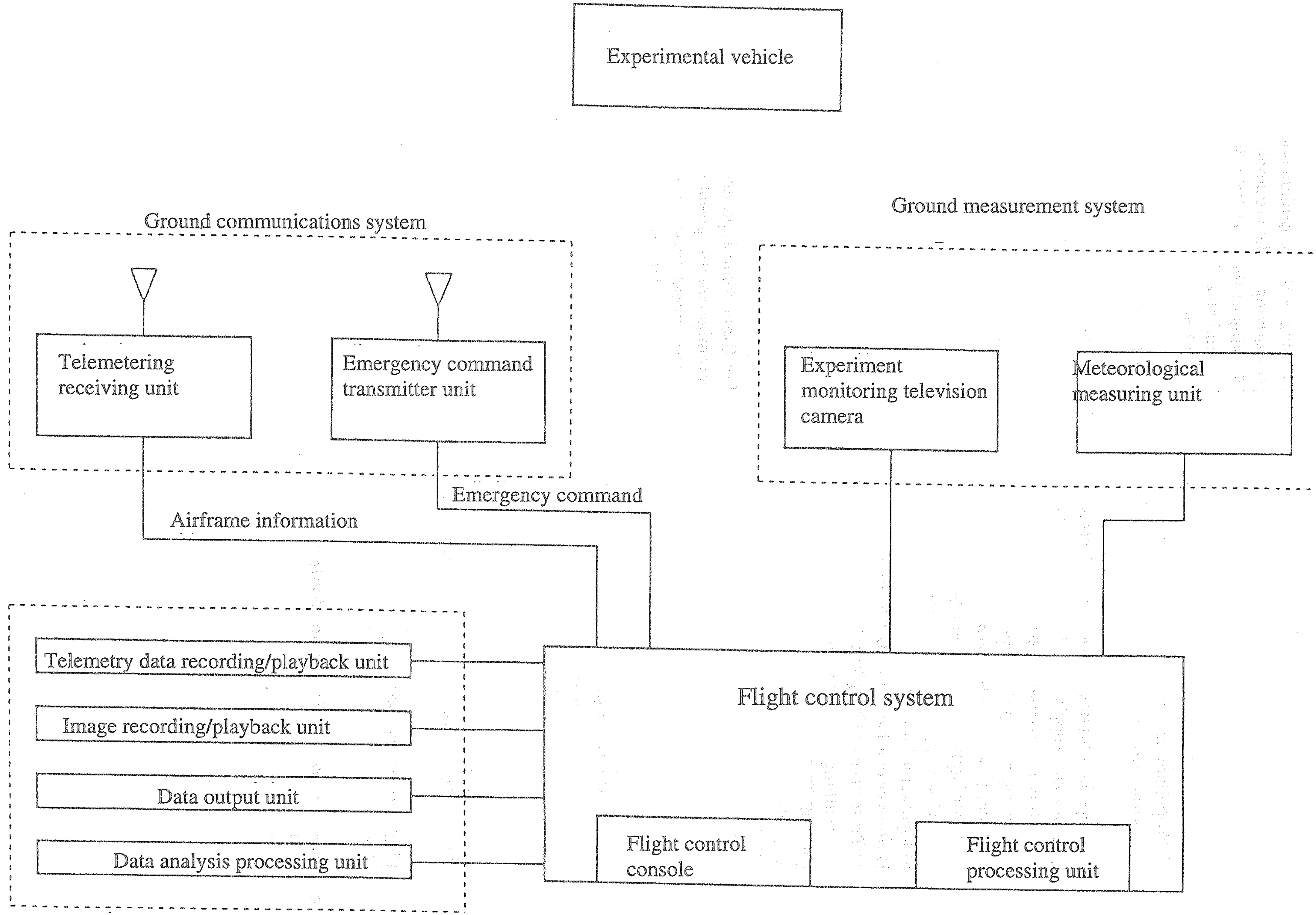


Fig.6.1-1 Schematic diagram of the ground facility configuration

6.7. Ground Communications System

The ALFLEX facilities are to be diverted in principle.

The facilities required for the tracking control are temporarily installed in the vicinity of the takeoff/landing point.

Command items are limited to those necessary for the loading of the propellant and actuation of the emergency systems. Telemetering items are limited to the HK data for the experimental vehicle.

(A) Configuration

The ground communications system is composed of the following components:

S-band communication antenna
S-band transmitter/receiver
S-band power amplifier
S-band high output power amplifier
S-band low noise amplifier
S-band diplexer
S-band antenna switch network
Switch network

(B) Performance

(1) Frequency band used

Up-link

S-band: selected from frequencies between 2025 and 2110[MHz]

Down-link

S-band: selected from frequencies between 2200 and 2290[MHz]

(2) Modulation mode

	Transmission mode	Line code system	Modulation system
Command signal	PCM	NRZ/L	PSK/PM
HK data	PCM	Bi ø-L	PSK/PM
Environment measuring signal	PCM	Bi ø-L	PSK/PM

(3) Transmission output

T.B.D.

(4) Minimum reception sensitivity

T.B.D.

(5) Antenna power /gain

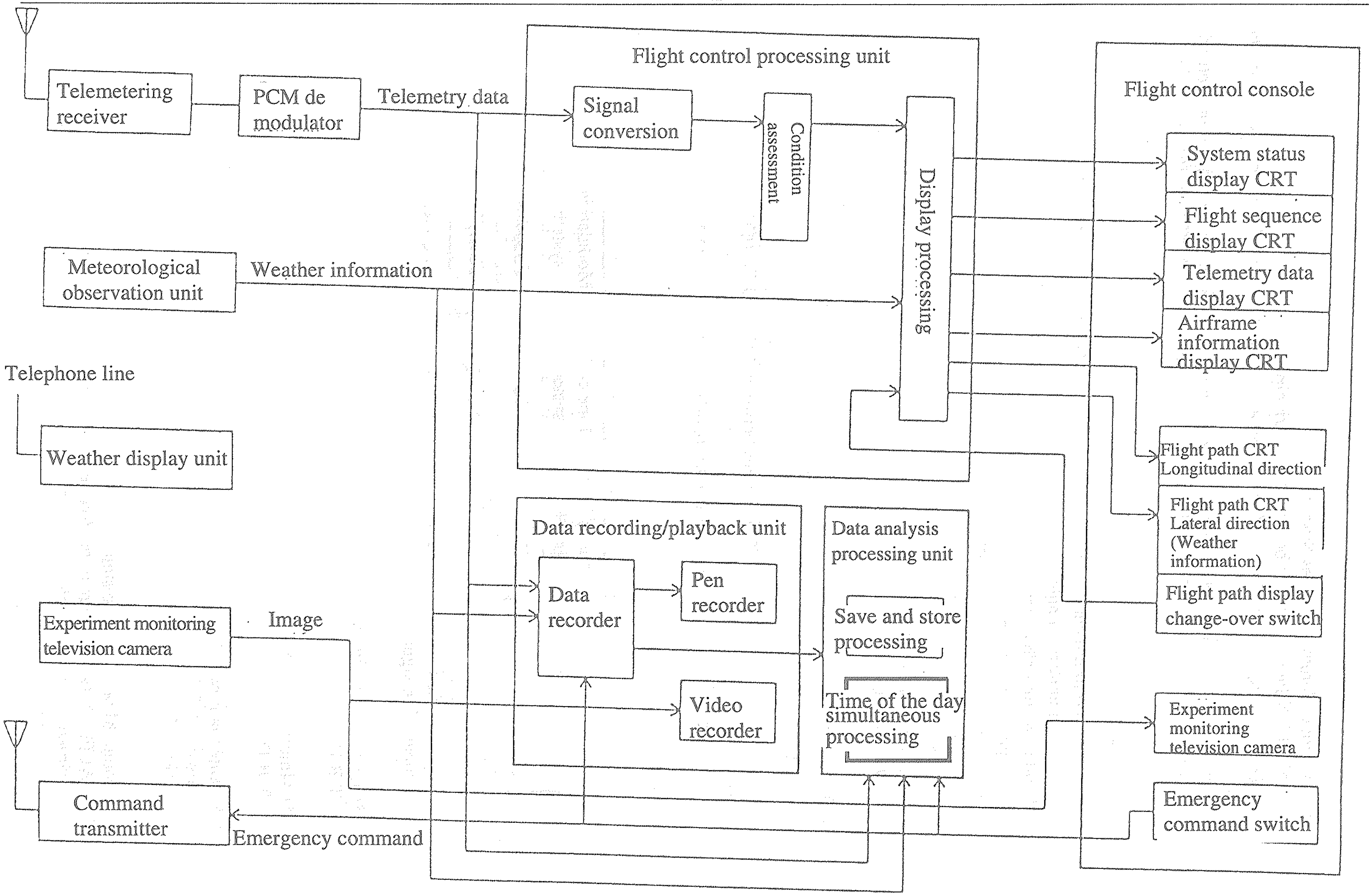
Power : T.B.D.
Gain : T.B.D.

6.8. Ground Measurement system

The ALFLEX facilities are to be diverted in principle.

Experiment monitoring television camera
Laser tracker
Meteorological measuring unit

