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FEASIBILITY STUDY OF A SINGLE-STAGE RE-USABLE BALLISTIC BOOSTER SYSTEM

B E T A

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of a

SINGLE-STAGE RE-USABLE

BALLISTIC BOOSTER SYSTEM

BETA

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FEASIBILITY STUDY OF A SINGLE-STAGE RE-USABLE BALLISTIC BOOSTER SYSTEM

ABSTRACT: The present study investigates the feasibility of a single-stage reusable launch vehicle, which for technical and economic reasons represents the simplest solution to the earth-space transport problem. The results of the studies show that such a vehicle seems possible at the present state of the art, and that with such a device, using a launch weight of 130 tons, a payload of 2.7 to 4.2 tons can be inserted into a 200-km orbit.

INTRODUCTION AND RESULTS

1. INTRODUCTION /2*

The tendencies in the development of modern booster systems aim in two directions:

(1) Simplification of the functions and the operation
(2) Material reduction of costs per launch.

A single-stage reusable booster system appears to be an ideal solution for both objectives, i.e., a final solution to the earth-orbit transport problem.

This concept was first developed at the end of the 1950's by Professor Eugen Sänger, but was thoroughly investigated as a winged device, using the long-range bomber. Studies carried out during the years 1963-64 by the then Junkers (now MBB) under the guidance of Prof. Sänger showed, however, that the winged single-stage space transport with an airplane take-off system cannot be developed in a foreseeable time. The reason lies in the high structural and dynamic requirements which, combined with a relatively unfavorable form from the point of view of the volume/surface ratio, lead to high structural weight.

The relationships are different, however, in the case of a single-stage ballistic rocket: the construction is appreciably more compact, the requirements are less (short-duration penetration of the atmosphere) and the structural weight can be appreciably less than for winged devices.

In order to make a single-stage booster rocket possible, high engine performance is required; this requires:

(1) Use of hydrogen/oxygen as propellants,
(2) Use of high-pressure motors (150 atmospheres chamber pressure),
(3) Use of a central plug-nozzle to obtain a high expansion ratio.

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*Numbers in the margin indicate pagination in the foreign text.

Hydrogen technology is today demonstrably able to cope; high-pressure motors have reached more than 200 atm. chamber pressure (with H_2/O_2), and the plug-nozzle was successfully tested experimentally. /3

The use of a central plug-nozzle not only makes possible an increase in performance, but at the same time it can serve as a heat shield for reentry.

The single-stage ballistic device with LOX/LH_2 dates from the year 1962/63, when several American firms were investigating a successor for SATURN V (NOVA, NEXUS) (Fig. 1-1).