Takeoff and landing flight experiment concept Ground facility plan



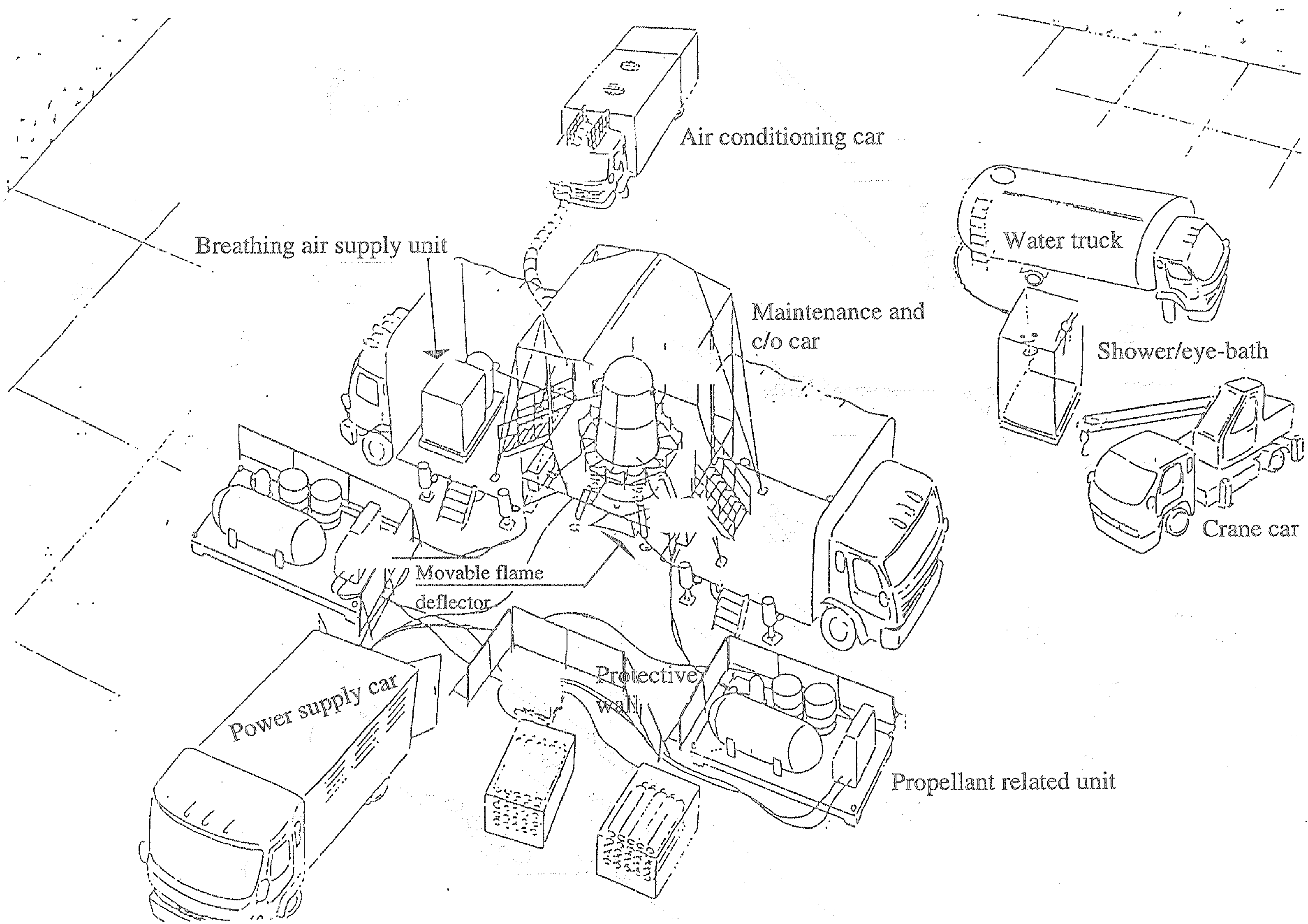


Fig.6.1-3 Outline of the takeoff and landing site facilities

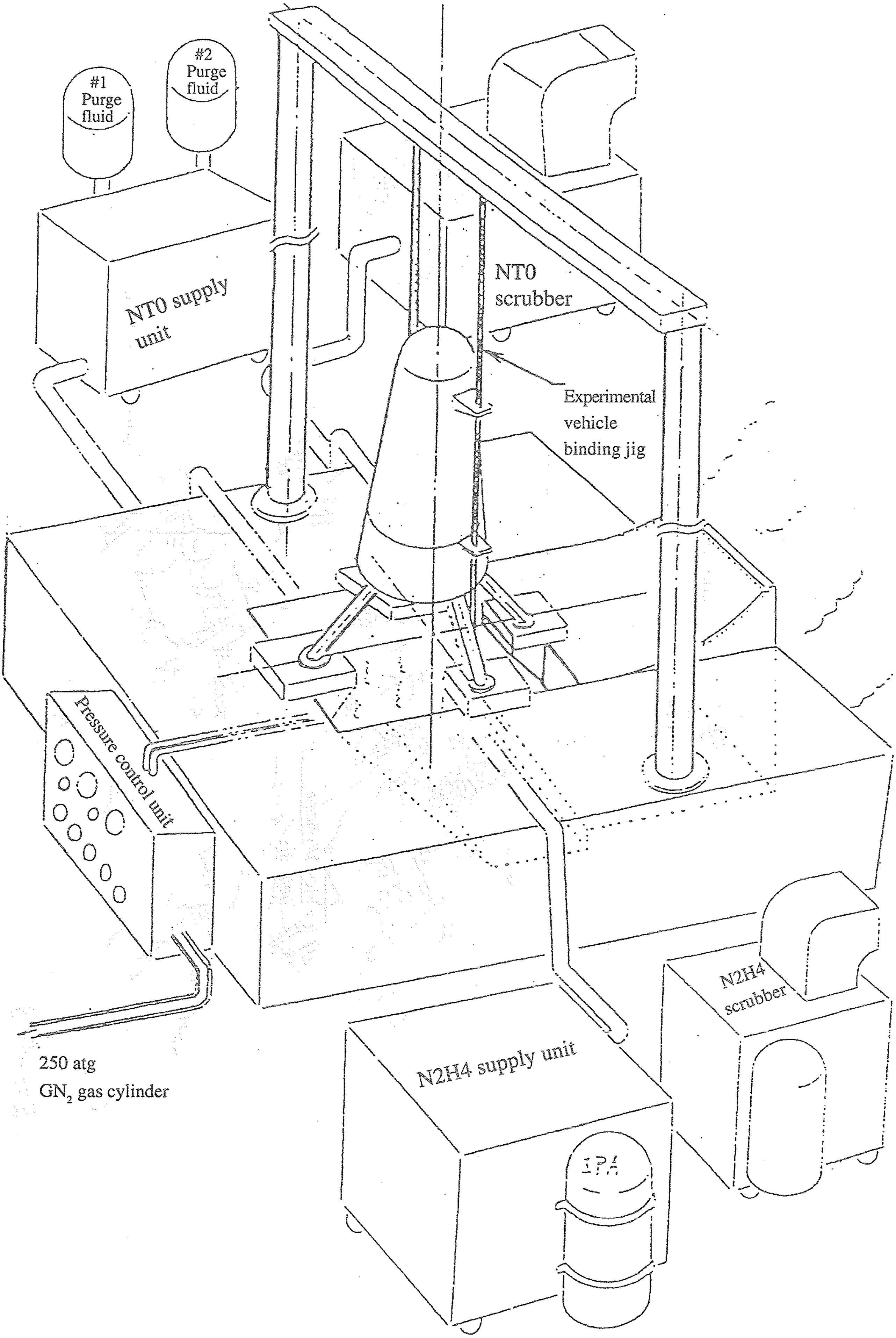


Fig.6.2-1 (No.1) Conceptual drawing for the test facilities

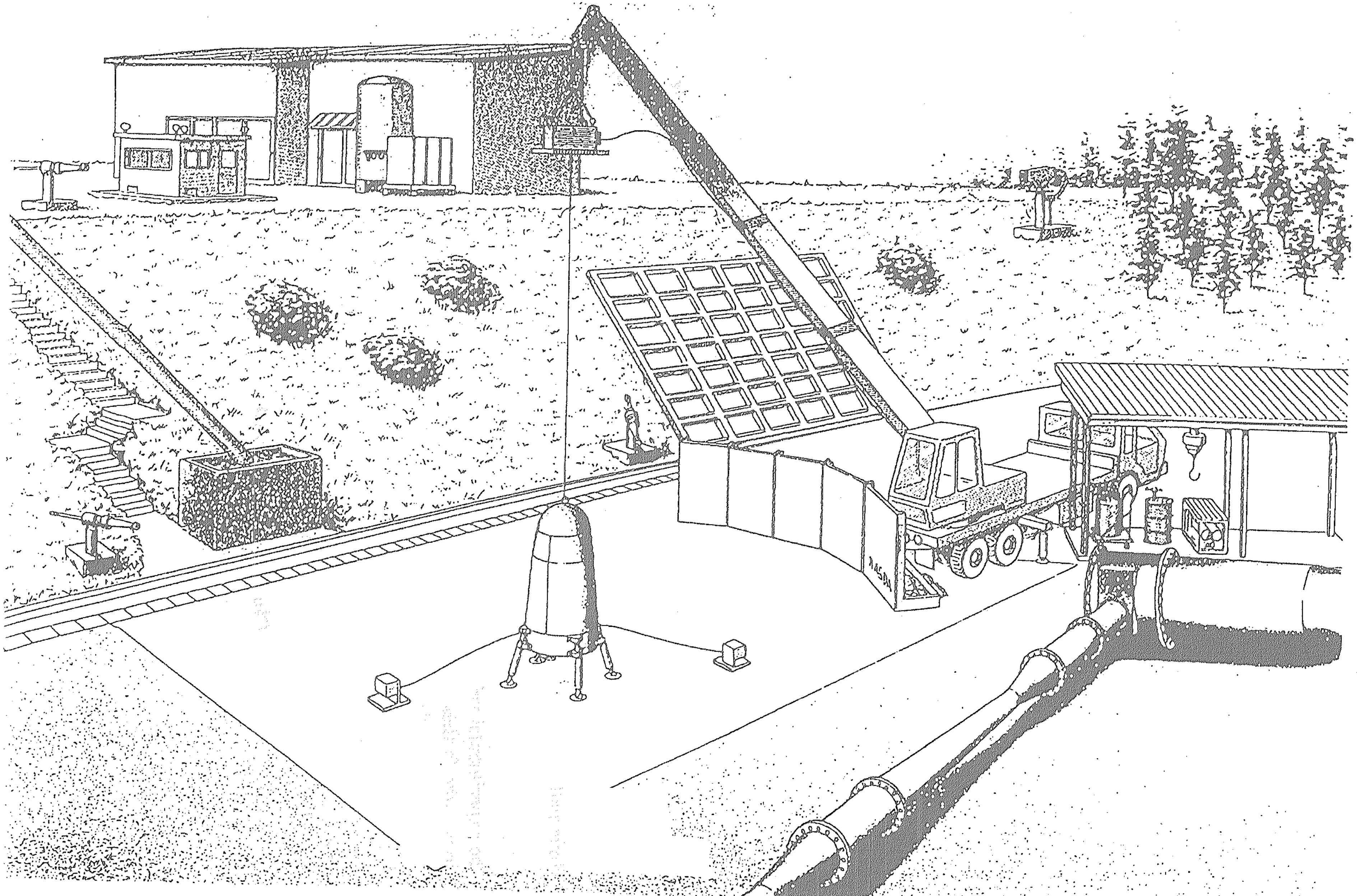


Fig.6.2-1 (No.1) Conceptual drawing for the test facilities

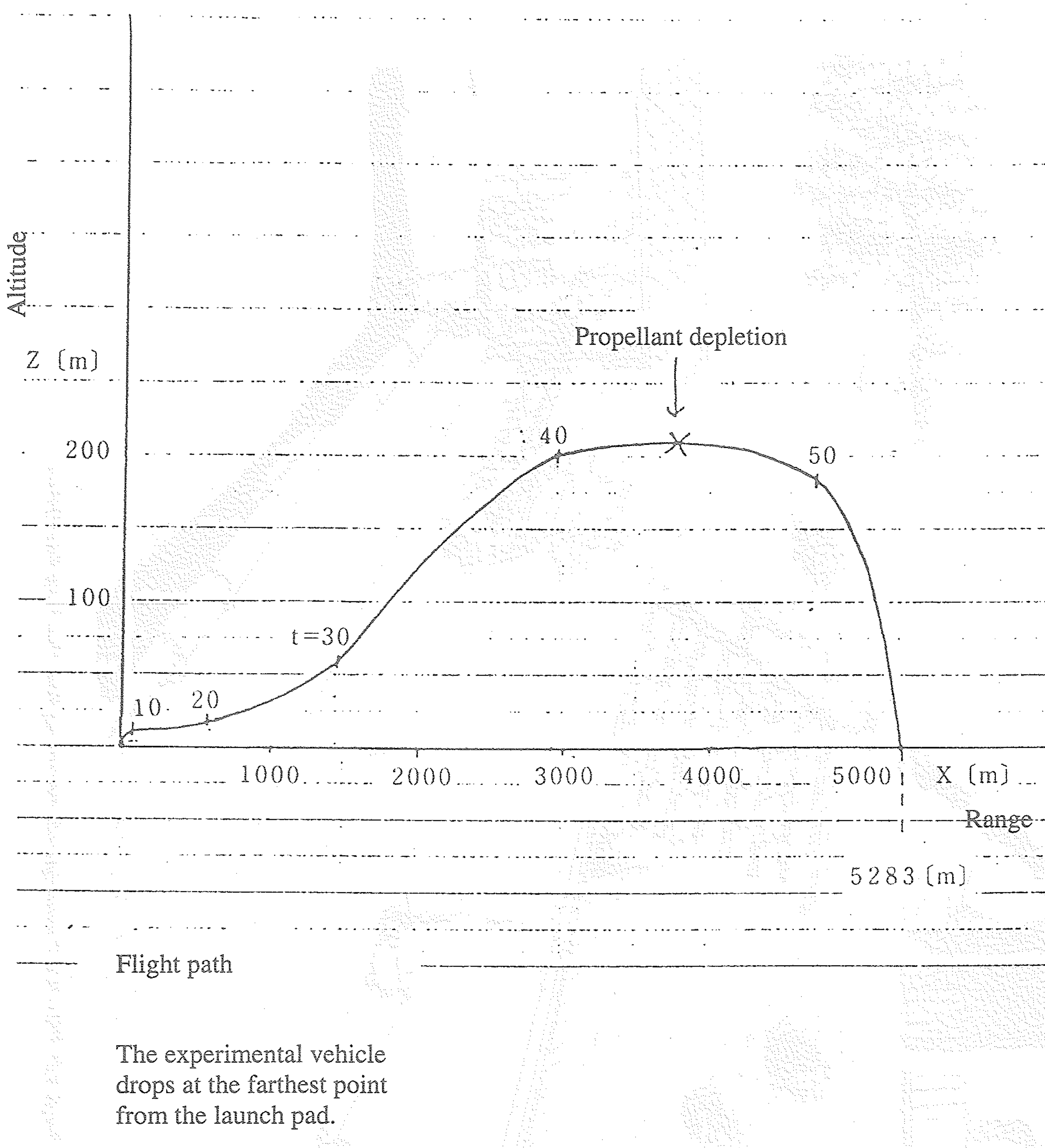


Fig.6.3-1 Flight path

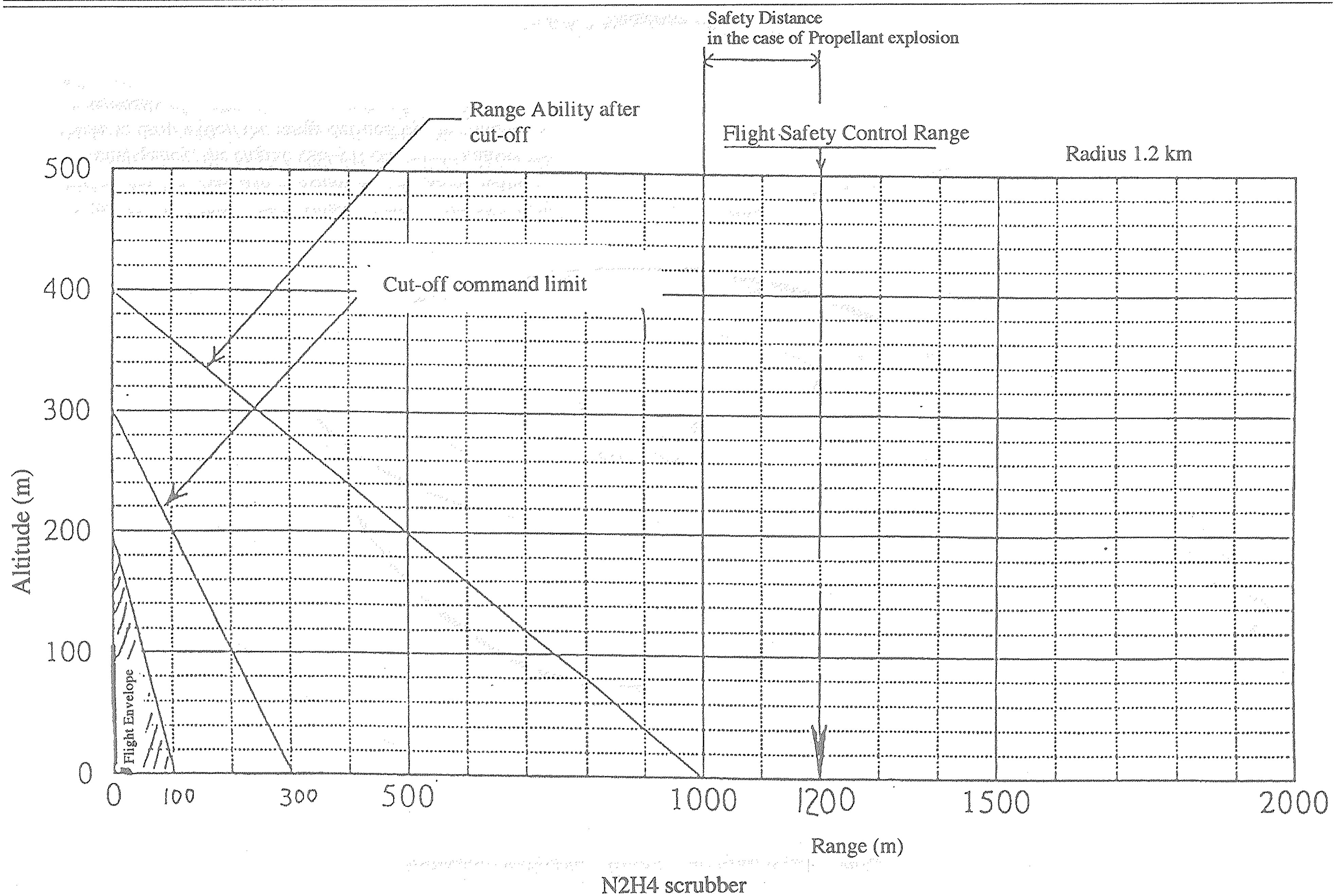
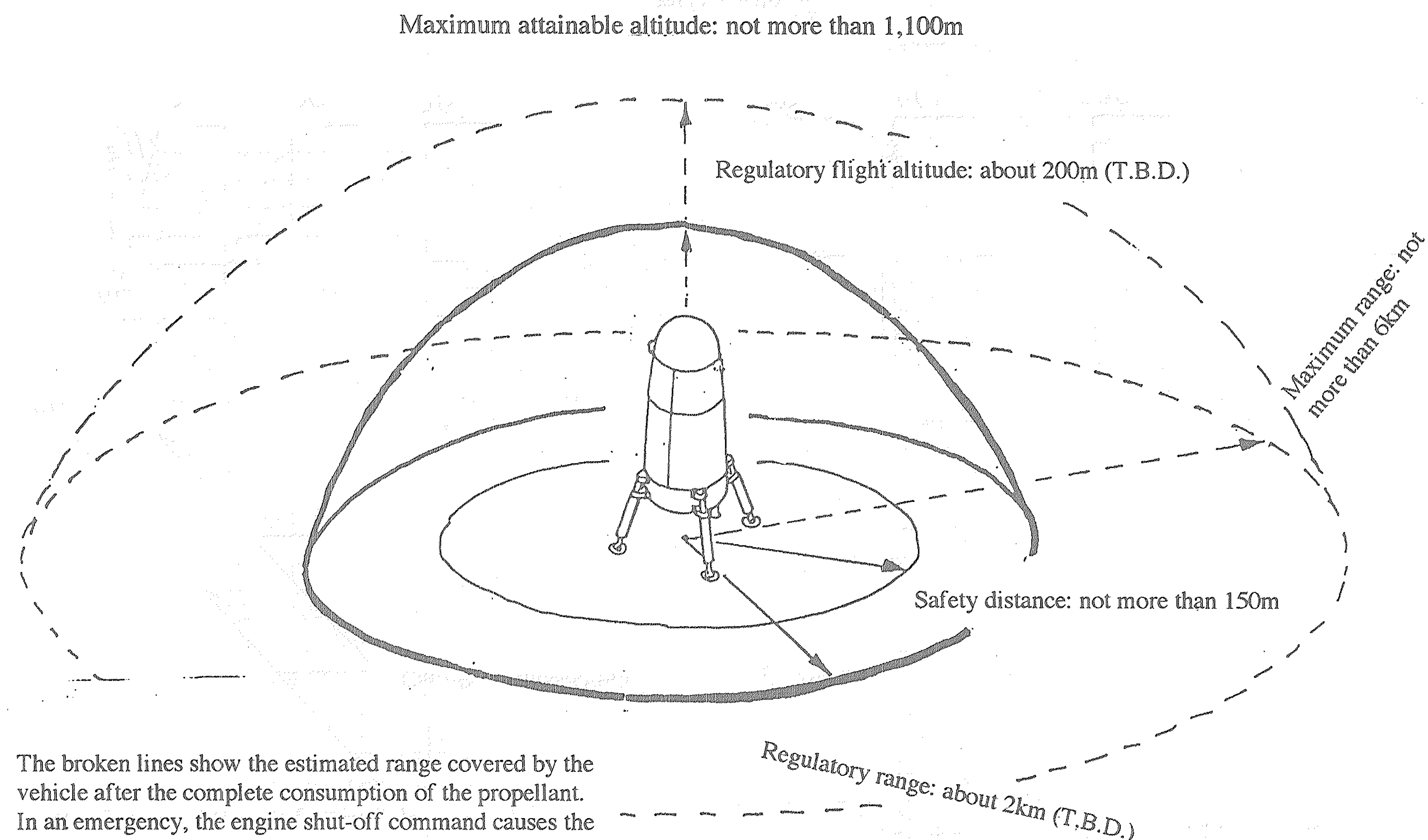


Fig. 6.3-2 Safety Control Area (Draft)



The broken lines show the estimated range covered by the vehicle after the complete consumption of the propellant. In an emergency, the engine shut-off command causes the vehicle to drop within the range defined by the solid lines. The necessity of destruction by command is examined in the future.

Fig.6.3-3 Expanse of the test site
Flight safety

7. Operational Plans

The basic policies for the operations of the experimental vehicle are as follows:

(A) The experimental vehicle is carried in the test site in the assembled condition;

(B) The pre-flight maintenance activities are performed on the maintenance and inspection car; before launching, the maintenance and inspection car moves out of the test area;

(C) The power supply car is equiped at the launch pad to feed power to the vehicle; before launching, the power supply is changed over from the power supply car to the onboard power supply;

(D) After the experimental vehicle has landed, the propellant leak inspection is conducted; after confirmation of safety, access to the experimental vehicle is permitted to close the isolation valves and perform activities such as pressure reduction, followed by servicing by the maintenance and inspection car;

(E) After this, the propellant is unloaded and recovered; the loading and unloading line is flushed, holding inner pressure by GN2

(F) After making certain that no danger exists, the experimental vehicle is inspected and data is recovered, and then the maintenance activities for the vehicle is started.

8. Investigation and Examination of Possibility of Component Diversion

To achieve the goal of developing the experimental vehicle in as short a period as possible with a minimum of cost, the basic development policy is to place emphasis on utilization and diversion of items that have been already developed, existing products, and items used for research activities, thereby structuring the entire vehicle by using existing technologies as extensively as possible.

From this viewpoint, methods of procuring equipment and components and histories of their performance were reviewed while the requirements for such equipment and components were confirmed.

(1) Component procurement plan

The procurement plan for all components (including pieces of equipment) composing the vehicle has been put to rearrangement, producing the results shown in table 8-1. The results of the examination of basic items are given sub system by subsystem as follows:

a) Propulsion system

* For the following items in the engine and thrust control systems, those used for the cluster combustion test are diverted:
  - Engine system: engines, propellant valves, three-way valves, pressure sensors
  - Thrust control system: regulating valves, filters, discharge valves
* For the following items in the tank systems, those used for the throttling combustion test are diverted:
  - Exhaust valves, loading/discharging valves, manual isolation valves, flowmeters, pressure sensors
* For the gimbal actuators, consumer motor-operated actuators are used;
* A spherical Ti tank is newly fabricated for a propellant tank;
* For the accumulator, the existing products (COMMETS) are diverted. (to be furnished by NASDA)
* For valves and sensors in the pressurizing system, proven products (HYFLEX) are procured.
* The control section of the propulsion system is separated from CIU in the navigation guidance and control system; it will be procured through modification of consumer products.

b) Structure system

* The landing gears are procured by developing new items through modification of consumer products. (motorcycle shock absorbers)

c) Navigation guidance and control system and communications system

* For the following components in the navigation guidance and control system, the existing products are diverted:
  - GCC, IMU (items used for research by NASDA)
  - RA and RA antenna (ALFLEX)
* For the following pieces of equipment, the existing products are diverted as stated in the NASDA original draft:
  - Telemetering receiver, PCM encoder (ALFLEX)
* For the command receiver, a consumer product (for radio-controlled flying model use) is used.

(2) Environment resistance of the equipment

The results of the examination of the environment resistance of basic items are given subsystem by subsystem as follows:

a) Propulsion system

* Vibration tests at the component and the subsystem level are not conducted; confirmation is collectively conducted through CFT.

b) Structure system
   * The landing gears and airframe bottom are covered with a thermal protection material.

c) Navigation guidance and control system, communications system, and power supply system
   * Since a damper cannot be applied to an IMU (item used for research by NASDA), it is necessary to conduct the acoustic test or random vibration test on an IMU alone.
   * The resistance to the launching environment of the pieces of electric and electronic equipment, excluding the IMU, have not been confirmed; therefore, they are mounted to the airframe through the medium of shock mounts or the like.
     (the equipment is exempted from the vibration test and other ones to alleviate the burden on it.)
   * The thermal environment during flight of the section accommodating the equipment is not severe; therefore, thermal protection measures are not necessary.
     Before launching, the maintenance and inspection building is air-conditioned by the ground support facilities.

(3) Acoustic environment of the airframe

To evaluate the effects of the acoustic environment on the airframe and onboard equipment, acoustic analysis was conducted with the acoustic effect at the takeoff as the reference.

* The sound level was higher than the inner sound field of the fairing of the H-II rocket; however, with the decay of the sound field by the effect of the outer shell taken into account, the sound level is considered to be weaker than that of the H-II rocket.
  - H-II rocket: 140dB (0.A)
  - Experimental vehicle: 142dB (0.A) *

  Note) * No sound field decay by the effect of the outer shell; acoustictransformation efficiency: 0.15%

Table 8-1 Component procurement plan (1/4)

System		Component designation	Quantity	Procurement method	Proven applications	Remarks	NASDA classification
Propulsion system	(1) Engine system	(a) Engine	2	Diversion of the items used for the cluster	J-1 EVE		321
		(b) One-fluid propellant valve	4	combustion test	COMMETS UPS		321
		(c) Three-way valve (propellant pilot valve)	4	Ditto			321
		(d) Pressure sensor	2	Ditto			321
		(e) Pipe lines/harnesses/brackets	1 set	Ditto			321
		(f) Fasteners	1 set	To be fabricated (new)			321
				To be purchased (new)			
	(2) Thrust control system	(a) Regulating valve	4	Diversion of the items used for the cluster			322
		(b) Filter	4	combustion test			322
		(c) Manual isolation valve (discharge valve)	4	Ditto			322
		(d) Pipe lines/harnesses/brackets	1 set	Ditto			322
		(e) Fasteners	1 set	To be fabricated (new)			322
				To be purchased (new)			
	(3) Gimbal system	(a) Gimbal actuator	4	To be purchased (new)			323
		(b) Gimbal mechanism	2	To be fabricated (new)			323
		(c) Harnesses	1 set	Ditto			323
		(d) Fasteners	1 set	To be purchased			323
	(4)Tank system	(a) NTO tank	1	To be fabricated (new)		Spherical Ti alloy tank	324
		(b) N2H4 tank	1	Ditto		Ditto	324
		(c) Exhaust valve	2	Diversion of the items used for the			324
		(d) Loading/discharging valve	2	throttling combustion test			324
		(e) Manual isolation valve	4	Ditto			324
		(f) Flowmeter	2	Ditto			324
		(g) Pressure sensor (low pressure use)	2	Ditto			324
		(h) Temperature sensor (low pressure use)	2	Ditto			324
		(i) Pipe lines/harnesses/brackets	1 set	To be purchased (new)			324
		(j) Fasteners	1 set	To be fabricated (new)			324
				To be purchased (new)			
	(5)Pressurizing system	(a) accumulator	1	Existing/to be furnished by NASDA	COMMETS UPS		325
		(b) Loading/discharging valve (high pressure	1	To be purchased (already developed)	HYFLEX		325
		use)	1	Ditto	HYFLEX		325
		(c) Isolation valve (high pressure use)	1	Ditto	HYFLEX		325
		(d) Filter	1	Ditto			325
		(e) Pressure control valve	2	Ditto			325
		(f) Check valve	2	Ditto			325
		(g) Pressurizing isolation valve	1	Ditto			325
		(h) Safety valve	1	Ditto	HYFLEX	Composed of the	325
		(i) Pressure sensor (high pressure use)	1	Ditto	HYFLEX	thrust magnitude	325
		(j) Temperature sensor (high pressure use)	1 set	To be fabricated (new)		control unit,	325
		(k) Pipe lines/harnesses/brackets	1 set	To be purchased (new)		regulating valve	325
		(l) Fasteners				driver, gimbal driver,	
	(6) Controller system	Propulsion system control unit (CIU)	1	To be newly fabricated/modification of consumer products		and valve driver.	334

Table 8-1 Component procurement plan (2/4)

System		Component designation	Quantity	Procurement method	Proven applications	Remarks	NASDA classification
Propulsion system	(7) Auxiliary thrust system	(a) Thruster/propellant valve	0				326
		(b) Filter	0				326
		(c) Manual isolation valve	0				326
Structure system	(1) Main structure	(a) Upper structure	1	To be fabricated (new)			311
		(b) Lower structure	1	Ditto			313
		(c) Outer shell	1	Ditto			314
		(d) Fasteners	1 set	To be purchased			311
	(2) Landing gears	(a) Landing gears	4	To be fabricated (to be newly developed)		To be developed by	312
		(b) Fasteners	1 set	To be purchased		modifying	312
Thermal control system	(1) Flame protection	(a) Airframe flame protection	1	To be fabricated (new)		commercially	361
		(b) Landing gear flame protection	4	Ditto		available products	361
		(c) Fasteners	1 set	To be purchased (new)			361
	(2) Temperature control	(a) White painting on the outer shell	1 set	To be purchased (new)		Thermal isolation	362
		(b) Thermal insulating materials	1 set	Ditto		spacer/block and the	362
	(3) Temperature measurement	(a) Temperature sensor	1 set	To be purchased (new)		like	362
Navigation guidance and control system		(a) Onboard software	1	To be fabricated (to be newly developed)	Items used for		331
		(b) Inertia sensor (IMU)	1	Existing/to be furnished by NASDA	research by		332
		(c) Radio altimeter (RA)	1	Existing/to be furnished by NASDA	NASDA		332
		(d) RA antenna	1	Existing/to be furnished by NASDA	ALFLEX	NASDA	332
		(e) Onboard computer (GCC)	1	Existing/to be furnished by NASDA	ALFLEX	Excluding the	333
		(f) Control interface unit (CIU)	1	To be fabricated (to be newly developed)	Items used for	propulsion system	334
		(g) Connector/harnesses	1 set	To be fabricated (new)	research by	control section	353
Communications system		(a) Telemetering transmitter	1	Existing/to be furnished by NASDA	ALFLEX		341
		(b) PCM encoder	1	Existing/to be furnished by NASDA	ALFLEX		341
		(c) Command receiver/demodulator	1	To be purchased (new)			342
		(d) Telemetering transmitter	1	Ditto			343
		(e) Command receiver antenna	1	Ditto			343
		(f) Connector/harnesses	1 set	To be fabricated (new)			343
Power supply system		(a) Onboard battery	1	To be purchased (new)		Including the	351
		(b) Distribution board	1	To be fabricated (to be newly developed)		power supply	352
		(c) Connector/harnesses	1 set	To be fabricated (new)		for the gimbal actuator.	353

Table 8-1 Component procurement plan (3/4)

System	Component designation	Quantity	Procurement method	Proven applications	Remarks	NASDA classification
Range system (1) Range facilities	(a) Launcher	1	To be fabricated (new)		Substituted for by a well-kept lot	Classification 4
	(b) Maintenance and inspection building	1	Ditto			
	(c) Air conditioning facility	1	To be rented		Including the umbilicals	
	(d) Power supply car	1	Ditto			
	(e) Crane car	1	Ditto			
	(f) Pipe line	1 set	To be fabricated (new)			
	(g) Harnesses	1 set	To be fabricated (new)			
(2) Airframe inspection and maintenance unit	(a) GCC checkout unit	1	To be rented		Software to be created	
	(b) Battery charging unit	1	Ditto			
	(c) Propulsion system checkout unit	1	To be fabricated (new)			
(3) Propellant related unit	(a) Propellant loading unit	2	To be fabricated (new)			
	(b) Propellant neutralizing unit	2	Ditto			
	(c) Pressurizing gas loading unit	1	Ditto			
	(d) Weight measuring unit	1	Ditto			
	(e) Propellant transportation vessel	2	To be borrowed (from the propellant manufacturer)	Propellant transportation vessel		
(4) Equipment for safety/environmental preservation	(a) Fire engine	1	To be borrowed (cooperation from the local authority)			
	(b) Water truck	1				
	(c) Waste liquid tank	1	To be rented			
	(d) NTO storage	1	To be fabricated (new)		Used containers to be used	
	(e) Hydrazine storage	1	Ditto			
	(f) Shower/eye-bath	1	Ditto		Ditto	
	(g) Protective clothing/gas densitometer	1 set	Ditto			
	(h) Breathing air supply unit	1 set	To be borrowed (from TNSC)			
			To be fabricated (new)			
(5) Others	(a) Operations building	1	To be fabricated (new)		Used containers to be used	
	(b) AGE storage	1	Ditto			
	(c) Truck	1	To be rented		Ditto	
	(d) Tools	1 set	To be borrowed (from TNSC)			
	(e) Others					

Table 8-1 Component procurement plan (4/4)

System	Component designation	Quantity	Procurement method	Proven applications	Remarks	NASDA classification
Control system	(a) Flight control console	1	To be borrowed/to be furnished by NASDA	Facility for ALFLEX		Classification 3 (344)
	(b) Flight control processing unit	1	Ditto	Ditto		Ditto
	(c) Flight control software	1	Ditto	Ditto		Ditto
Experiment data processing system (1) Recording/playback unit	(a) Telemetry data recording/playback unit	1	To be borrowed/to be furnished by NASDA	Facility for ALFLEX		Classification 3 (344)
	(b) Image data recording/playback unit	1		Ditto		Ditto
	(c) Data output unit	1	Ditto	Ditto		Ditto
(2) Analysis processing unit	(a) Data analysis processing unit	1	Ditto	Ditto		Ditto
Ground communications system						
(1) Antenna	(a) Antenna	1	To be borrowed/to be furnished by NASDA	Facility for ALFLEX		Classification 3 (344)
(2) Telemetering reception	(a) Telemetering receiving unit	1	Ditto	Ditto		Ditto
	(b) PCM demodulator	1	Ditto	Ditto		Ditto
	(c) Telemetering receiving antenna	1	Ditto	Ditto		Ditto
(3) Emergency command transmission	(a) Command transmitter	1	Ditto	Ditto		Ditto
	(b) Command transmitting antenna	1	Ditto	Ditto		Ditto
Ground measurement system						
(1) Meteorological measuring unit	(a) Meteorological observation unit	1	To be borrowed/to be furnished by NASDA	Facility for ALFLEX		Classification 3 (344)
	(b) Weather display unit	1	Ditto	Ditto		
(2) Experiment monitor	(a) Experiment monitoring television camera	1	Ditto	Ditto		Ditto

9. Development Schedule

Table 9-1

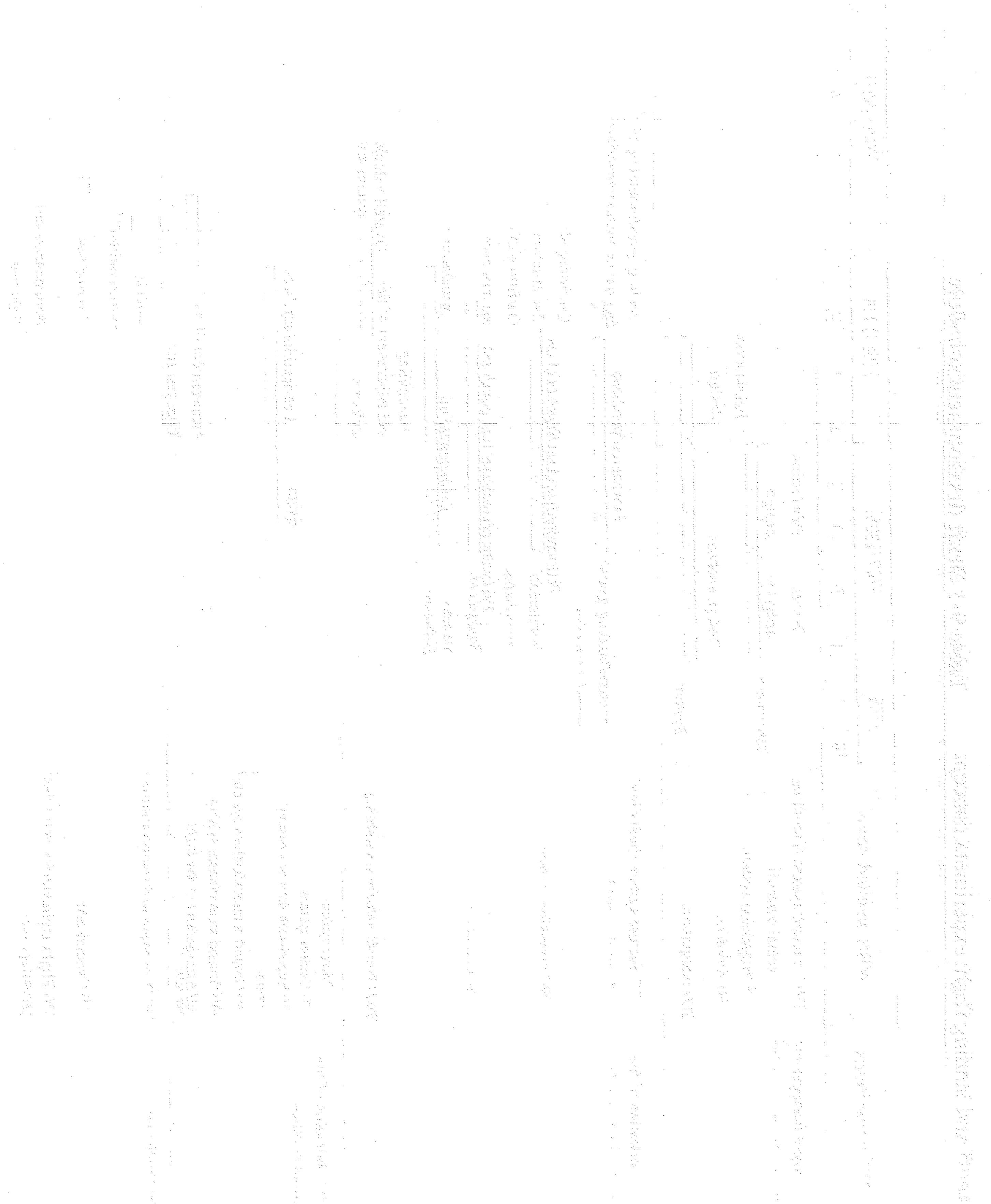


Table 9-1 Draft development schedule

WBS major items	WBS detailed items	FY06		FY07 (1995)				FY08 (1996)				FY09 (1997)				
		10	1	4	7	10	1	4	7	10	1	4	7	10	1	
100 Project management 200 Development design	210 Structure system (including the thermal control) 220 Propulsion system 230 Avionics	Subsystem		Design analysis		Fabrication design										
	280 Integration	System		Design analysis				Fabrication design								
300 Fabrication of the experimental vehicle	310 Structure system (including the thermal control)	Structure/landing gears/ thermal protection				Fabrication/purchase				Landing gears/mounting of thermal protection materials						
	320 Propulsion system			Equipment/ components		Fabrication/purchase/functional test				Outfitting of the structure						
	330 Avionics			Equipment/ harness		Fabrication/purchase/ functional test				Outfitting of the structure						
				Software		Design/creation				Installation						
	380 Overall vehicle assembling							Assembling and adjustment of the airframe				Overall vehicle system test				
400 Fabrication of the ground facilities	410 Range system 420 Control system 430 Experiment data processing system 440 Ground communications system 450 Ground measurement system					Design		Fabrication/purchase								
	460 Arrangement of the flight test site							Arrangement of the flight test site								
500 Flight test	510 Packaging and transportation									Vehicle transportation						
	520 Ground test									Ground test						
	530 Flight maintenance activities 540 Flight test									Maintenance and flight test						

10. Basic Tests

10.1 Test of the Operation of the Attitude Control Gas Jet under the Atmospheric Pressure

The 150[N] thruster, which has already been developed, is likely to be used as the attitude control gas jet for the experimental vehicle. Since the 150[N] thruster has been developed originally for operation in a vacuum, the nozzle was modified into a shorter form to achieve the optimum expansion under the atmospheric pressure; the modified thruster was put to the combustion test under the atmospheric pressure to acquire performance data.

Fig.10.1-1

Normal combustion performance

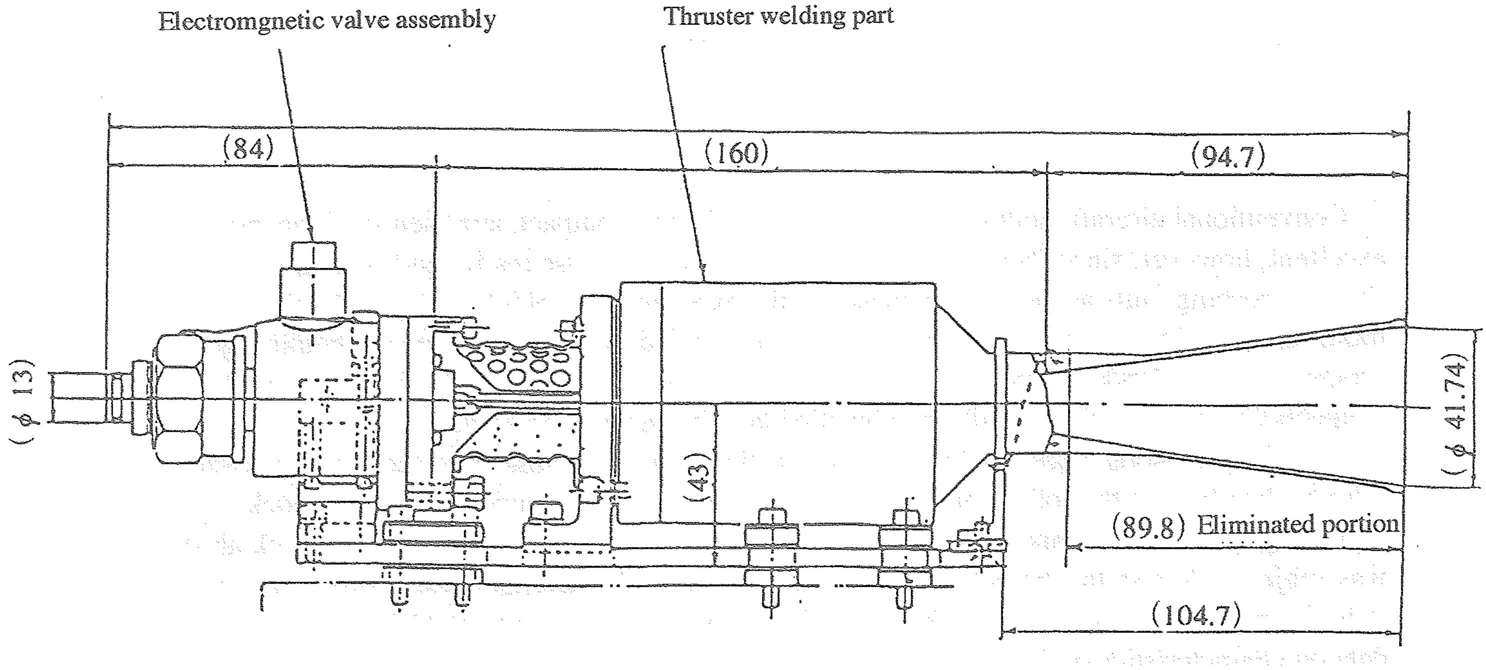
It was confirmed that all the normal combustion performance meets or exceeds the specified requirements; the evaluation for each of the performance items is shown in the following table:

Item	Test standard	Results of the test
Thrust under the atmospheric pressure	100±5N	The pressure at the propellant valve inlet is in the specified range of 16 - 18kgf/cm2a.
Combustion pressure	7.5±1.0 kgf/cm2a,	The result obtained is 7.5±0.5kgf/cm2a, which is within the specified limits.
Specific impulse	morethen 138sec	It was confirmed that specific impulse values ranging between 155 and 160sec were obtained that are above the standard.

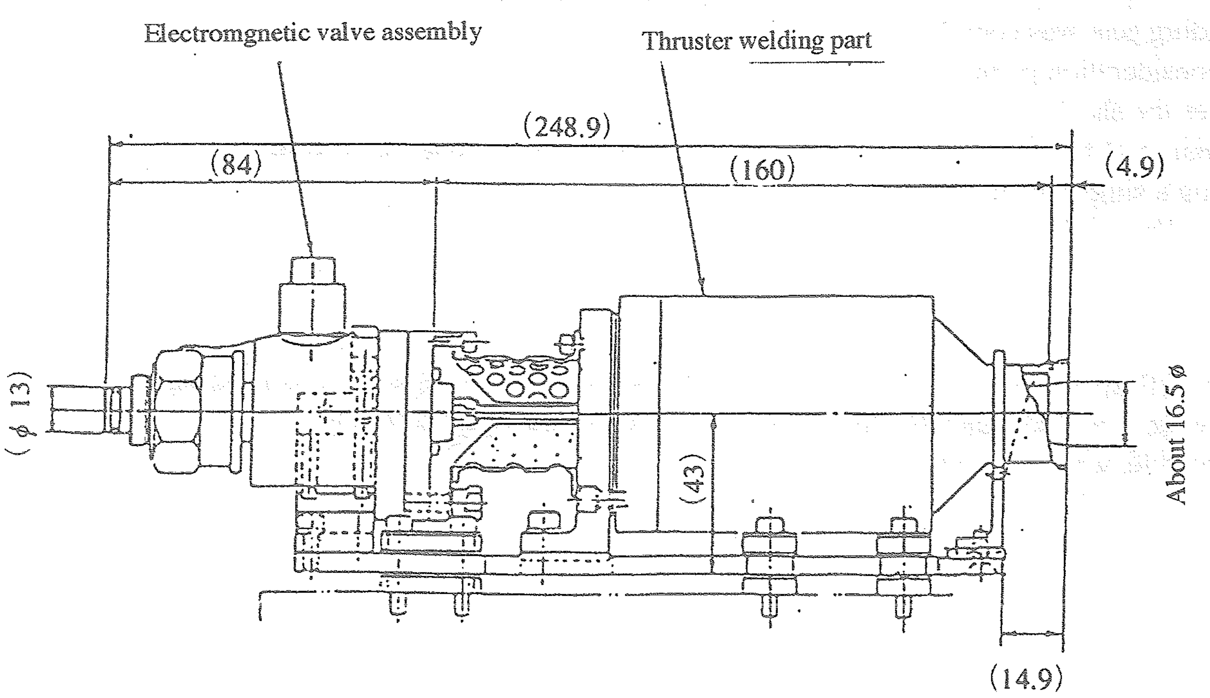
Performance of the propellant valve functions

It was confirmed that all the propellant valve functions met the requirements set by the standard. The evaluation for each of the performance items is shown in the following table:

No	Test designation	Test standard	Test result
1	Electric resistance test		
a	Insulation resistance test	Not less than 100 M ohm	Accepted
b	Coil resistance test	30±1.5 ohm (at 21°C)	Accepted
2	Pressure test	No deformation and leak	Accepted
3	Airtightness test		
a	Positive pressure inner leak	Leak rate GN^2 not more than 3.3cc/10min	Accepted
b	Positive pressure inner leak	GN^2 not more than 3.3cc/10min	Accepted
c	Negative pressure inner leak	GN^2 not more than 3.3cc/10min	Accepted
4	Current waveform measuring test	Valve opening response not more than 22±3ms	Accepted
		Valve closing response not more than 21±5ms	Accepted
5	Flow rate test	Not less than 37.95 for Valve Cv	Accepted



Thruster assembly before modification



Thruster assembly after modification

Fig.10.1-1 Thruster assembly

10.2 Engineering Model Fabrication and Testing of the Landing Gear Elements

10.2.1 Purpose

Conventional aircraft landing gears are lightweight and compact, and their performance is excellent; however, since they are expensive, it is desirable to use ready-made consumer use shock absorbing units as the landing gears for the experimental vehicle if they meet the minimum function and performance requirements. In addition, to determine the landing gear arrangement and landing gear load, data such as the load characteristics of the ground striking elements (hereinafter called pad) on unlevelled land becomes necessary.

For these reasons, a test landing gear (consisting of a shock absorbing section to be called a "shock absorber" in the following and a pad), which used a ready-made consumer shock absorbing unit, and a simulated contact surface was designed and fabricated. The shock absorber was subjected to a static characteristic test and a dynamic characteristic test (a dropping test) to judge whether it could be used as the landing gear for the experimental vehicle and to acquire data on characteristics of the pad striking unlevelled land.

10.2.2 Test Landing Gear

(1) Shock absorber

The investigation revealed that a shock absorber for off-road motorcycle use front suspension (the telescopic type) manufactured by Kawasaki Heavy Industries Co., Ltd. is the best fit that meets the system requirements for the shock absorbing gear for the experimental vehicle.

The test landing gear was completed by modifying the shock absorber as extensively as cost and schedule consideration permitted.

Fig.1 outlines the shock absorber.

The performance of the modified shock absorber was verified by carrying out simulation calculation using a simplified model.

Fig.2 outlines the results of the simulation.

(2) Pad

Pads of three different shapes and dimensions, which, with the system requirements taken into consideration, were considered suitable for use with the shock absorbing gear for the experimental vehicle, were designed and fabricated.

Fig.3 outlines the pad.

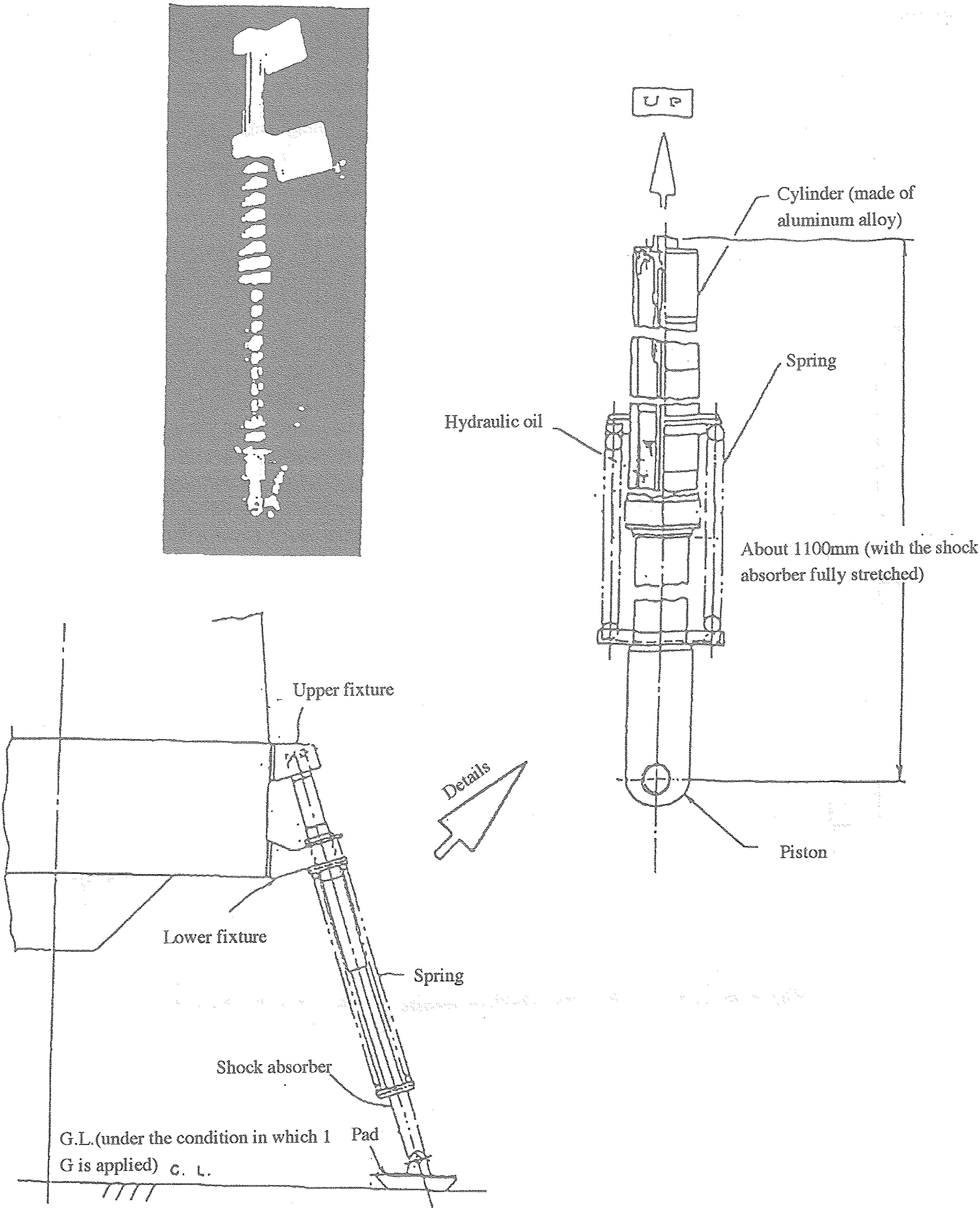


Fig.1 Schematic diagram of the shock absorber

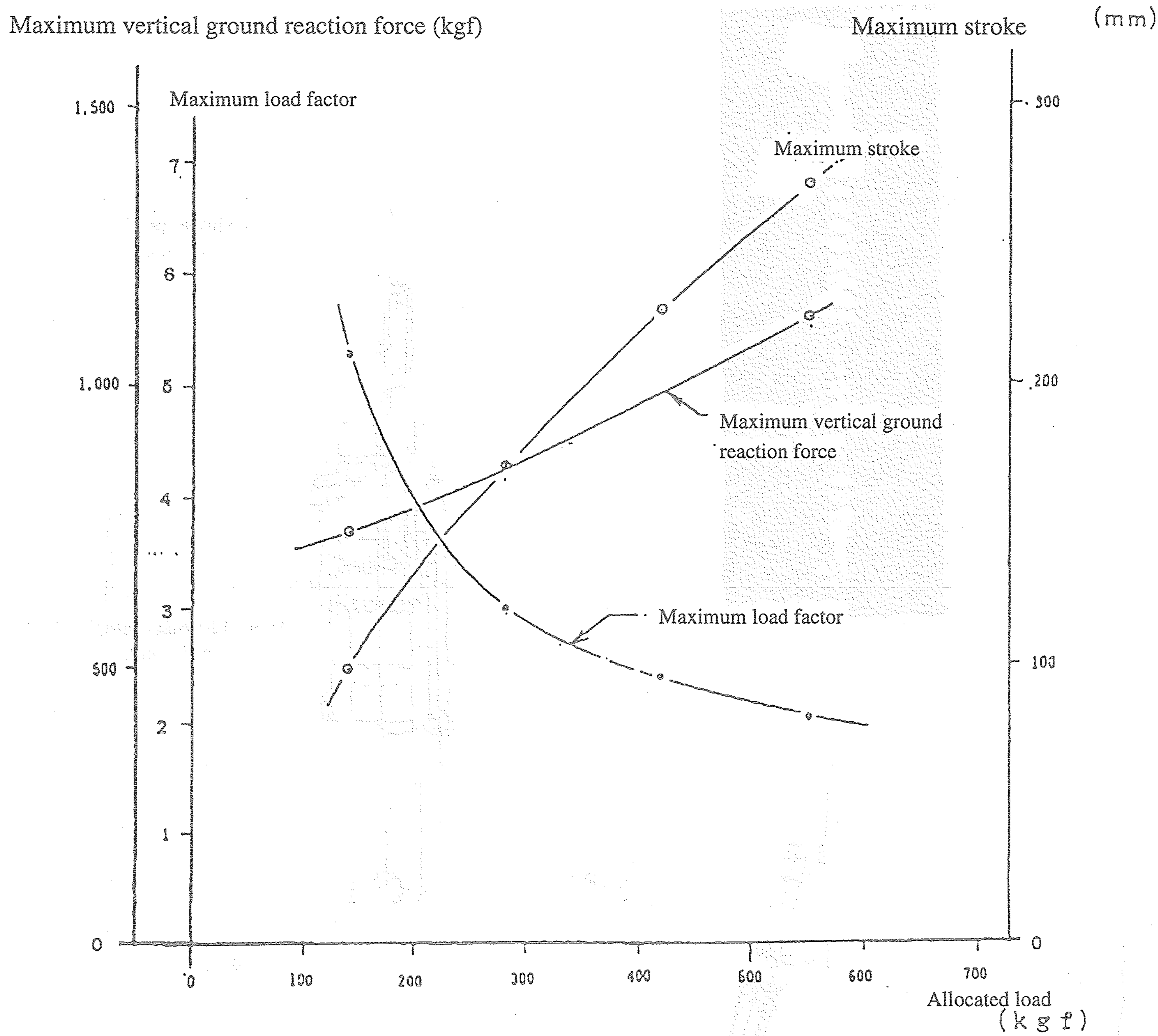


Fig.2 Results of the simulation made on the shock absorber

1. Shapes and dimensions: refer to the table below.
2. Material: steel
3. Other descriptions: All of Pads A, B, and C are coupled to the lower end of the shock absorber with a pin and rotatable.