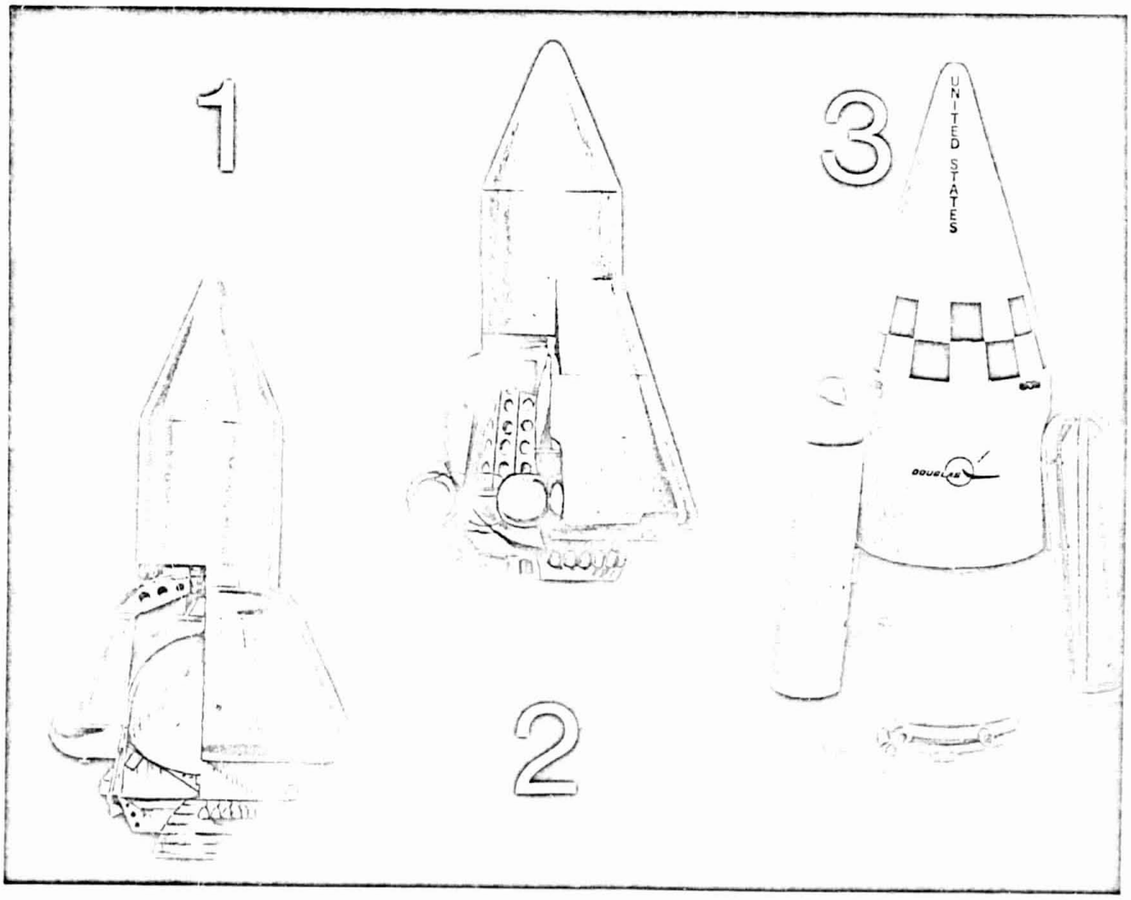

Figure 1-1: First U. S. Design of a Single-stage Booster System of the NOVA Class (500 Tons Payload), from the Year 1963. 1) NOVA by GD/A (One and a Half Stages), 2) NEXUS by GD/Astronautics (One Stage), 3) RHOMBUS by Douglas Co. (One Stage with LH_2 Tanks which can be Dropped).

The one-stage device was also investigated later by Douglas (Phil BONO) in a smaller version, as Project RHOMBUS, with droppable LH_2 tanks (one and /4 a half stages); and finally, as a single-stage device with only 100 tons launch weight, under the designation SASSTO (Single Stage to Orbit), in the year 1966.

The design shown in Fig. 1-2 already exhibits all the essential criteria of the BETA — Project investigated in this study; it provided the stimulus for carrying out the present investigations.

It was determined within the scope of this study that the design factor (structural weight) quoted for SASSTO is relatively optimistic, but is by no means decisive for the success of the project.

The results of this study can be summarized as follows:

(1) The development of a single-stage ballistic booster rocket is basically possible at the present stage of technology; there is no need to wait for appreciable improvements in technology or performance.

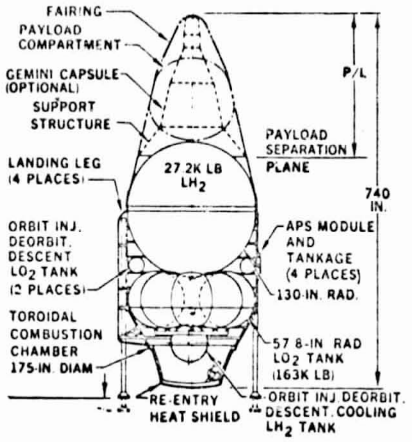

Figure 1-2. The SASSTO Project (Saturn Single-stage to Orbit) of Phil BONO, Douglas Astronautics, [1] 1967. /5

(2) The earth-orbit payload transport costs can be reduced at least by a factor of 5 with the BETA concept (in comparison with EUROPA II or other similarly wasteful devices).

(3) The payload (useful load) of the rocket is of course sensitive to variations of the net weight, but it is high enough to pose no risk.

Moreover, in the first stage of development the reentry capability can be waived, which increases the payload and reduces the cost of development (of course increases the construction costs, since more devices are required).

(4) The range of application of the single stage BETA design, which is planned for a low circular orbit, can be extended by one or two additional stages to include geostationary and interplanetary missions.

(5) The single-stage plan opens up completely new possibilities for launch and return, since it affords no danger from droppable parts (tanks or stages).

On this account it becomes possible to carry out launchings from Europe also.

(6) The single-stage ballistic booster rocket can be regarded as a final solution to the earth-orbit transport problem, because it combines simplicity of operation and low costs.

MISSION AND POWER REQUIREMENTS

2.1. BASIC PRINCIPLES

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The power requirement must be considered first, as the most important design parameter. On this value depends the design of the rocket, whose performance must subsequently be confirmed by trajectory calculations.