Shapes and dimensions of the ground striking element

Item	Pad A	Pad B	Pad C
Diameter	Large	Medium	Small
Height	50	50	50
Tip angle	Small	Small	Large

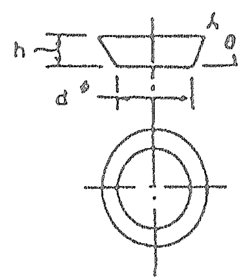
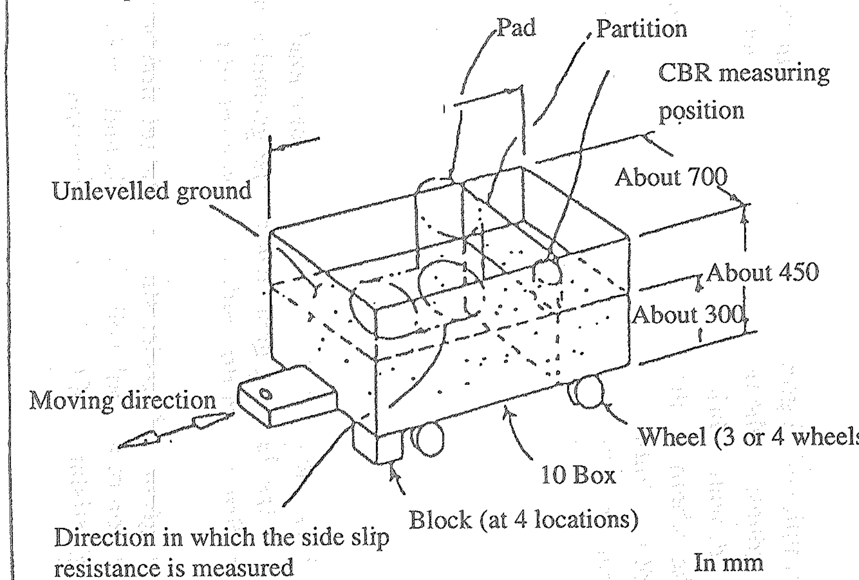


Fig.3 Schematic diagram of the ground striking element

1. Types
Four types: Unlevelled grounds I, II, and III and concrete surface
Whenever an experiment is carried out, it causes the surface condition to change, making it necessary to correct the surface condition after each experiment to measure the CBR values.

2. Hardness of the contact surface
   - Unlevelled ground I: small CBR value
   - Unlevelled ground II: medium CBR value
   - Unlevelled ground III: great CBR value

3. Shapes and dimensions



*1. The specified contact surfaces are fabricated by smoothing out the surface and then pressing it with a plate or the like evenly.
*2. Equivalent to JIS A 1222.

Fig.4 Schematic diagram of the simulated contact ground surface

(3) Simulated contact surface

Three different unlevelled ground specimens and a concrete ground specimen, which simulate the hardness of the unlevelled grounds on which the experimental vehicle is supposed to land, were fabricated.
Fig.4 shows the schematic diagram of the simulated grounds.
10.2 Results of the Static Characteristic Test
In the static characteristic test, the shock absorber was tested on the load-stroke static characteristic and the pad on the unlevelled ground characteristics.

10.2.3.1 Test Conditions

In the load-stroke static characteristic test, the static load-stroke characteristic was measured with the test landing gear striking the ground at a low velocity, and the measurement confirmed that the static characteristic of the test landing gear met the design objective.
In the unlevelled ground characteristic test, the relation between the ground striking load and the compression of the unlevelled grounds and that between the ground striking load and the side slip resistance of the pad were measured.
Table 1 shows the number of test cases, test conditions, and measurement items

10.2.3.2 Evaluation of the Results of the Test

(1) Load-stroke static characteristic test

As shown in Fig.5, the test landing gear exhibits characteristics very close to the design objectives which are acceptable.
In addition, the friction was not excessive.

Table 1 Number of test cases, test conditions, and measurement items in the static characteristic test

Test No.	Designation of test		Number of test cases	Test conditions			Measurement items
				Ground striking element	Simulated contact surface	Load	
1	Load-stroke static characteristic test		1	None	———	1 round trip	* Load versus stroke
2	Unlevelled ground characteristic test	Measurement of the compression of unlevelled grounds	* 1 9	3 different elements [Pad A Pad B Pad C]	3 different contact surfaces [Unlevelled Ground I Unlevelled Ground II Unlevelled Ground III]	Measured at intervals of about 1.000 while the ground striking load is increased gradually from 0 to 1,650.	* Ground striking load versus compression of unlevelled grounds * CBR
3		Measurement of side slip resistance	* 3 48	3 different elements [Pad A Pad B Pad C]	4 different contact surfaces [Unlevelled Ground I Unlevelled Ground II Unlevelled Ground III Concrete]	4 different values of load [200 600 1,100 1,650]	* Compression of unlevelled grounds * Ground striking load versus side slip resistance * CBR

Notes

*1. Number of tests=Pad (3 different pads) x Contact surface (3 different surfaces) x Load (single value) = 9 test cases

*2. Measured on the compression stroke only

*3. Number of tests=Pad (3 different pads) x Contact surface (4 different surfaces) x Load (4 values) = 48 test cases

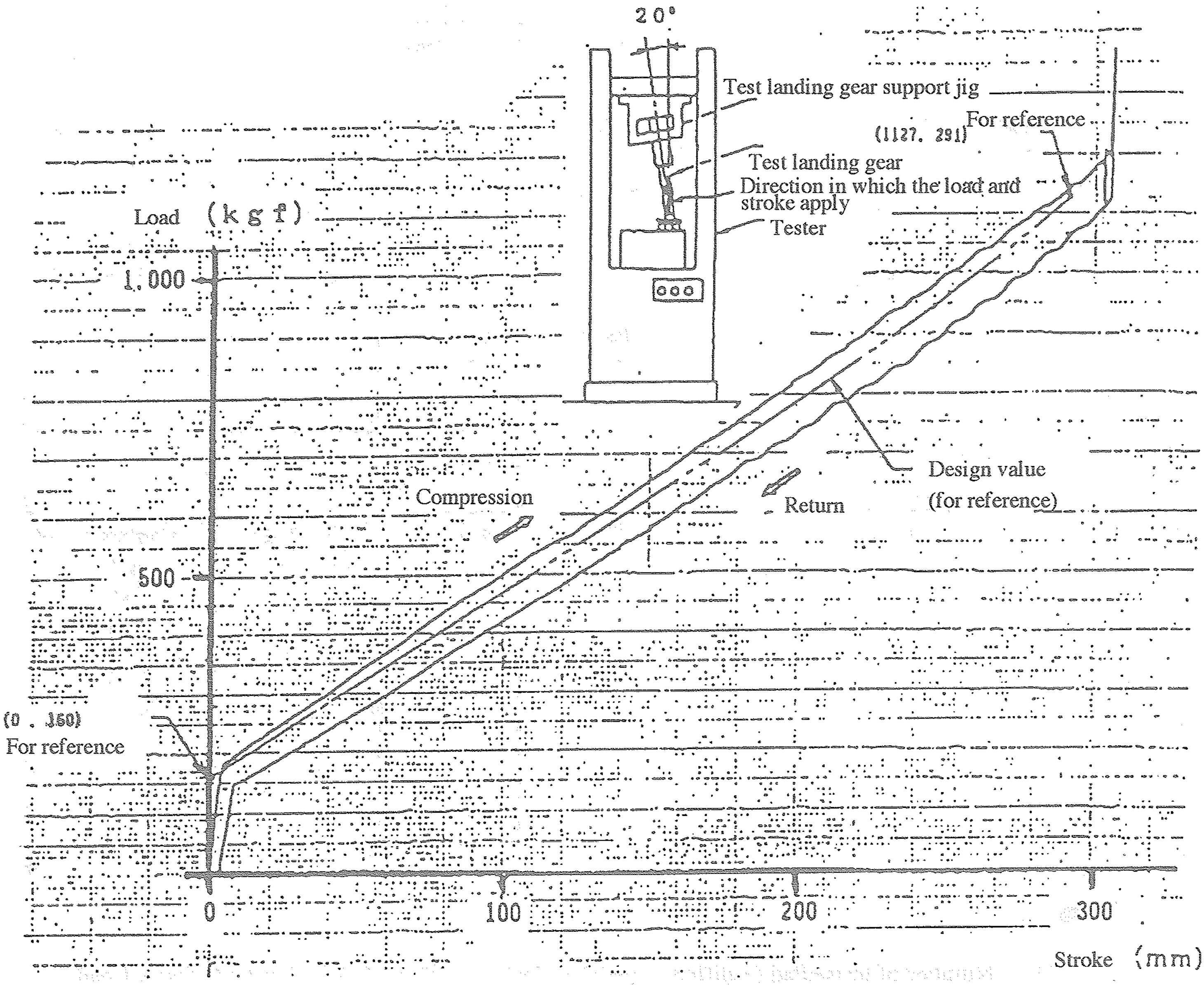


Fig.5 Results of the load-stroke static characteristic test

(2) Unlevelled ground characteristics

Under the test conditions shown in Table 1, data was acquired on the compression of unlevelled grounds due to the ground striking load for a total of 9 test cases and on side slip resistance due to the ground striking load for a total of 48 test cases.

Analysis of the data has yielded the relation shown in Figs.6 and 7.

These are valuable design reference materials which can be utilized for designing landing gears for the ground striking element of the experimental vehicle or lunar/planetary landing craft.

10.2.1.4 Results of the Dynamic Characteristics Test

The maximum stroke of the shock absorber, maximum vertical ground reaction force, and unlevelled ground compression occurring when the test shock absorbing gear was dropped on the 4 different simulated contact surfaces were measured against changes in the ground striking velocity and allocated weight. Using the data obtained through these tests, the energy absorbing capability and shock absorbing characteristics were examined to judge whether the test shock absorber could be used as the shock absorbing gear of the experimental vehicle.

10.2.1.4.1 Test Conditions

The test was conducted with the following conditions combined in various ways:

<1>	Allocated weight (kgf):	3 different values (450, 500, 550)
<2>	Ground striking velocity (m/s):	2 different values for weight allocations of 400kgf and 500kgf, and one value for a weight allocation of 550kgf (1.7-3.0)
<3>	Pad shape:	one type (Pad A) Pad A was selected on the basis of the results described in paragraph 3.4.1.3.
<4>	Simulated contact surface:	4 different surfaces (Unlevelled grounds I, II, and III and a concrete surface)

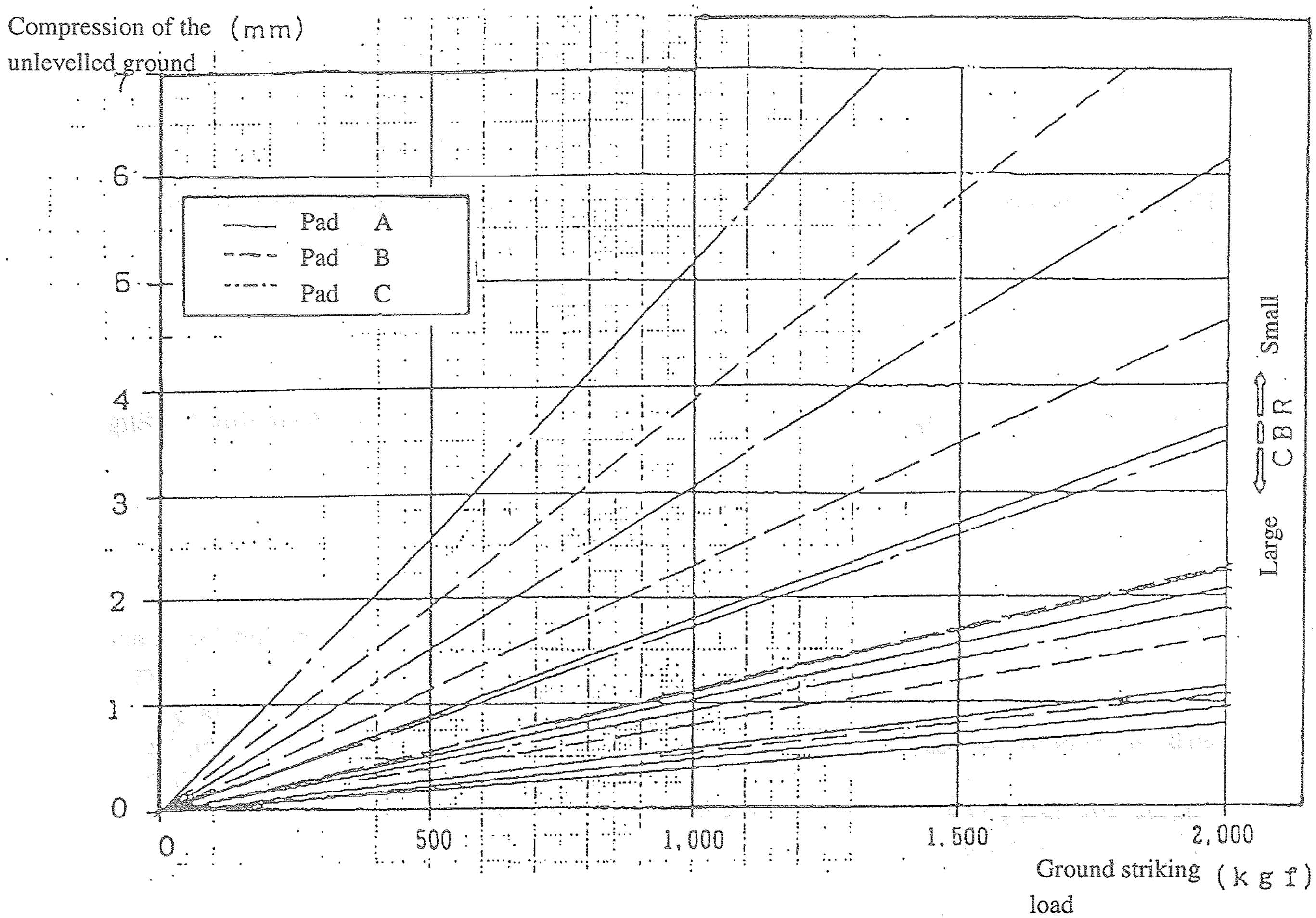


Fig.6 Ground striking load versus compression of the unlevelled ground

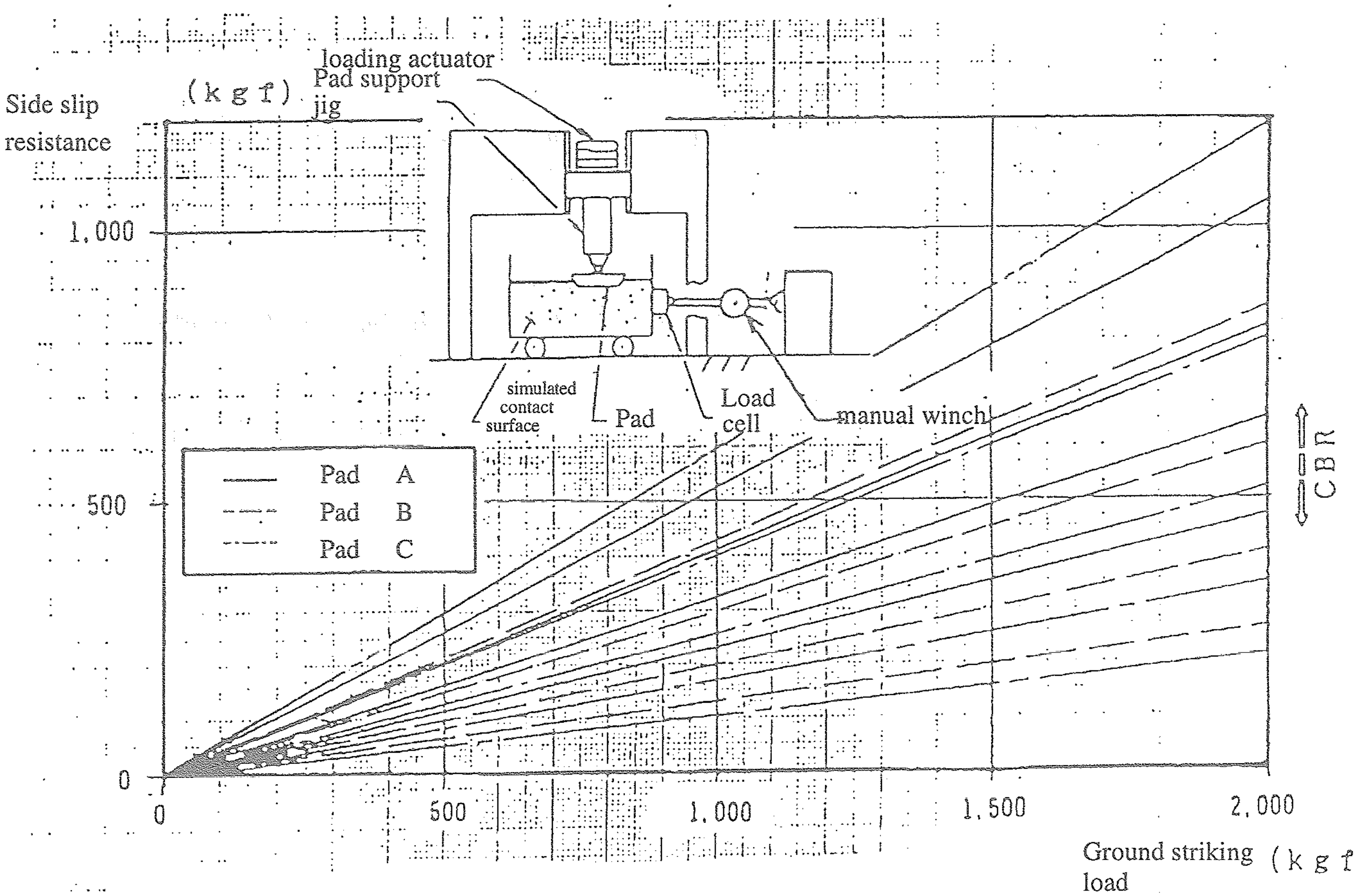


Fig.7 Ground striking load versus side slip resistance

10.2.1.4.2 Item to be Measured

<1> Relation between the ground reaction force and time
<2> Relation between the stroke and time
<3> Compression of unlevelled grounds
<4> CBR, allocated weight, and ground striking velocity

10.2.1.4.3 Evaluation of the Results of the Test

Fig.9 shows the relation between the ground striking velocity and maximum stroke of the test shock absorber, and Fig.9 the relation between the ground striking velocity and maximum vertical ground reaction force, both obtained from the data for the dropping test. (the typical data are shown in Fig.8.)

(1) Energy absorbing capability

The test shock absorber exhibited deformation at a weight allocation of 450kgf and a ground striking velocity of 3.0m/s; however, the dropping test data (refer to Fig.9.) and analysis of the simulation confirmed that the test shock absorber has a considerable bottom striking margin even at the maximum landing conditions of the experimental vehicle *1 (landing with a single landing gear at a landing gear equivalent weight allocation *2 =360kgf and a ground striking velocity of 3.0m/s), being sufficiently capable of absorbing energy.

---

*1. Landing on a single landing gear at the maximum landing weight (=550kgf) and maximum ground striking velocity (=3.0m/s)

*2 Although the rocket engine thrust equivalent to the self-weight (=550kgf) acts on the experimental vehicle at the time of landing, the thrust was not simulated because an existing dropping test unit was used for the dropping test.
Under this condition, the landing gear equivalent weight allocation equals 360kgf; on the basis of the capacity of the test unit, however, the minimum weight allocation was set at 450kgf, and tests were conducted further with weight allocations of 500kgf and 550kgf to be able to evaluate the effect of changes in the weight on the shock absorbing characteristics.

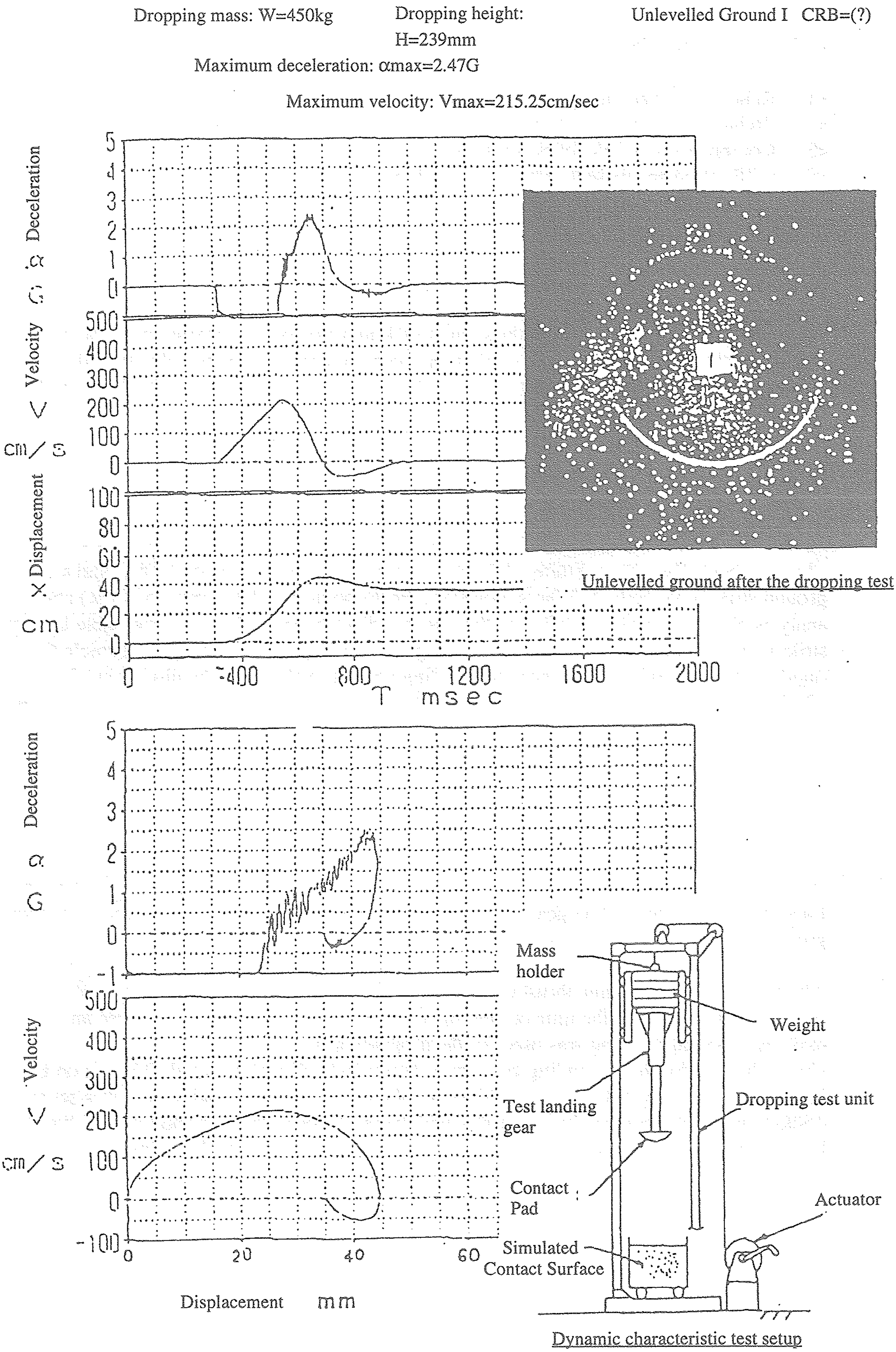


Fig.8 Example of the data obtained through the dropping test

Note)

*1 When the test shock absorber was dropped under this condition, it was deformed.

Symbol			
Concrete	(symbol)	450 kgf	○
Unlevelled Ground I	(symbol)	500 kgf	⊕
Unlevelled Ground II	(symbol)	550 kgf	⊖
Unlevelled Ground III	(symbol)		

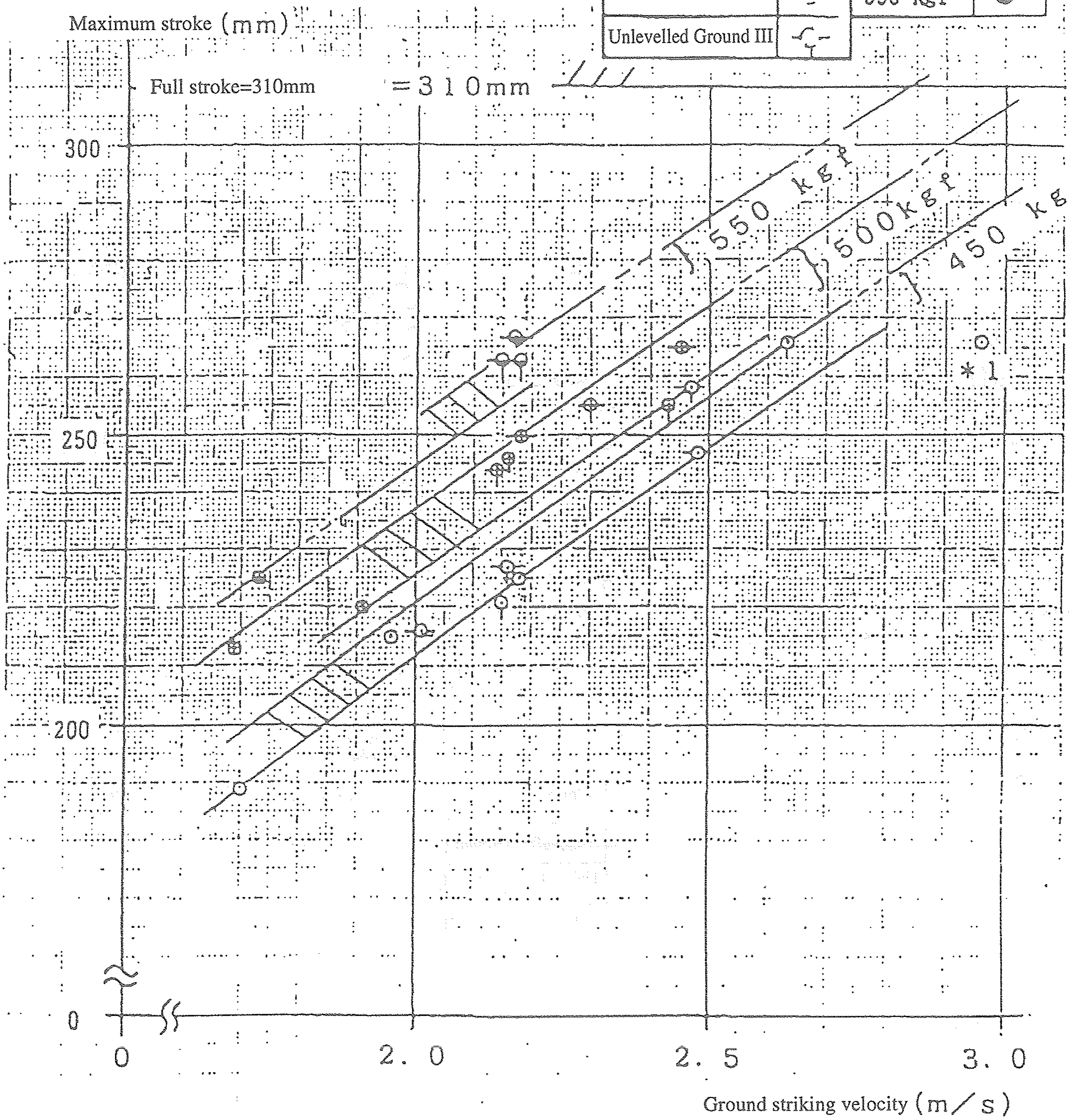


Fig.9 Ground striking velocity versus maximum stroke

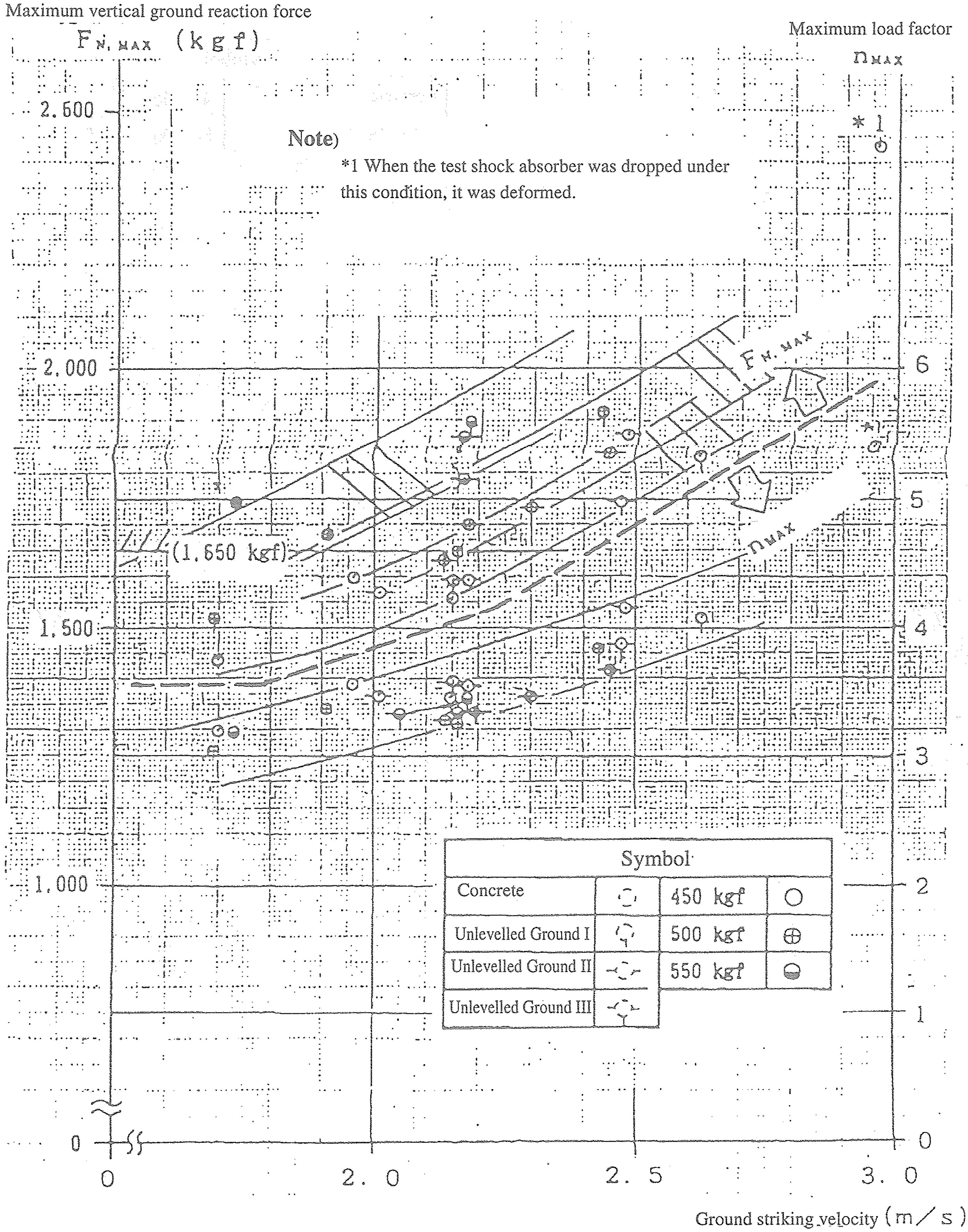


Fig.10 Ground striking velocity versus maximum vertical ground reaction force

(2) Maximum vertical ground reaction force

The test data and analysis of the simulations allows estimating the maximum vertical ground reaction force, under conditions of a landing gear equivalent weight allocation of 360kgf and a ground striking velocity of 3.0m/s, at 1,900kgf, which is above the target value of 1,650kgf (=550kgf x 3.0g).

With this shock absorber, reducing the spring force is most effective in reducing the maximum ground reaction force. On the basis of the test, it is safely estimated that, when the spring force is reduced to 60% of that of the one prototyped this time, the maximum vertical ground reaction force can be reduced to 1,500kgf with the maximum stroke of 280mm.

(3) Possibility of diversion of the test shock absorbing gear to the experimental vehicle

On the basis of the discussion made in paragraphs (1) and (2), it was confirmed that, subject to some design modification, the test shock absorbing gear meets the shock absorbing and strength requirements posed by the system; therefore, the test shock absorbing gear was judged to be fit for diversion to the experimental vehicle.

10.2.1.5 Future Problems

In order to develop a landing gear for the experimental vehicle on the basis of the results of the test of the prototype landing gear elements, the following efforts will be required:

(1) Optimized design of the oleo gear characteristics (reduction in the spring force), and the confirmation of the design by means of the dropping test;

(2) Improvement in the rigidity of the shock absorber;

(3) Reduction in weight;

(4) Mounting of thermal protection elements against engine plume

10.3 Designing, Prototyping, and Testing of the Wide Range Thrust Control Valve

10.3.1 Objective

In order to prepare for the wide range thrust control combustion test scheduled in the fiscal 1995, a thrust control valve (oxidizer and fuel) is designed, prototyped, and tested as an advance test to the "Takeoff and Landing Flight Experiment." To be more specific, the following basic characteristic parameters are acquired that are necessary to design cavitation Venturi valves *:

(1) Flow rate characteristic;

(2) Pressure characteristic; and

(3) Responsivity.

It is the objective of the designing, prototyping, and testing activity for the wide range thrust control valve to examine the design of the propulsion system and acquire data necessary to define the required specifications for the wide range thrust control valve on the basis of the basic characteristic data.

* As the wide range thrust control valve, a cavitation Venturi valve is adopted. The cavitation valve, which allows cavitation to occur in the throat, keeps the flow rate unaffected by the change in the pressure on the secondary side. By using this phenomenon, the cavitation Venturi valve makes it possible to control the flow rate stably without being affected by the combustion vibration which is likely to accompany deep throttling.

10.3.2 Examination of the Design

Table 10.3-1 shows the ratings for a wide range control valve. Three different pintles were prototyped to acquire data on the shapes of the flow rate control section. Figs. 10.3-1 and 10.3-2 show the overall shape and appearance, respectively.

10.3.3 Characteristics Confirming Test

(1) Objective of the Test

In order to confirm the characteristics of the prototype valve, water and the working fluids (NTO and N2H4) are made to flow through the valve. It is the primary object of the experiment to determine the flow rate controlling conditions and acquire design equations for designing a practical use valve by observing the valve cavitation that occurs during the experiment.

(2) Fig.10.3-3 shows the test flow.

In the water flow test, all of the 6 different prototype pintles (3 different pintles for 2 different fluids) were tested to determine the conditions in which cavitation occurs. On the basis of the results of the test, a pintle (No.2) that exhibited a flow rate close to the objective value was selected. The working fluid flow test was conducted on this single pintle to acquire basic data such as the working fluid flow rate characteristics.

Figs. 10.3-4 and 10.3-5 show the test configuration for the water flow test and working fluid flow test, respectively.

(3) Evaluation

(a) Flow rate characteristic

The flow rate characteristic of a cavitation Venturi valve is given by the following equation; when cavitation occurs, the flow rate is the function of the inlet pressure and opening only.

$$Q = C_D A \sqrt{2g\rho(P_I - P_V)}$$

where

Q: Flow rate;

C_D: flow coefficient;

A: Area of the passage;

P_I: Inlet pressure;

P_V: Vapor pressure.

The results of the flow rate test designed to determine the flow rate characteristic accompanying the occurrence of cavitation are shown in the following figures:

Fig.10.3-6 Relation between the opening and the flow coefficient, flow rate, and flow velocity (N2H4); and

Fig.10.3-7 Relation between the opening and the flow coefficient, flow rate, and flow velocity (NTO).

The flow coefficient of the wide range thrust control valve is almost constant independently of the opening; the results of the test are given below:

Flow coefficient C_D

	NTO	N2H4
Predicted value	0.85	0.85
Result obtained through the test	0.76	0.74
(Reference) Water flow test	0.81	0.81

The flow coefficient value listed above will be used for the future examination of the propulsion system.

(d) Pressure loss

On the basis of the results of the cavitation test, the relation between the differential pressure and flow rate is summarized in the following figures:

Fig.10.3-8 Relation between the differential pressure and flow rate (N2H4)

Fig.10.3-9 Relation between the differential pressure and flow rate (NTO)

A sufficient differential pressure causes the cavitation to occur, with the flow rate being the function of the upstream pressure only. In order for cavitation to occur, a differential pressure exceeding a certain threshold value is required, which is considered to be the minimum pressure loss in a cavitation Venturi valve.

In order to determine the pressure loss characteristic, the incipient cavitation cavitation number is plotted against the cavitation parameter that is given by the following equation:

Cavitation parameter

$$Kcv = \frac{\Delta P}{Pin - Pv}$$

where

Pin: Inlet pressure;

P_v: Vapor pressure; and

ΔP: Differential pressure

Using the following inequlity, the pressure loss that is required of the valve for cavitation to take place over the entire flow rate control range can be calculated from the incipient cavitation number:

$$\Delta P > Kcv(Pin - Pv)$$

The incipient cavitation coefficient plotted against the flow rate are shown in the following figures:

Fig.10.3-10 Incipient cavitation coefficient (water flow test, for N2H4)

Fig.10.3-11 Incipient cavitation coefficient (water flow test, for NTO)

Fig.10.3-12 Incipient cavitation coefficient (working fluid flow test, for N2H4)

Fig.10.3-13 Incipient cavitation coefficient (working fluid flow test, for NTO)

The graphs make clear the following characteristics of the incipient cavitation:

* The greater the valve opening is, the greater the incipient cavitation coefficient is;

* With the same valve opening, the smaller the flow rate is, the greater the incipient cavitation coefficient is. This is considered due to increase in the resistance coefficient associated with decrease in the flow rate under the condition of an identical shape; and

* Regarding the difference in the incipient cavitation coefficient due to that in the shape of the pintles in the water flow test, the smallest incipient cavitation number was obtained with No.3 pintle. This is considered due to the fact that the tip angle of No.3 pintle is the largest among the three different pintles.

* The incipient cavitation numbers are as follows:

Incipient cavitation coefficient

	Opening 100%	Opening 50%	Opening 25%
N2H4	0.27~0.33	0.17~0.24	0.11~0.24
NTO	0.40~0.43	0.35~0.36	0.26~0.28

The following table shows the comparison between the target specified values of the pressure loss (pressure at the valve inlet 20[kg/cm2]) and estimated values based on the incipient cavitation coefficient described above.

Pressure loss

	Opening	Specified target	Estimated value based on the results of the test
NTO	100%	2.8[kg / cm2]	8.2[kg / cm2]
	10%	8.8[kg / cm2]	5.3[kg / cm2]
N2H4	100%	1.0[kg / cm2]	6.6[kg / cm2]
	10%	7.0[kg / cm2]	4.8[kg / cm2]

For an opening of 100%, both NTO and N2H4 cannot meet the specified target; however, they meet the specified target sufficiently in the low flow rate domain. As a measure to be taken in the future, increasing the pintle tip angle may be contemplated; however, it is necessary to review the pressure characteristics of the propulsion system and revise the target specifications if required.

(3) Responsivity

It was confirmed that the delay time of the flow rate with respect to the pintle opening is in the range between 30 and 50msec regardless of the frequency and meets the target specification sufficiently.

10.3.4 Summary

In order to prepare for the wide range thrust control combustion test scheduled in the fiscal 1995, a thrust control valve (oxidizer and fuel) is designed, prototyped, and tested in an advance test to the "takeoff and landing flight experiment." Basic data on the fluid performance and dynamic mechanical performance was acquired from the results of the water flow test and working fluid flow test. The major results are as follows:

(1) Data on the flow coefficient at the occurrence of cavitation was acquired which will provide the design data for designing the flow path shape in the future;

(2) Basic data on the pressure loss such as incipient cavitation numbers was acquired which will provide basic data for designing the entire propulsion system in the future; and

(3) The test confirmed that the specified responsivity target (4.7[Hz]) was met.

10.3.5 Future Problems

The following are the future problems to be solved:

(1) Combustion test using the wide range thrust control valve
To demonstrate the effectiveness of a cavitation Venturi valve in an actual combustion test using the test valve modified for use in the combustion test;

(2) Required specifications for the valve
To define the specification requirements for the wide range thrust control valve with the overall pressure characteristics of the propulsion system taken into consideration;

(3) Valve design

Because the valve used for the present test was developed basically for acquiring data on the flow rate characteristics and the like, the development of space use valves requires considering the following characteristics additionally:

(a) Environment conditions: vibration environment, vacuum environment, and the like;

(b) Leakage: Use of metal bellows for sealed portions; and

(c) Reduction in weight.

Table 10.3-1 Required specifications versus design specifications

	Item			Required specifications	Design specifications		
1	Working fluid						
		Oxidizer		NTO (specific gravity: 1.449; vapor pressure:1.136kg/cm2)			
		Fuel		N2H4 (specific gravity: 1.009; vapor pressure:0.0185kg/cm2)			
2	Flow rate control range (kg/s)				No. 1	No. 2	No. 3
		Oxidant	100% thrust	0.70	0.4	0.7	1
			10% thrust	0.07	0.04	0.07	0.1
		Fuel	100% thrust	0.70	0.4	0.7	1
			10% thrust	0.07	0.04	0.07	0.1
3	Pressure loss						
		Oxidant	100% thrust	$2.8kg/cm^2G$ At the flow rate of 0.70(kg/s)	To be confirmed by the characteristics test		
			1 10% thrust	$8.8kg/cm^2G$ At the flow rate of 0.70(kg/s)	To be confirmed by the characteristics test		
		Fuel	100% thrust	$1.0kg/cm^2G$ At the flow rate of 0.70(kg/s)	To be confirmed by the characteristics test		
			10% thrust	$7.0kg/cm^2G$ At the flow rate of 0.70(kg/s)	To be confirmed by the characteristics test		
4	Rated pressure (primary pressure)			20 (kg/cm2G)	←		
5	Guaranteed pressure			30 (kg/cm2G)	←		
6	Pressure tightness			50 (kg/cm2G)	←		
7	Control ratio			10 : 1	To be confirmed by the characteristics test		
8	Responsivity			75ms (time required for the transition from 100% to 10% thrust)	4.7Hz (time required for the transition from 100% to 10% thrust)		
9	Power consumption			Max. 56w, DC 23-34V	AC servo motor (rating:30w)		
10	Weight			Not more than 3kg			
11	Port size				MS33649-8		