The power requirement is subdivided into five parts, for which either exact computations are possible or adequate statistical values are available:

(1) Circular orbit speed (at end of burning),

(2) Gravitational losses from the form of the ascent path and the thrust angle program,

(3) Air drag losses,

(4) Braking impulse at the initiation of the return,

(5) Braking impulse at landing.

In addition,

(6) the rotation of the earth

must be considered (for polar orbits, = ± 0 independent of the launch site; for equatorial orbits, = - 464 m/sec; for inclined orbits, corresponding intermediate values).

2.2. MISSION PROFILE AND POWER REQUIREMENT

On the one hand, the nominal orbit heights are to be as low as possible, in order to obtain a maximum payload; on the other hand, the stability or lifetime in this orbit is to be great enough so that maneuvers can be accomplished there.

In addition to the orbit height, the ballistic parameter of the rocket determines the lifetime (Fig. 2-1); for the sketch shown here, this might lie between $2.5 \cdot 10^{-3}$ and $4.0 \cdot 10^{-3}$ kg/m^2. According to Fig. 2-1, this corresponds for the CIRA atmosphere and 5° inclination, to a lifetime range of 7 to 9 days for a near-equatorial 200 km. orbit. /8

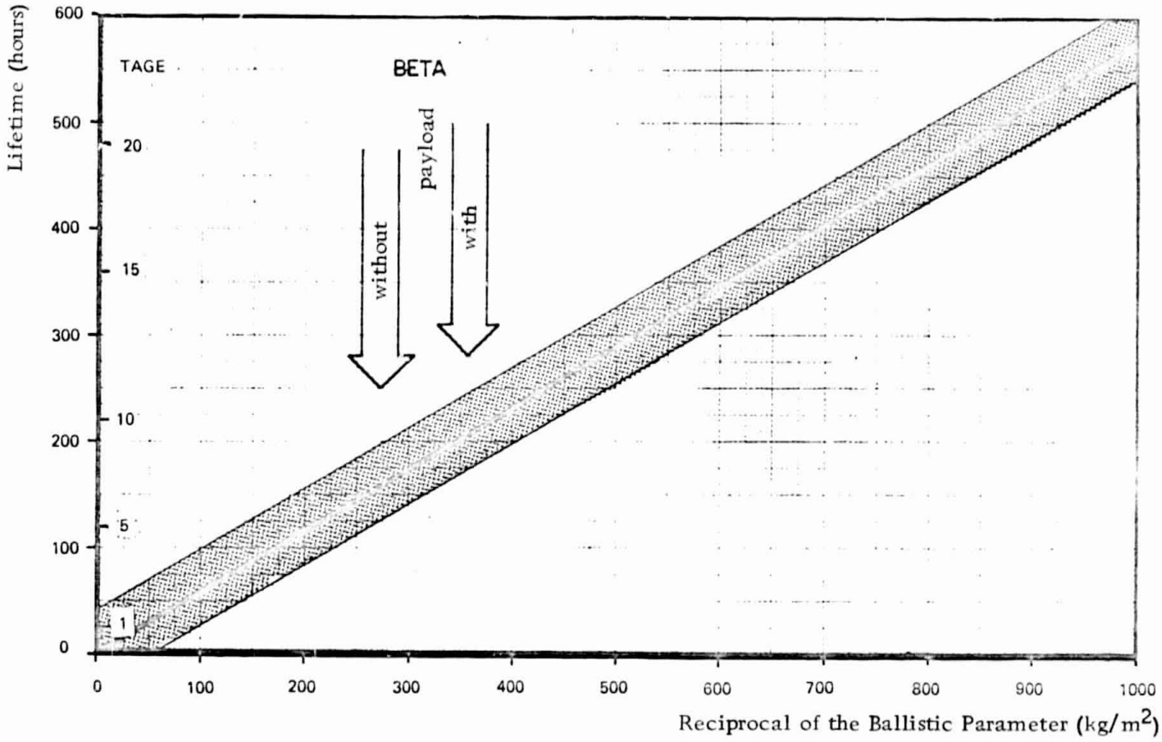

Figure 2-1. Lifetime of Space Objects in a 200 km Orbit, as a Function of the Ballistic Coefficient.

The computation is still to be done for the power requirement for the orbital braking maneuver upon starting the return, and the power requirement for reducing the residual speed to zero before landing. /9

Figure 2-2 shows the curve of speed versus height; according to this a braking impulse of $\Delta V = 50$ m/s is sufficient, without especially high speeds or accelerations entering in; this is due to the small ballistic coefficient of 2.0 to $5.0 \cdot 10^{-3}$. /10

It can also be inferred from Fig. 2-2 that the impact speed is in the neighborhood of $\Delta V_A = 70$ m/s.