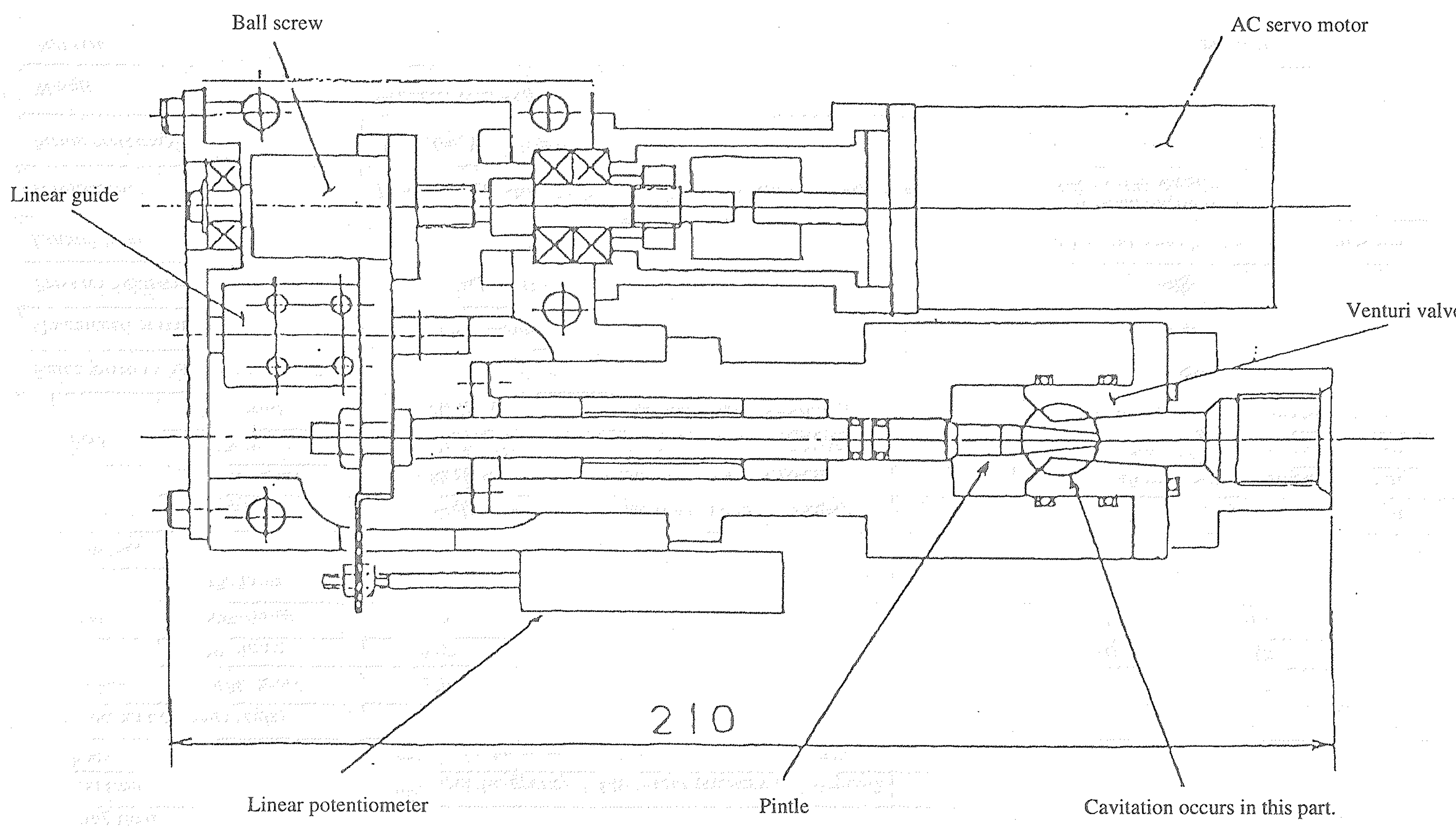


Fig.10.3-1 Overall shape of the wide range thrust control valve

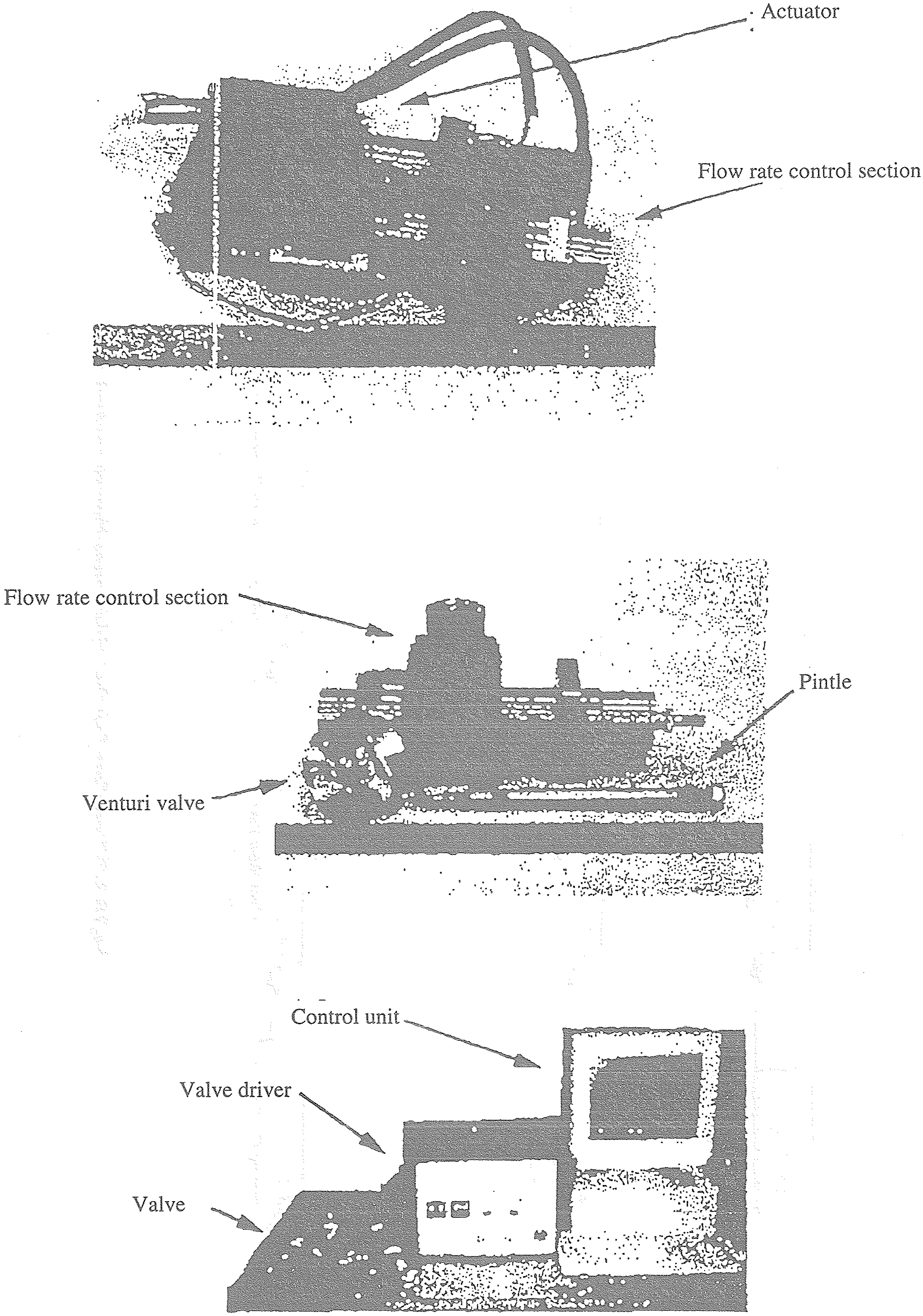


Fig.10.3-2 Appearance of the wide range thrust control valve system

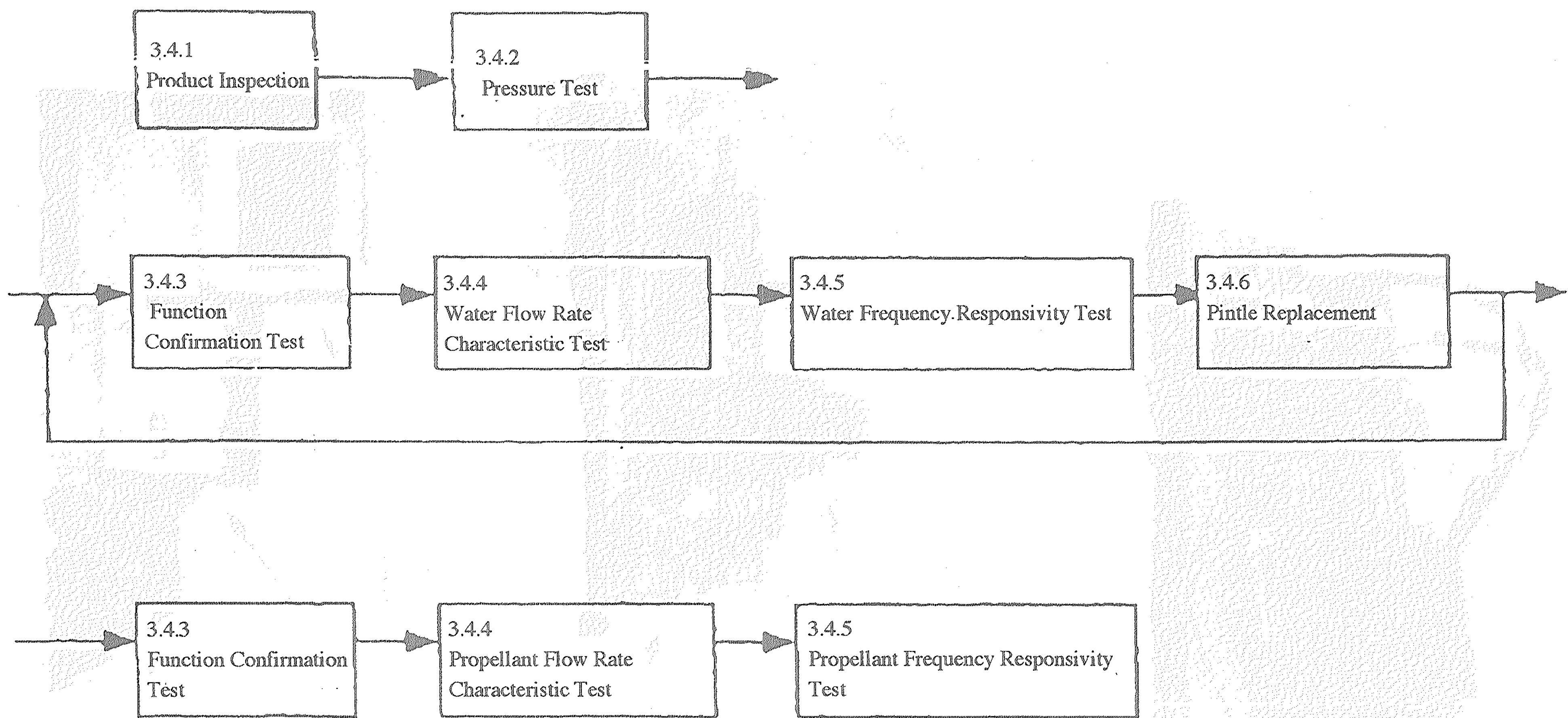
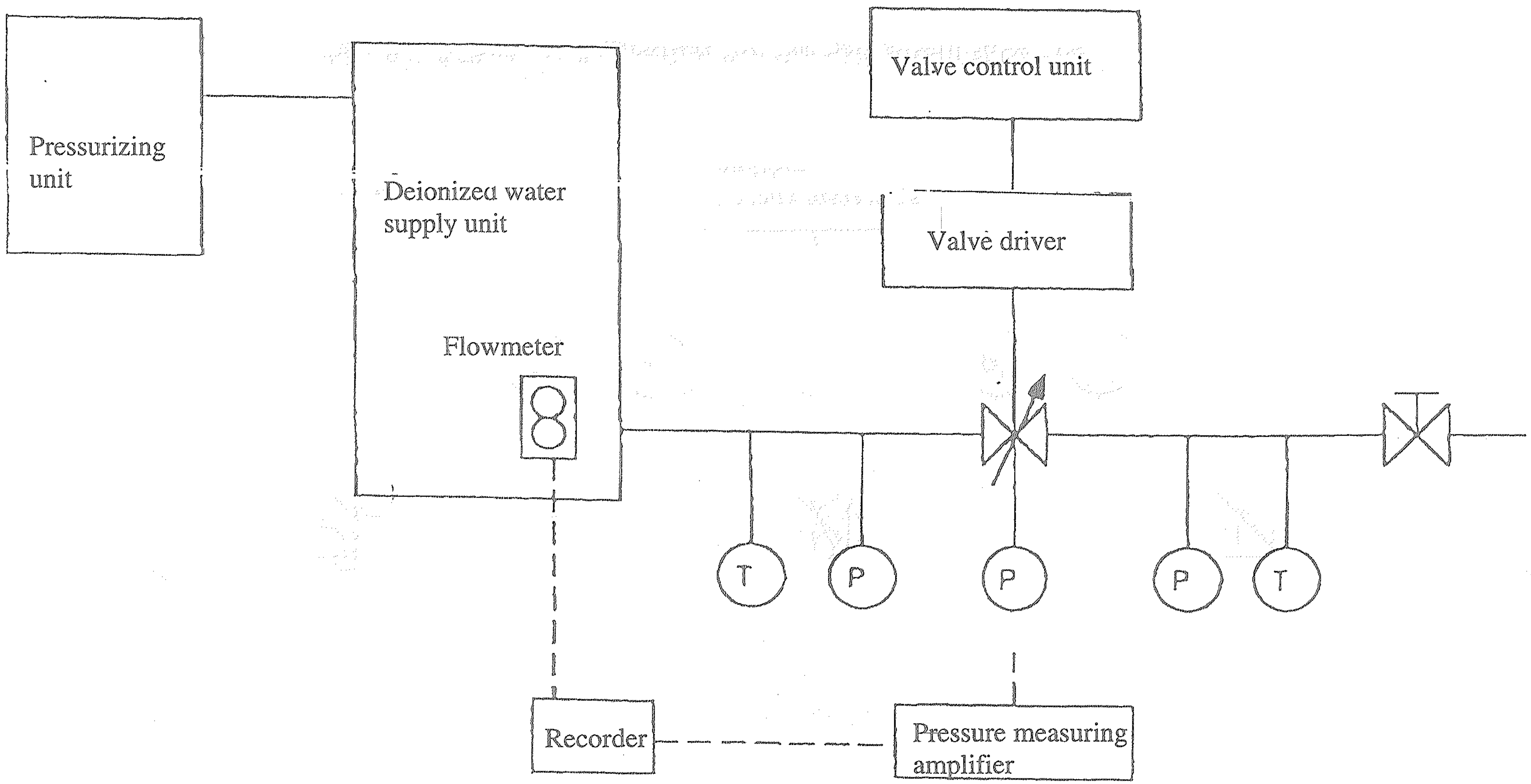


Fig.10.3-3 Test flow for the wide range thrust control valve



Figs. 10.3-4 Test configuration for the water flow test

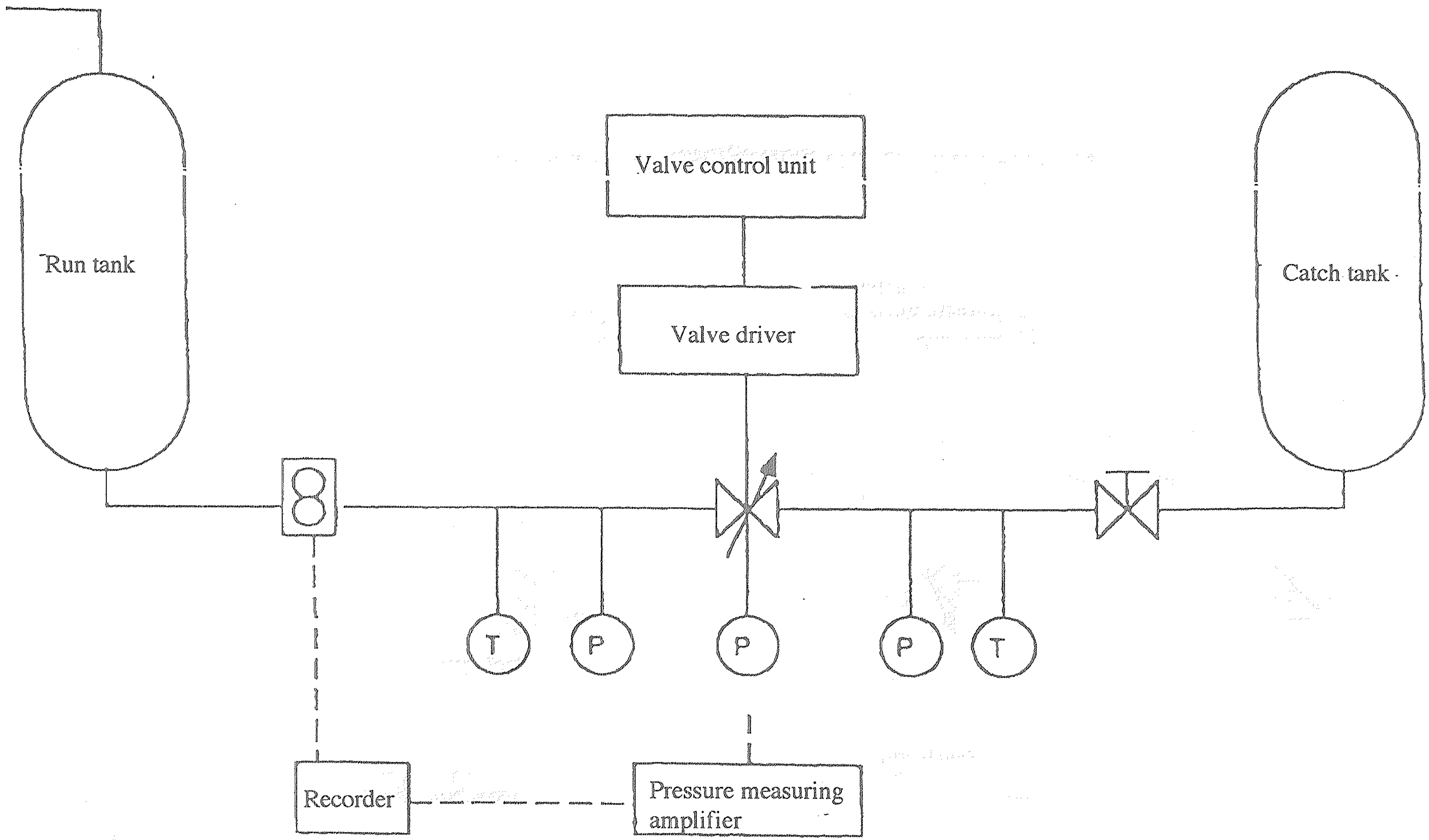


Fig. 10.3-5 Test configuration for the propellent flow test

N 2 H 4

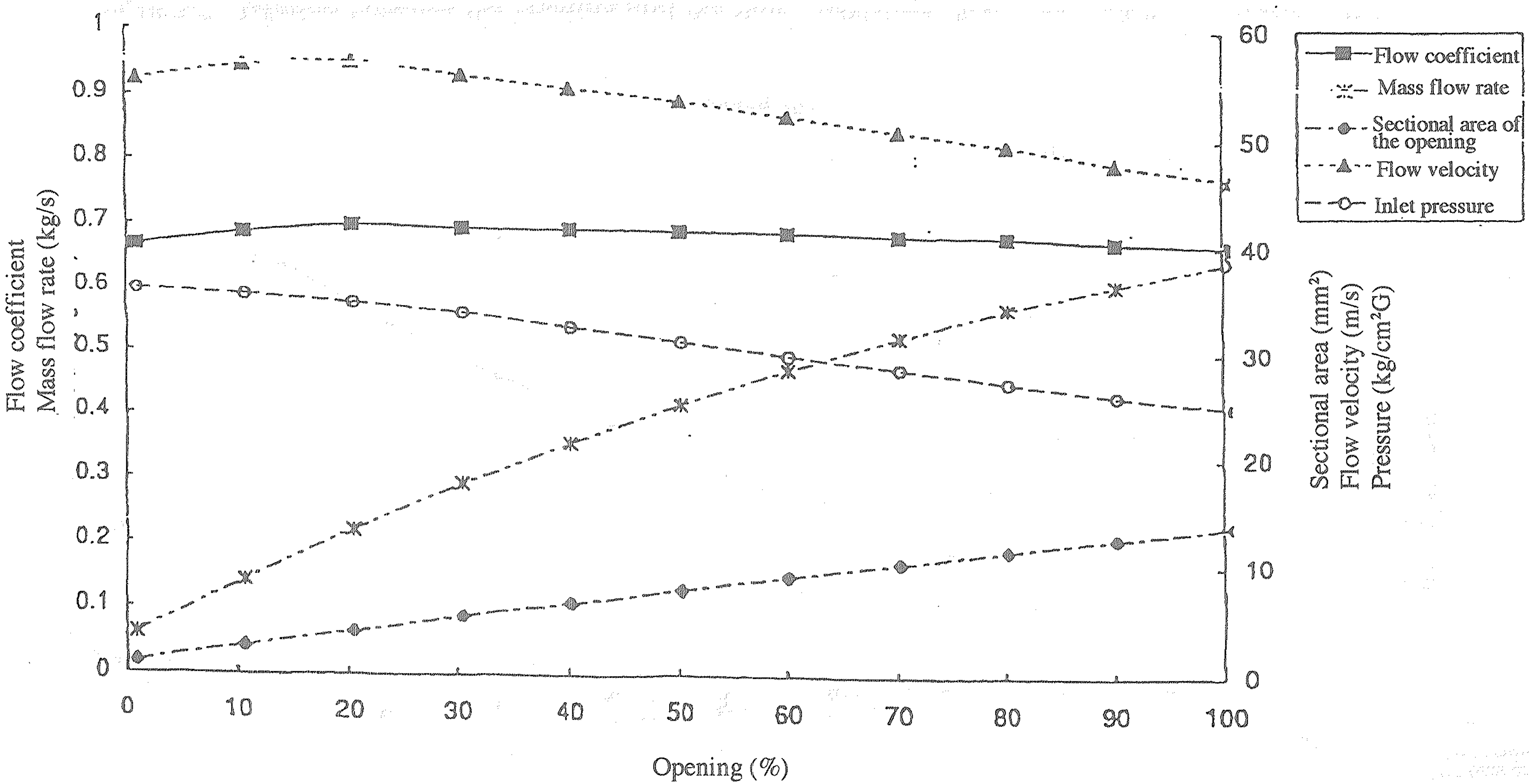


Fig.10.3-6 Relation between the opening and the flow coefficient, flow rate, and flow velocity (N2H4)

NTO

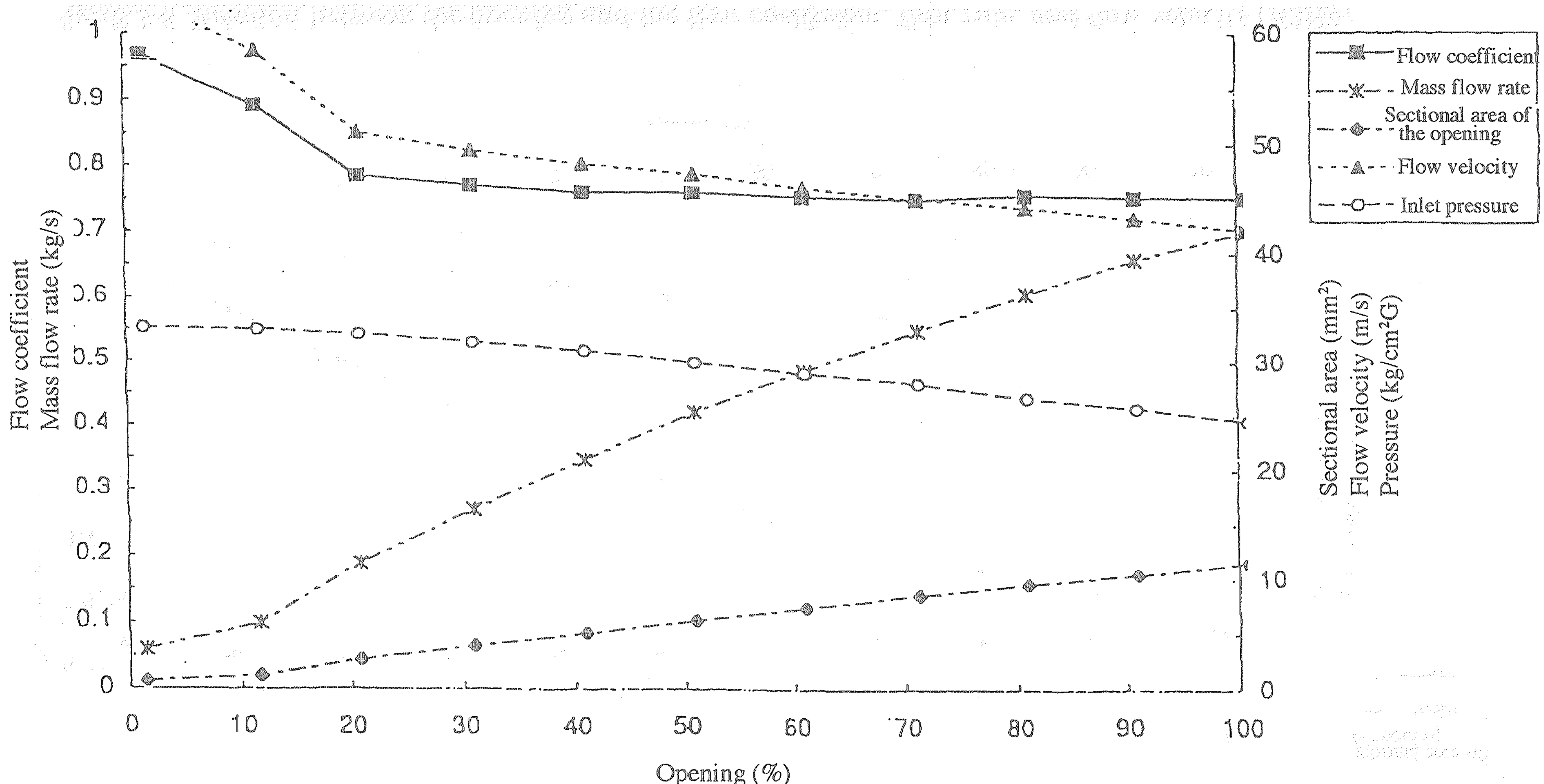


Fig.10.3-7 Relation between the opening and the flow coefficient, flow rate, and flow velocity (NTO).

Relation between the differential pressure and flow rate (N2H4)

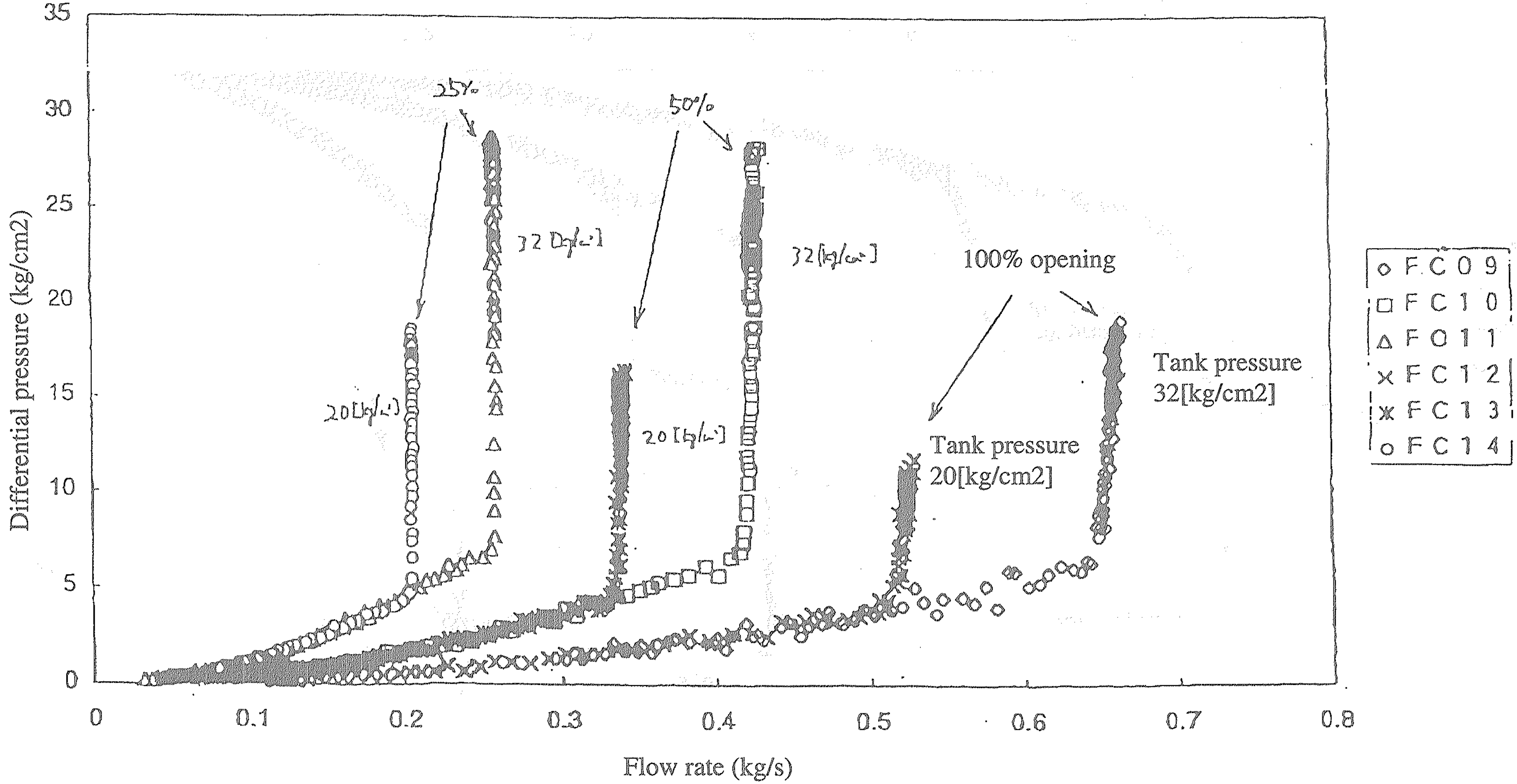


Fig.10.3-8 Relation between the differential pressure and flow rate (N2H4)

Relation between the differential pressure and flow rate (N2H4)

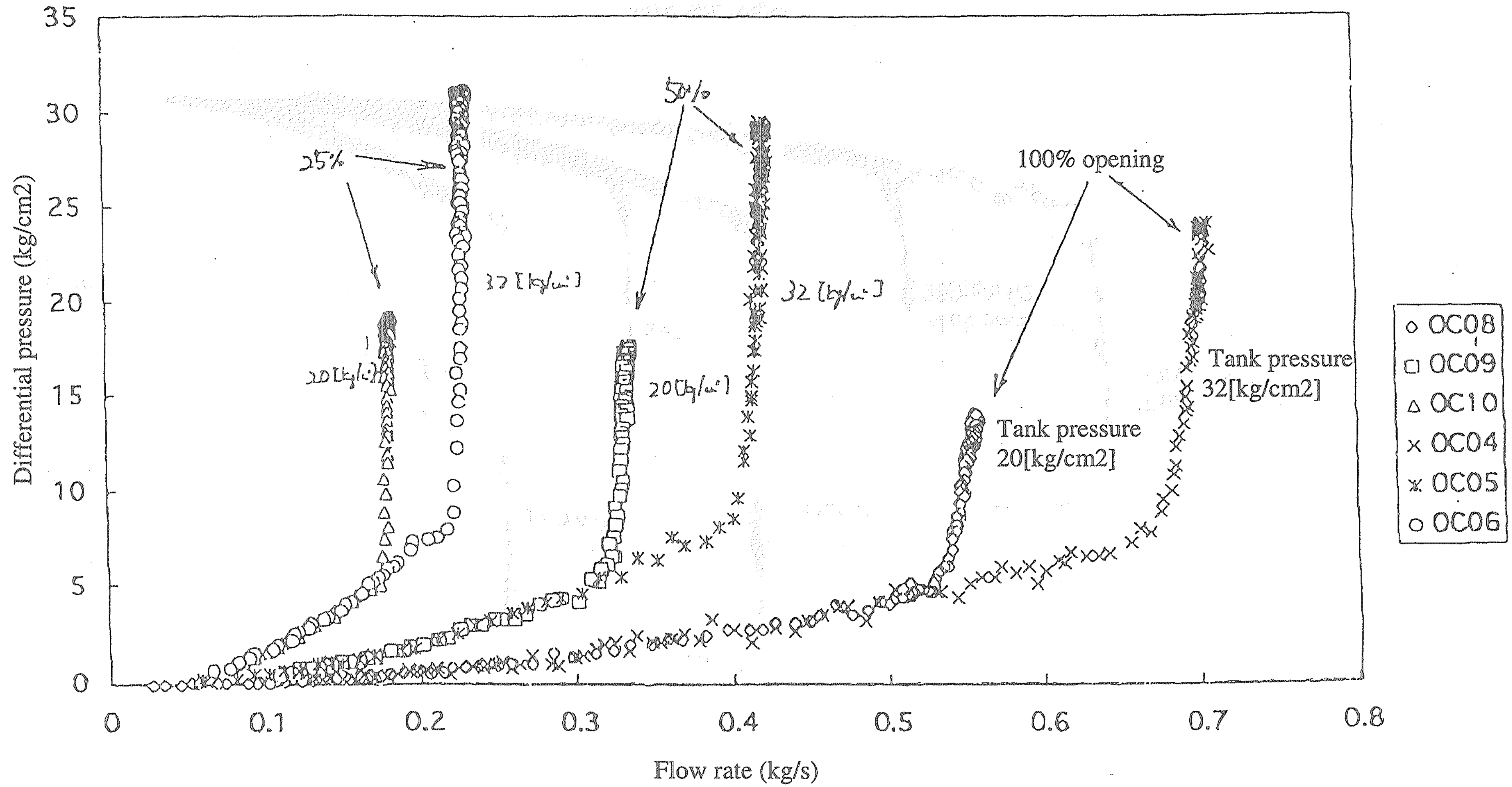


Fig.10.3-9 Relation between the differential pressure and flow rate (NTO)

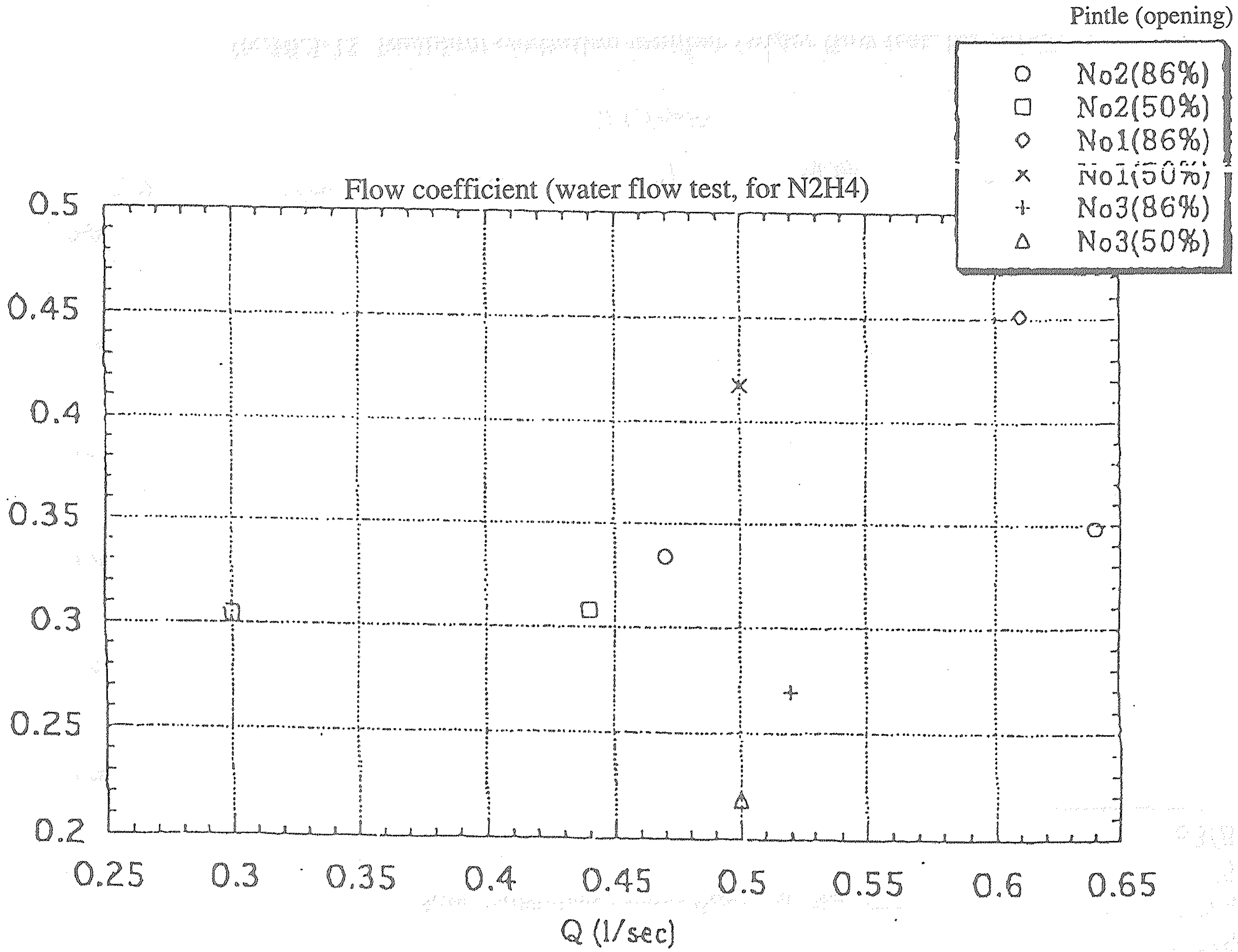


Fig.10.3-10 Incipient cavitation number (water flow test, for N2H4)

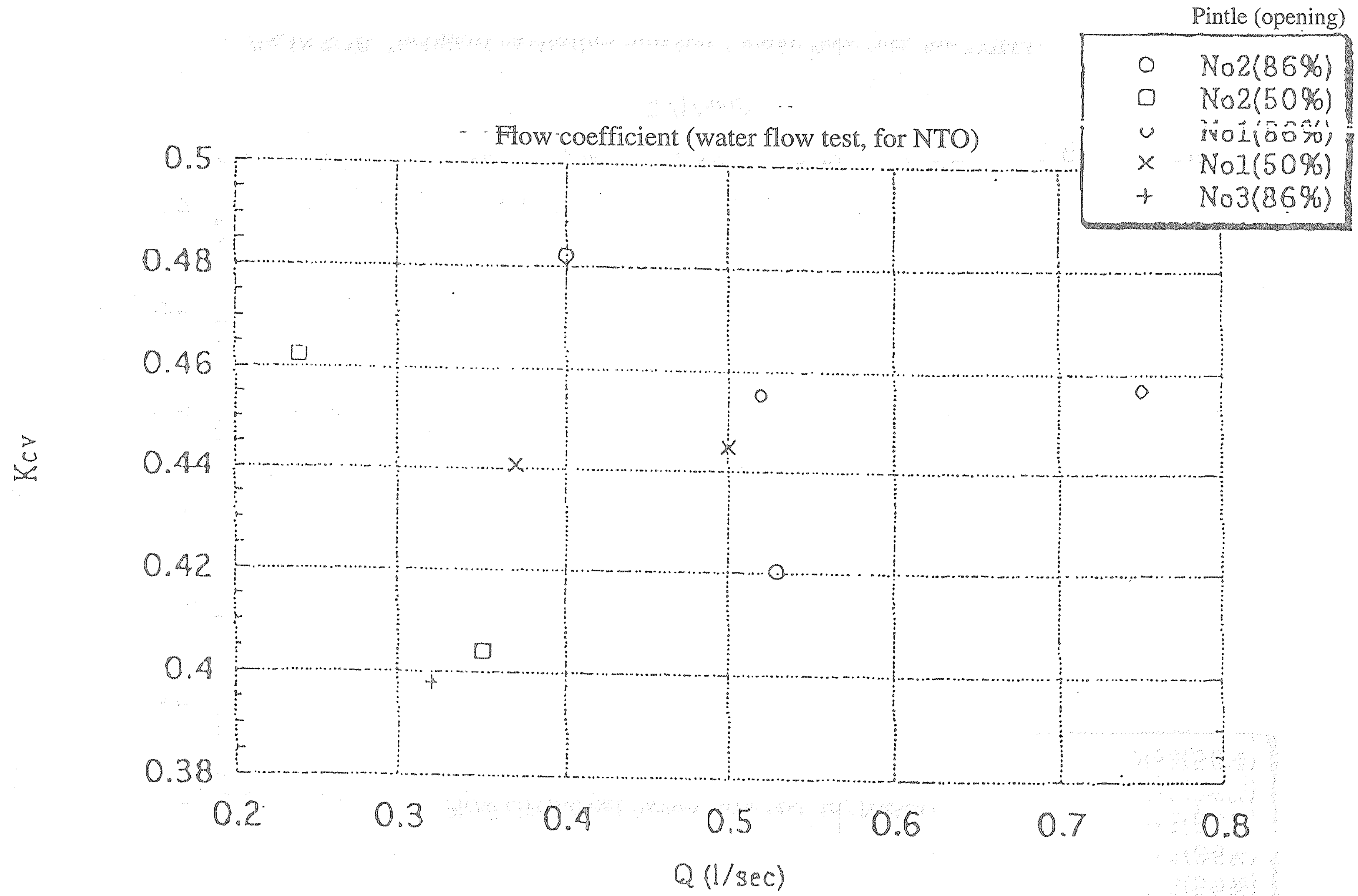


Fig.10.3-11 Incipient cavitation number (water flow test, for NTO)

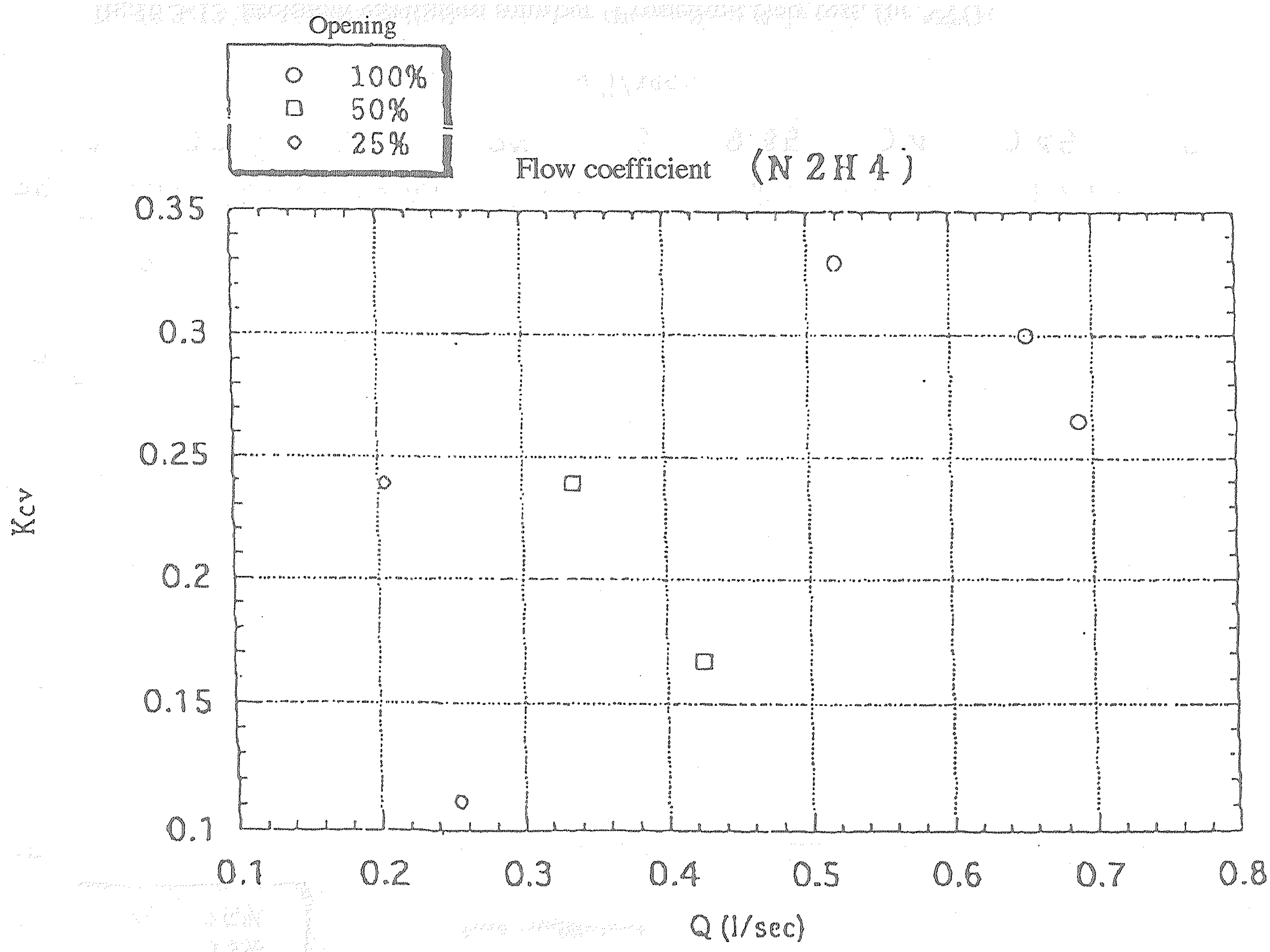


Fig.10.3-12 Incipient cavitation number (Propellant flow test, for N2H4)

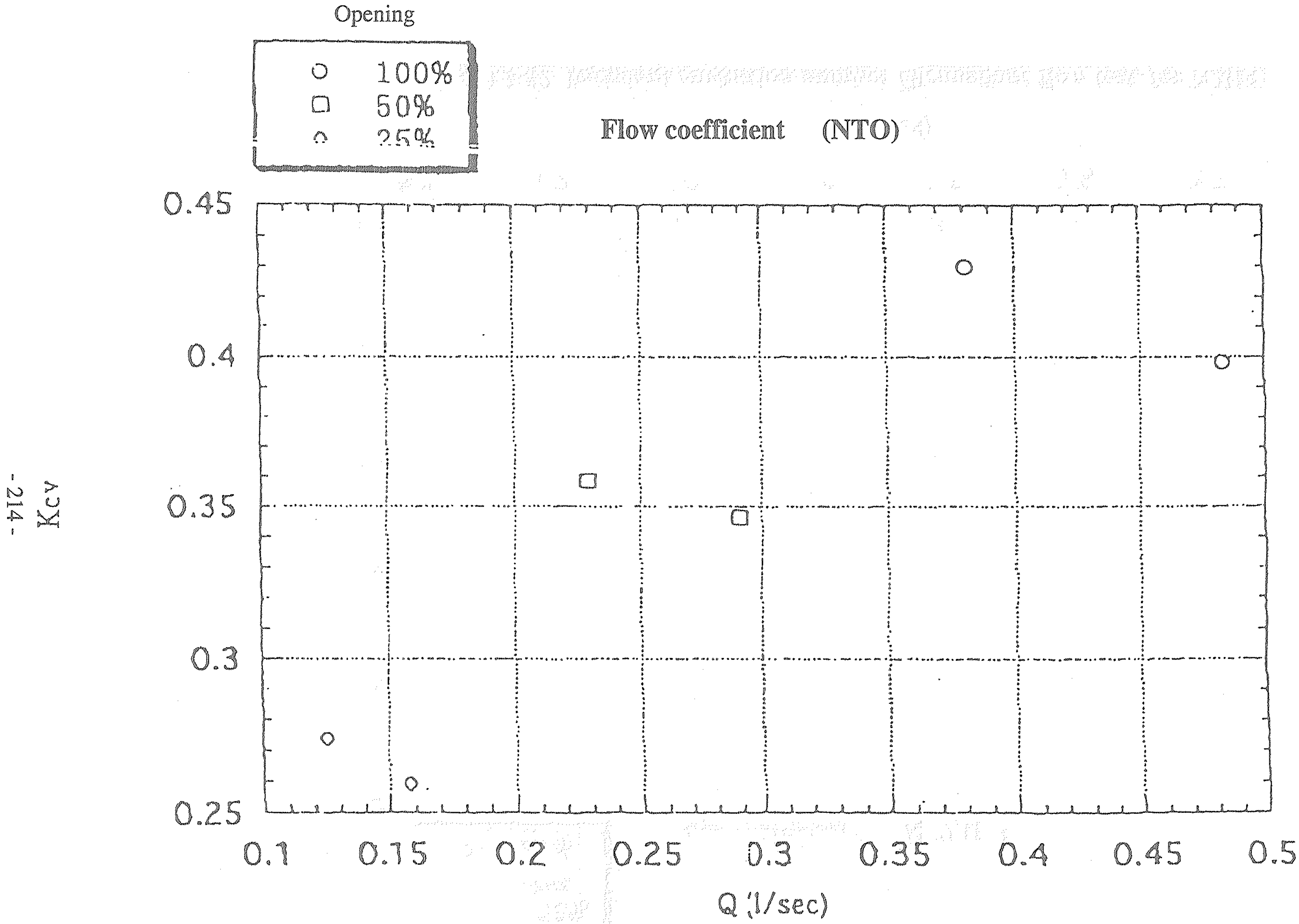


Fig.10.3-13 Incipient cavitation number (Propellant flow test, for NTO)