Taking account of course corrections and maneuvers before landing (160 m/s), the impulse requirement for the return trip is set at a total of 280 m/s. This forms only about 3 % of the total impulse requirement. Thus launch into a polar orbit poses the following requirement: /11

(1)	Circular orbit speed at a height of 200 km	7774 m/s
(2)	Gravitational and air drag losses	2346 m/s
(3)	Braking impulse for return	50 m/s
(4)	Landing impulse	70 m/s
(5)	Reserve for maneuvers	160 m/s
		10400 m/s

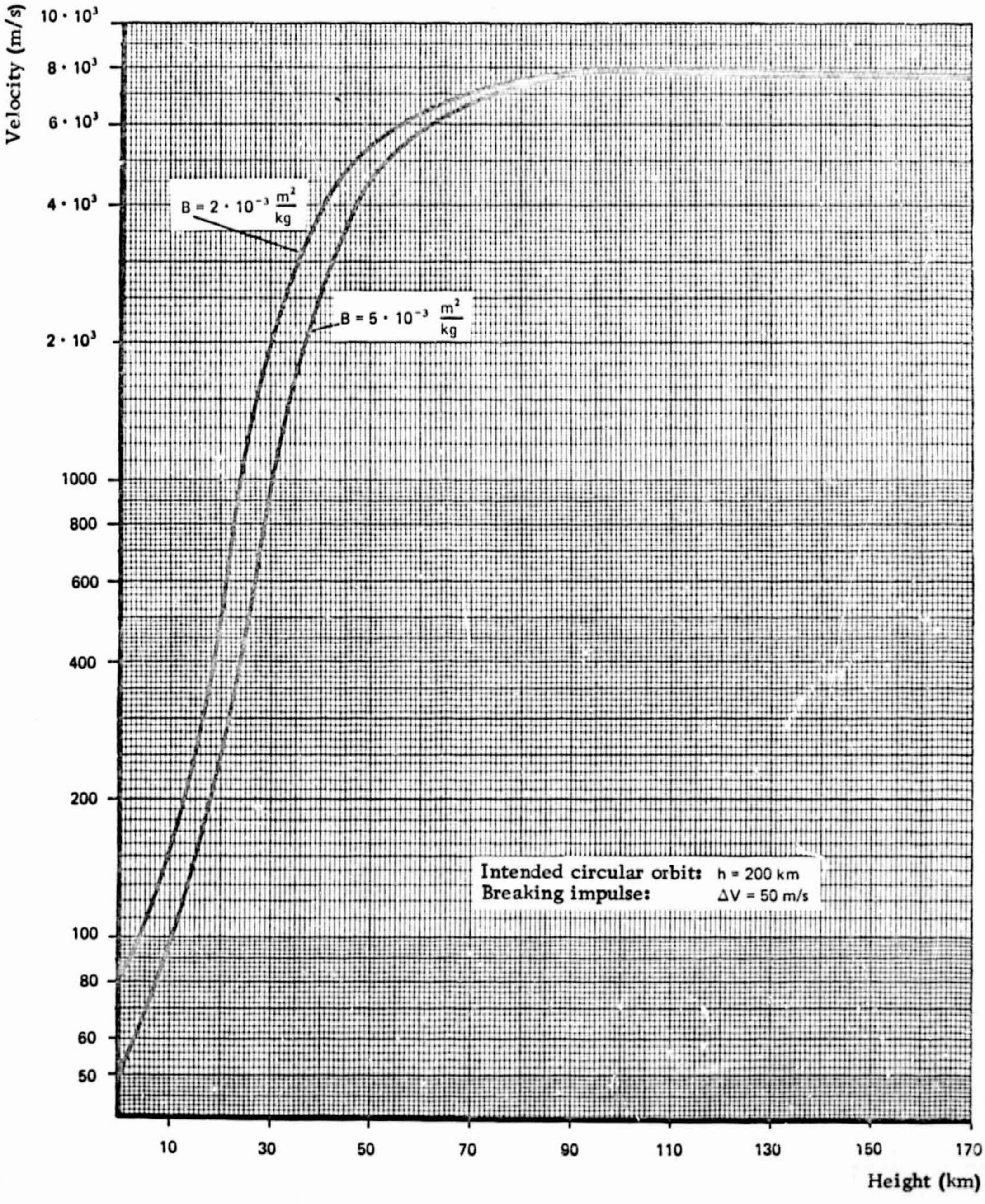

Figure 2-2. Velocity Curve for the Ballistic Return from a 200 km Orbit for BETA.

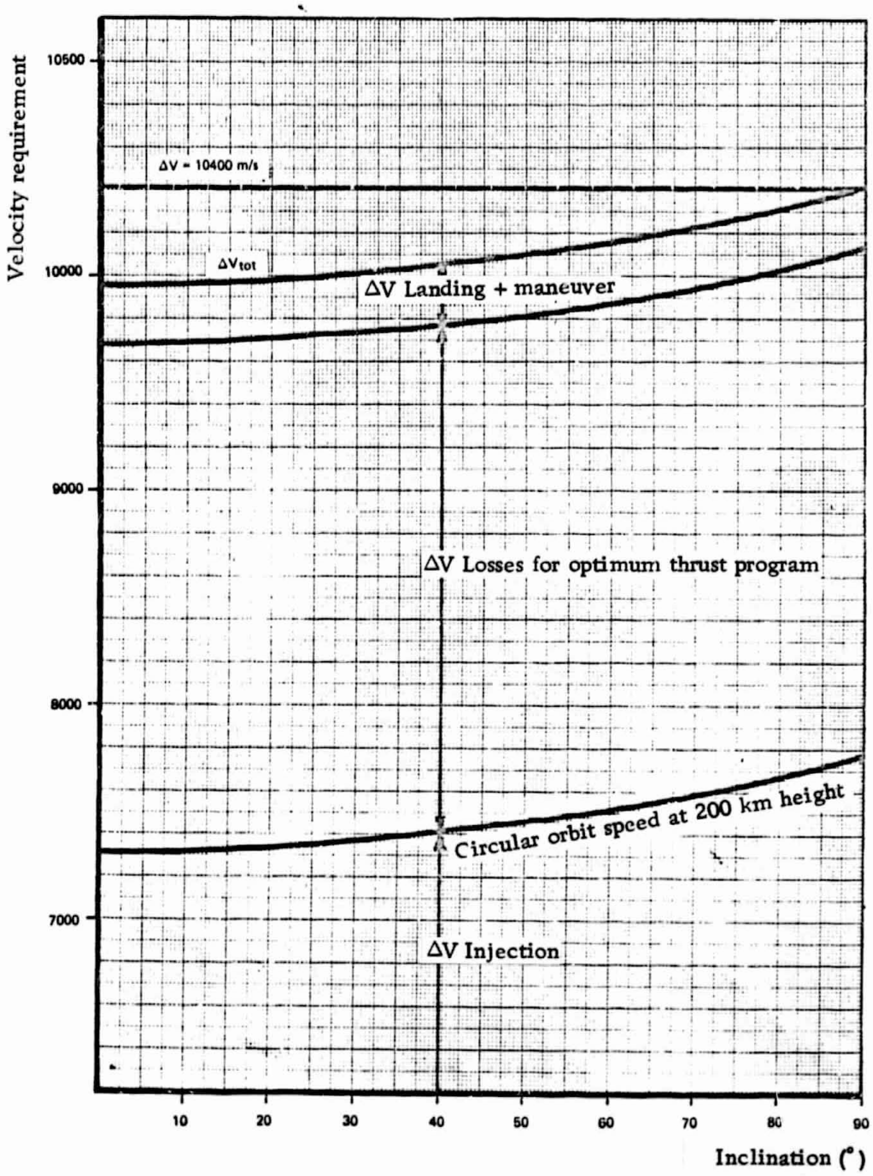

Figure 2-3. Power Requirement for a Single-stage Booster Rocket.

Conditions are more favorable for orbits at inclinations other than 90°, as shown in Fig. 2-3. An orbit at a height of some 450 km can be reached with the same impulse requirement from an equatorial launch base.

2.3. PERFORMANCE SUMMARY

2.31. Design factor K

The net weight of the booster (or the K-factor) plays an especially important part in single-stage devices. Figure 2-4 shows a summary for existing stages;

CENTAUR is shown twice, since the mass at the end of propellant burning must be computed without the insulation panels (which are blown off earlier), but in a comparison with the other stages the insulation must be included in the calculation.

An attempt was made in Fig. 2-4 to define a model region which rates the technical progress. Thus the upper limit is defined by the stage of the sixties, the lower limit by the SASSTO design point [1], whose technology can scarcely be realized before 1980.