10.4 J-1 EVE Combustion Test (to be conducted by the engine group)

For the J-1 EVE combustor, 3 EM's (PO1, PO2, and PO3) have already been fabricated and 2 FM's are scheduled to be fabricated. The experimental vehicle is scheduled to be equipped with PO2 and PO3 EM's with the modified nozzle extension (with the opening ratio decreased from 2.3 to 2.0).
Figs.10.4-1 through 10.4-4 show the results of the test.
With the results of the combustion test taken into account, the thrust and Isp requirements for the combustor are revised as follows:

Design point	Pc=9.0[kgf], MR=1.0
Thrust	Not less than 278[kgf] at SL
Isp	Not less than 196[s]

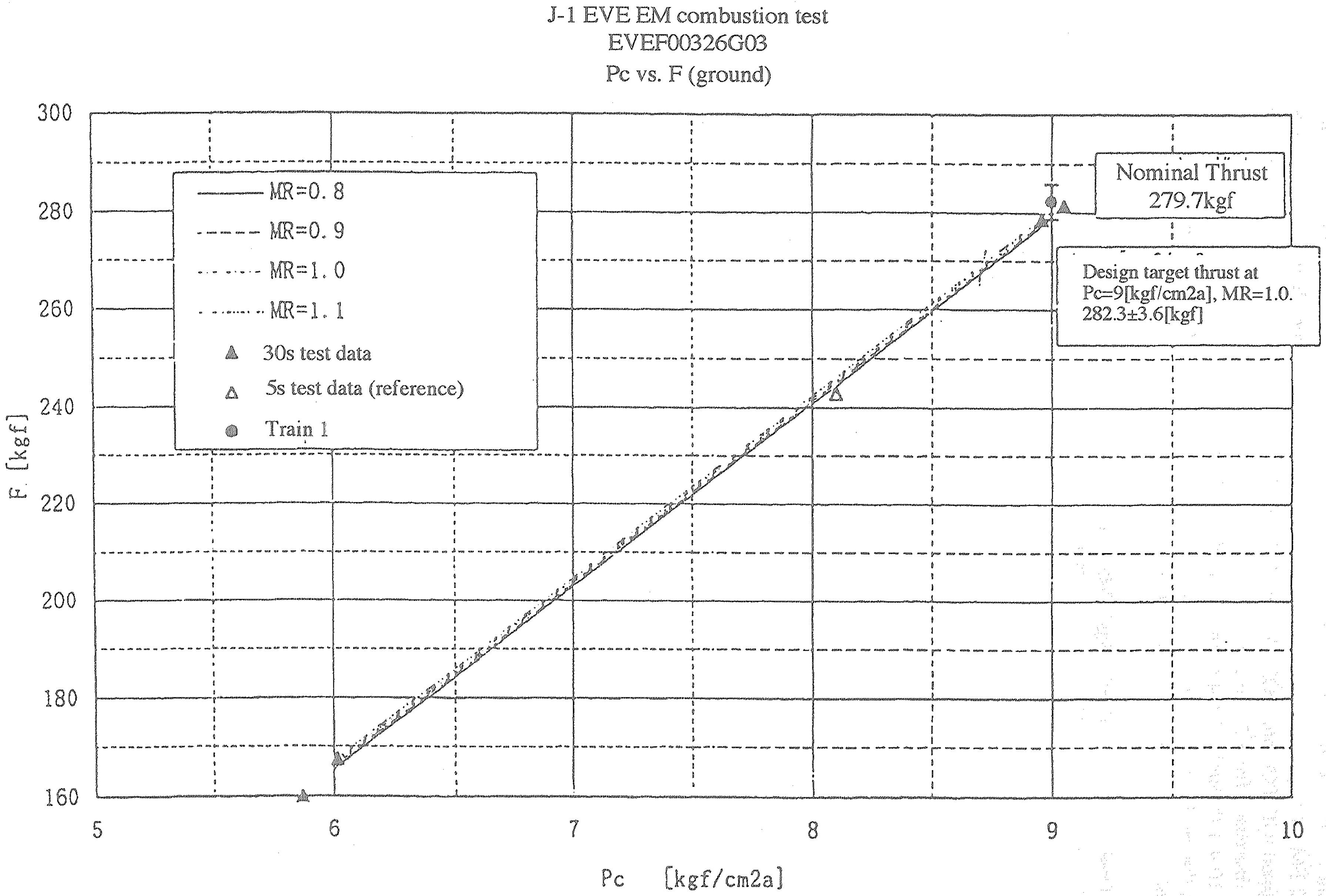


Fig.10.4-1 Combustion pressure versus thrust

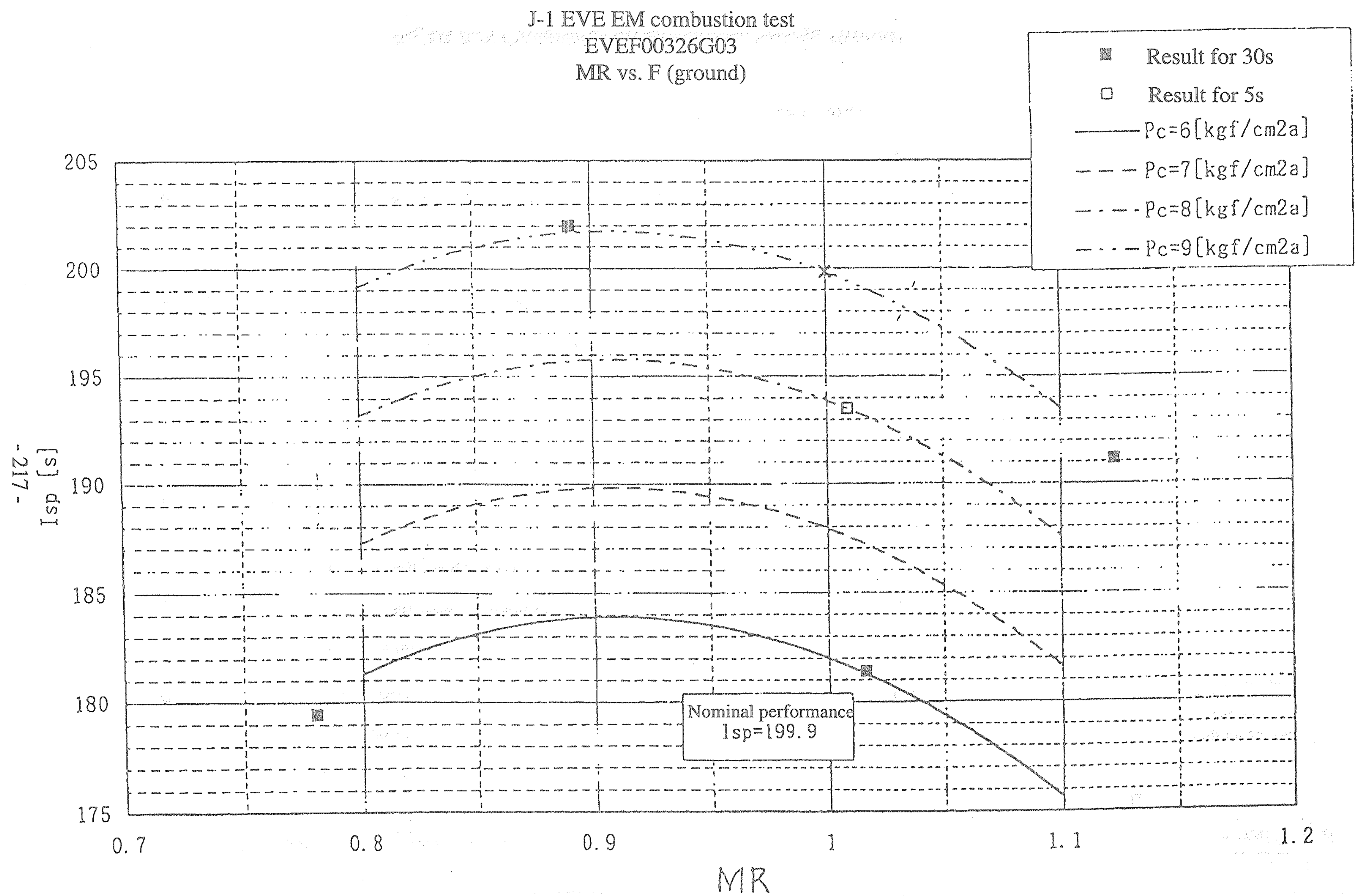


Fig.10.4-2 Mixture ratio versus Isp

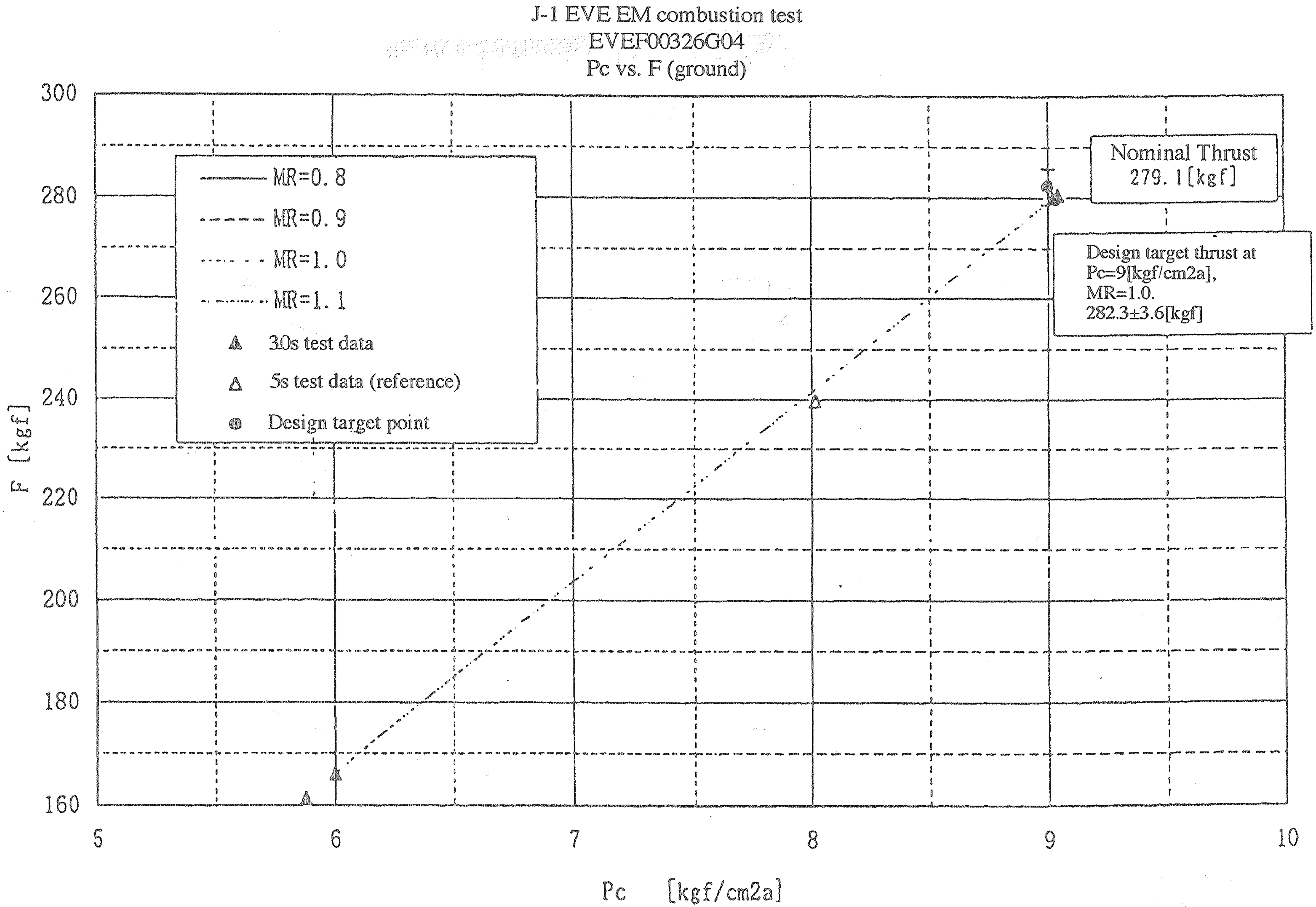


Fig.10.4-3 Combustion pressure versus thrust

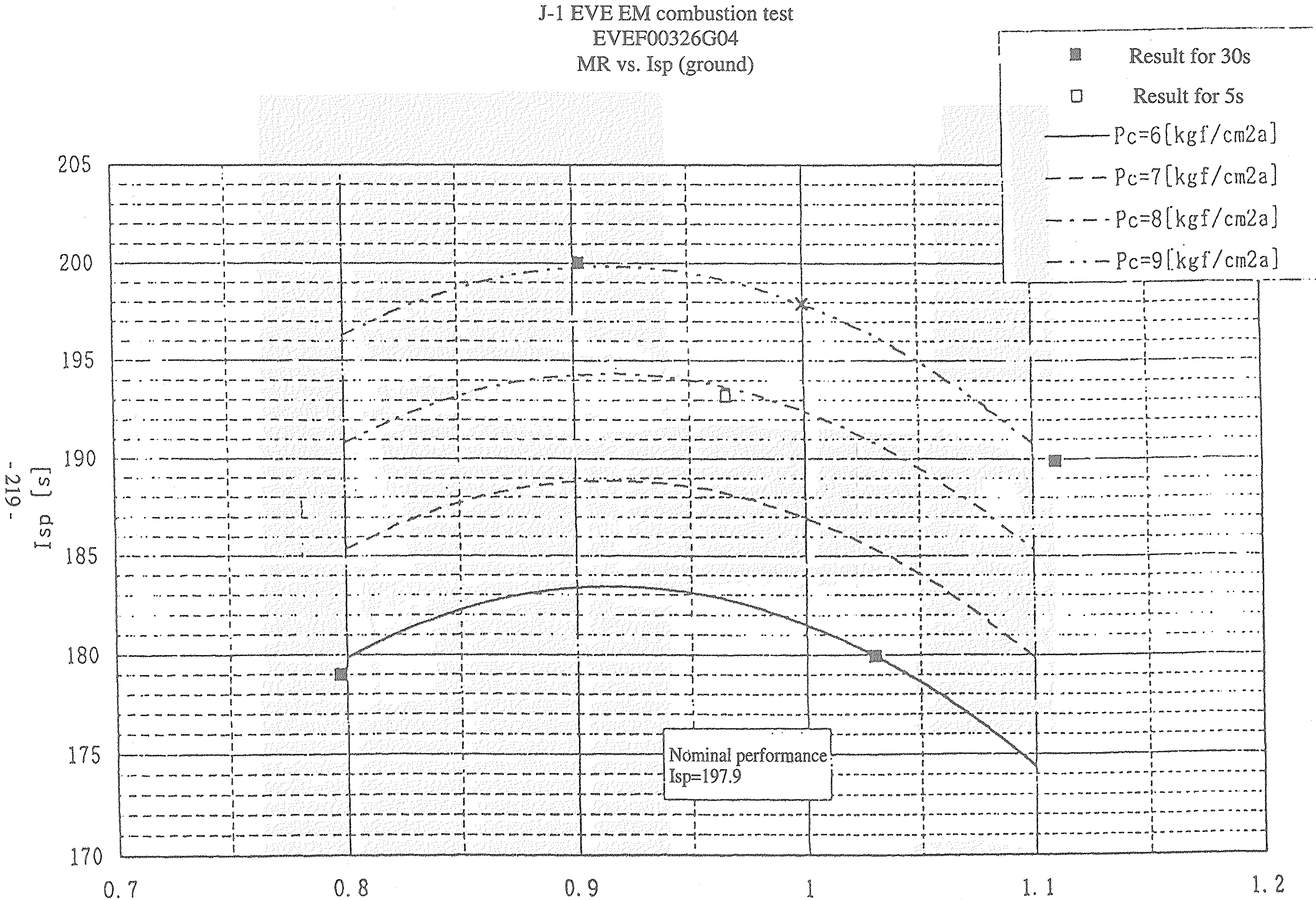
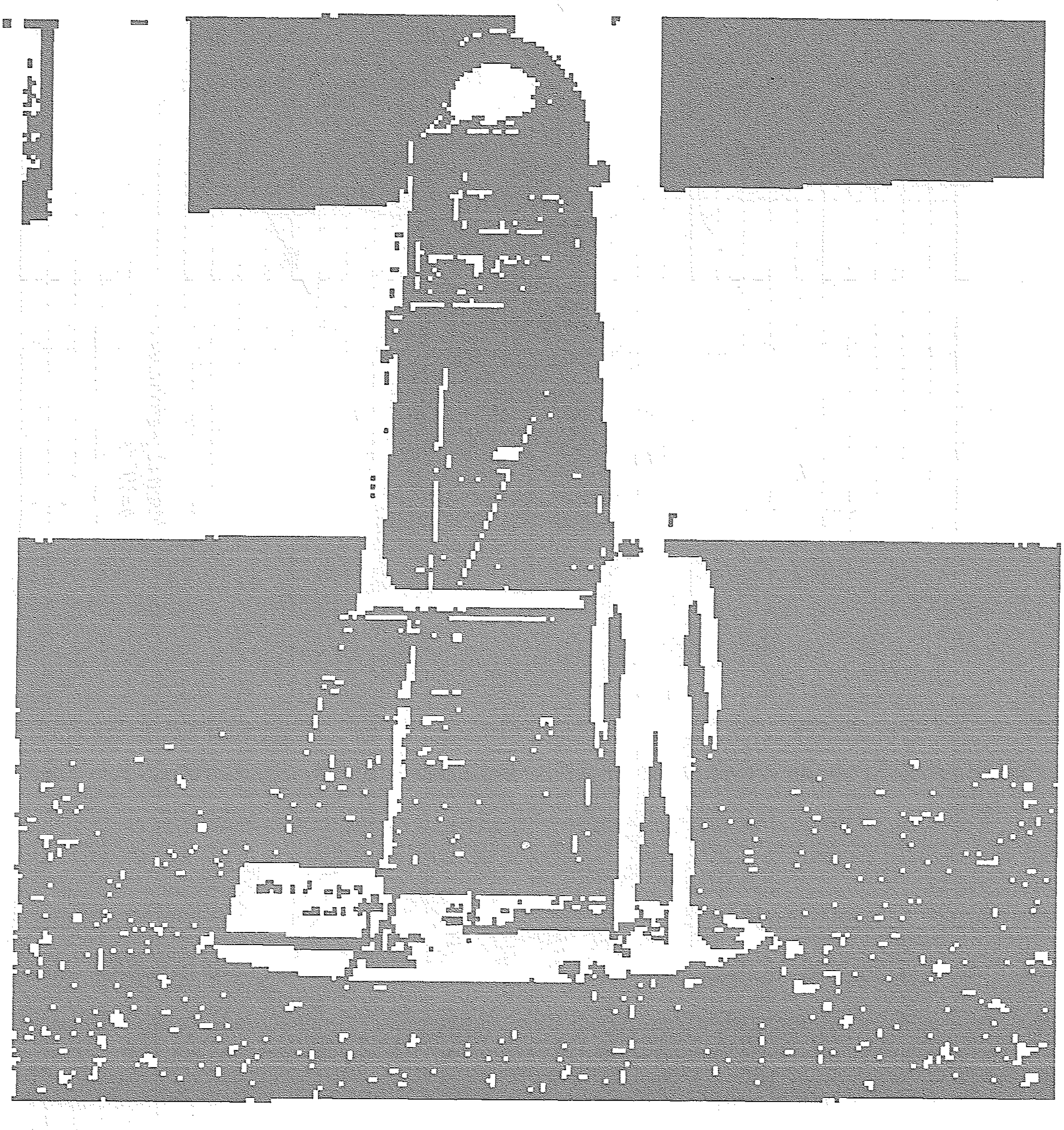


Fig.10.4-4 Mixture ratio versus Isp

11. Fabrication of a Model

A 1/7 scale model of the experimental vehicle, reflecting the results of the examination up to the previous section, was fabricated.
The dimension is 35[cm] in diameter x 155[cm].



Vertical takeoff and landing flight experimental vehicle

12. Summary

The Future Space System Laboratory was able to confirm the technological feasibility of the vertical takeoff and landing experimental vehicle during the current fiscal year. For the future study, it is necessary to continue, in more detail, examination and designing for the experimental vehicle, with emphasis placed on the engine thrust control, guidance and attitude control, and flight safety.

13. Associated Documents

The fiscal 1994 Space Transportation Symposium
"On the Concept of the Flight Experiment of Takeoff and Landing by Means of Rocket Engine Thrust Control" NASDA

Acknowledgment

This study is conducted under cooperation with Japanese heavy-industrial companies. I express my gratitude to many engineers who take part in this study, especially to Dr. Chikashi Motoyama, Dr. Koichi Yonemoto and Toshiyuki Yoshida of Kawasaki heavy-industries, LTD, Dr. Shigeyasu Iihara and Kotaro Shiina of Ishikawajima-Harima heavy-industries Co., Ltd.
Recently, Reusable Launch Vehicle, that can be recovered by vertical soft landing, is suggested. Experimental Landing for Moon survey is planned in Japan also. In fact, Vertical soft-landing technology using propulsion is almost only one way to arrive the surface of celestial body with no atmosphere. I suggest to verify this technology as soon as possible.

NASDA Technical Memorandum（NASDA-TMR-950012T）
Date of Issue : October 15, 1996
Edited and Published by :
National Space Development Agency of Japan
2-4-1, Hamamatsu-cho, Minato-ku,
Tokyo, 105-60 Japan

Inquiries and suggestions on the Report
should be addressed to :
Technical Information Division
External relations Department
2-4-1, Hamamatsu-cho, Minato-ku,
Tokyo, 105-60 Japan
FAX : +81-3-5402-6516
* This report is an English version of NASDA-TMR-950012

NASDA

NATIONAL SPACE DEVELOPMENT AGENCY OF JAPAN