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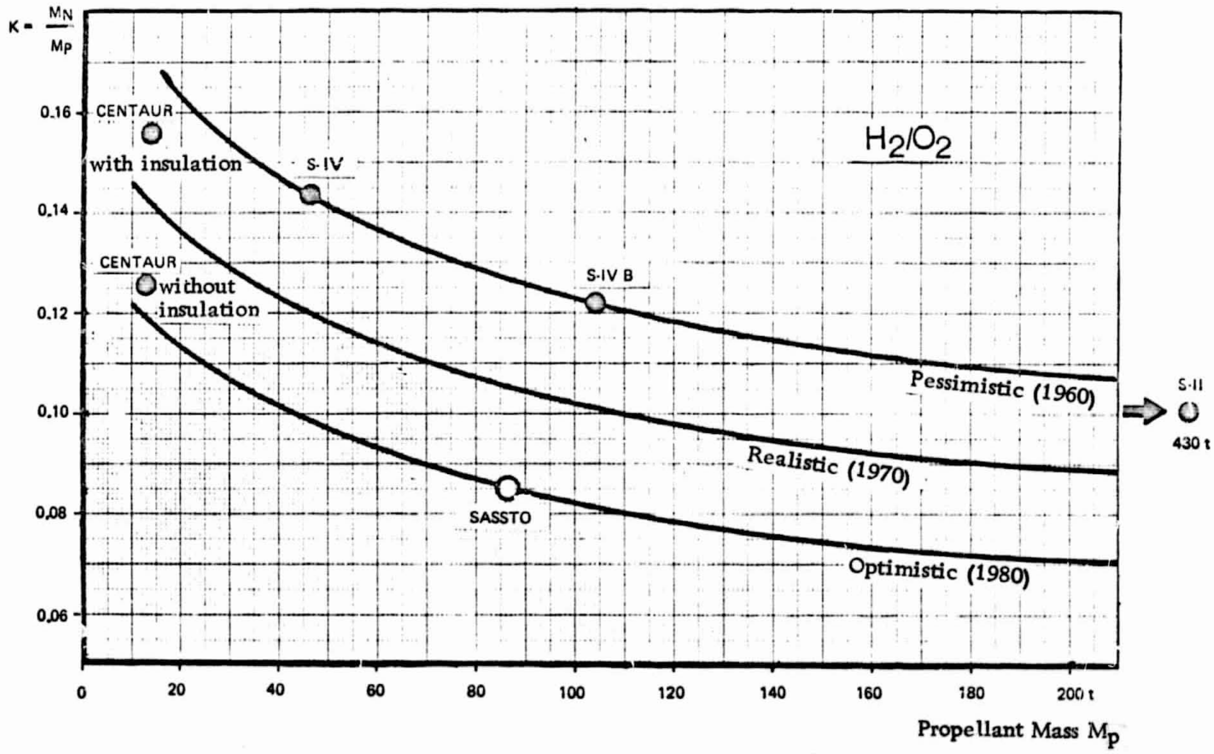

Figure 2-4. Model Development for the Design Factor K for the Technology of 1960, 1970 and 1980.

The middle curve was used as a reference line for the present study (realistic, or 1970 standard).

2.32. Thrust level

The thrust level or the thrust pattern during ascent is extraordinarily critical as regards the payload.

Different thrust profiles have been calculated for propellant weights of 100, 115 and 120 tons, at steps of:

		Quantity of propellant used, in %						
Thrust level	1.00	70	70	60	40	40	40	30
	0.66	30	20	20	60	40	20	30
	0.33		10	10		10	20	30
	0.10			10		10	20	10

The propellant mass plus the net weight for maneuver, rendezvous and return was assumed constant at 1000 kg in all calculations and was added to the net weight. /13

Because of basic considerations and the limited scope of this study, the number of path calculations was limited to a total of 32. The results are shown in Figs. 2-5 to 2-7.

Figure 2-5 shows that, independent of the total quantity of propellant (100, 115, 130 tons), for a constant launch acceleration (1.4 g) there is an optimum rise time of T ≈ 500 sec. There is, for example, an almost optimal rise time for the following program:

<u>Phase 1:</u> Thrust: $F = F_o$ (launch thrust)
Mass of propellant used: $m_{p_o} = 0.6\ m_{p_{total}}$ (60%)

<u>Phase 2</u>: Thrust: $F_1 = 0.66\ F_o$
Mass of propellant: $m_{p_1} = 0.20\ m_{p_{total}}$ (20%)

<u>Phase 3</u>: Thrust: $F_2 = 0.33\ F_o$
Mass of propellant: $m_{p_2} = 0.10\ m_{p_{total}}$ (10%)

<u>Phase 4</u>: Thrust: $F_3 = 0.10\ F_o$
Mass of propellant: $m_{p_3} = 0.10\ m_{p_{total}}$ (10%)

This thrust program gives the following payloads for the three total propellant masses covered by the preceding path computations:

Propellant Mass	Payload
100 t	1550 kg
115 t	2650 kg
130 t	3600 kg

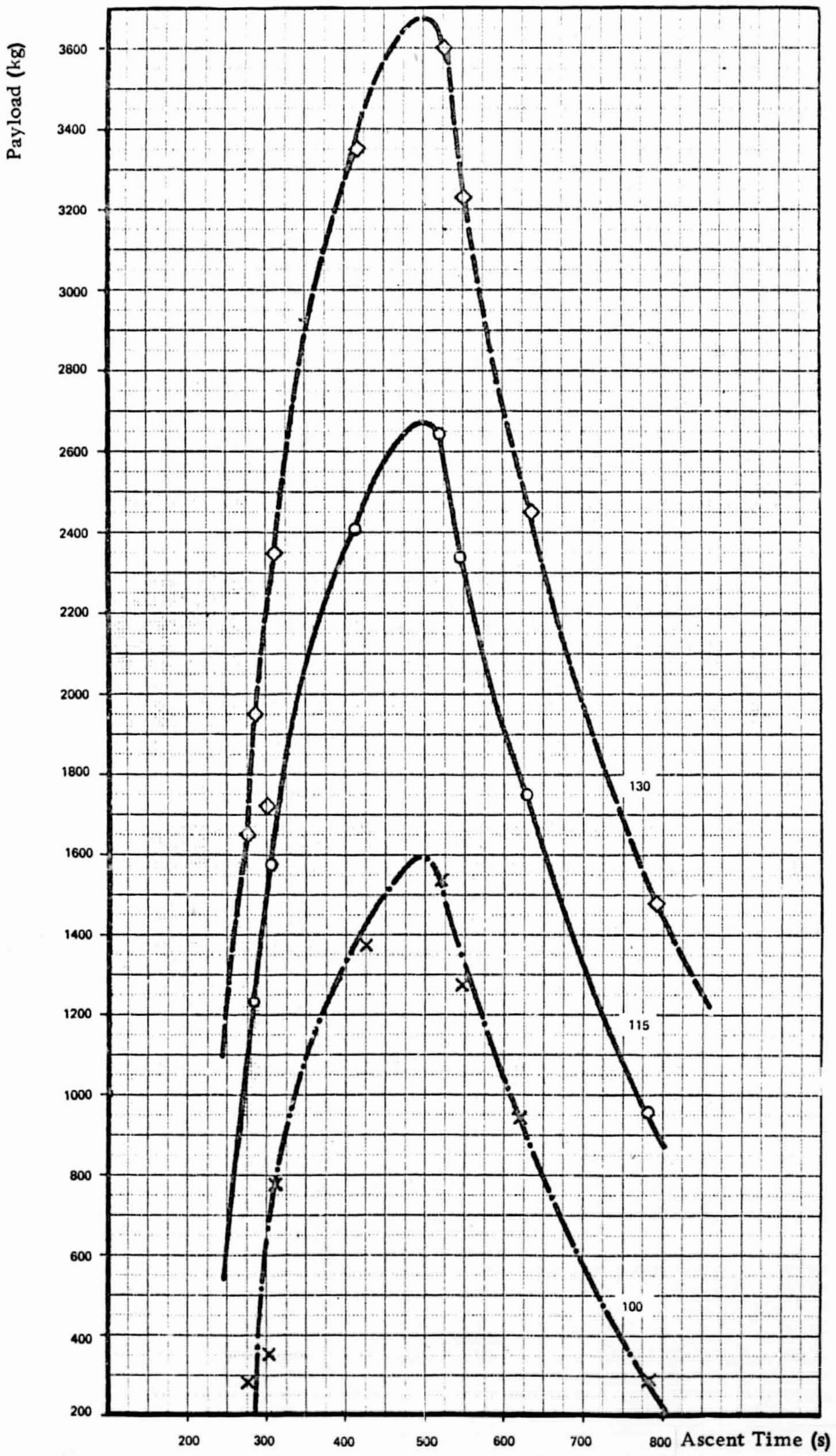

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Figure 2-5. Payload as a Function of the Ascent Time = Burning Time for 100, 115 and 130 t Propellant Mass.

Interpolating from these values, a payload of 3000 kg can be obtained for a total propellant mass of 120,000 kg. This would correspond to a total mass of 136,000 kg for the booster aggregate.

Figure 2-6 shows that payload advantages are to be obtained by increasing the launch acceleration up to 1.6 g. The increasing engine weight was taken into consideration, but not the likewise increasing structural weight occasioned by the greater dynamic loading. For this reason the launching acceleration was limited to 1.4 g for the time being. /15

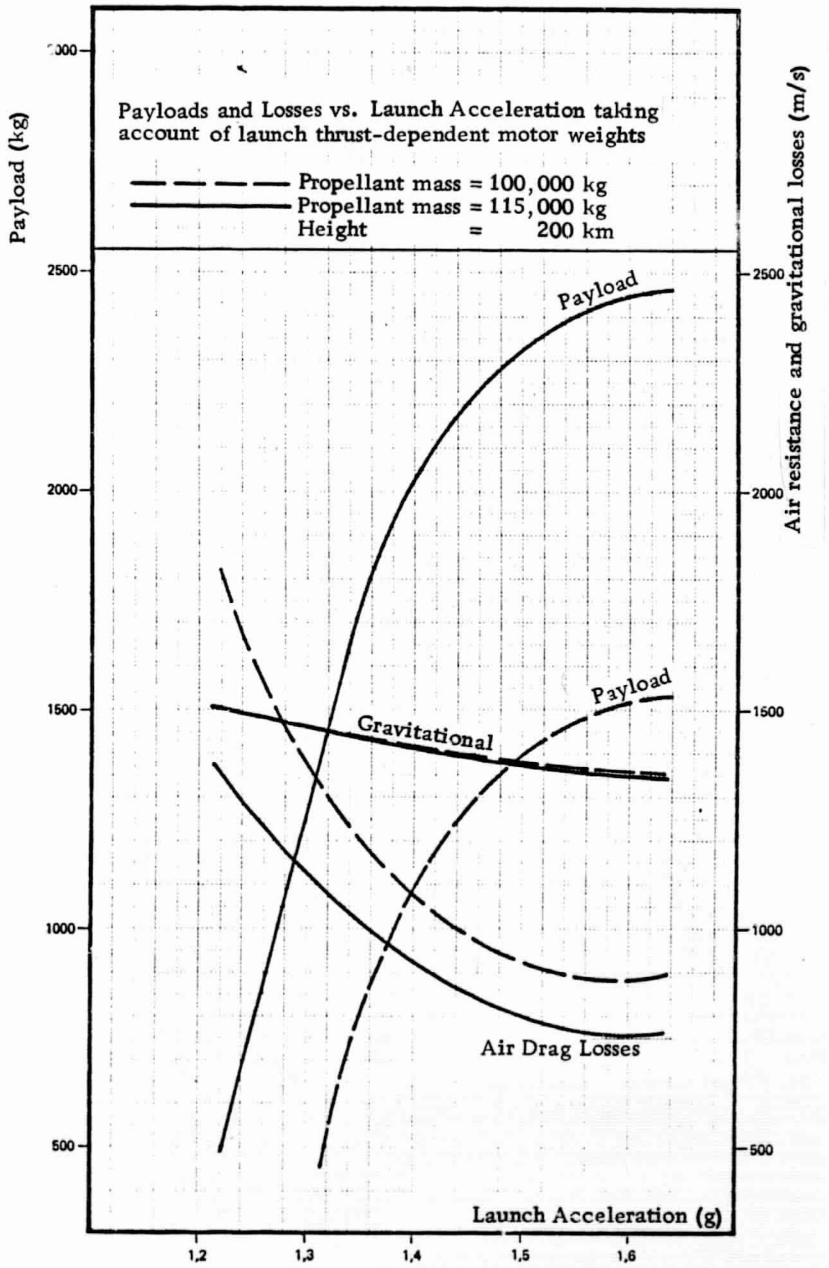

Figure 2-6. Payload, Gravitational and Air Drag Losses versus Launch Acceleration.

Figure 2-7 shows the dynamic pressure in N/m² for different launch accelerations, versus the height as a measure of the aerodynamic load. The dynamic pressure Q is, however, a controlling factor not only for the high air drag losses in m/sec, which are expected to be about 900 - 1050 m/sec for an acceleration of, for example, 1.4 g. Air drag losses increase surprisingly at lower initial accelerations: the analysis of this effect shows a second maximum of the dynamic pressure (Fig. 2-7), which can be avoided with more rapid penetration of the denser air layers. /16 /17

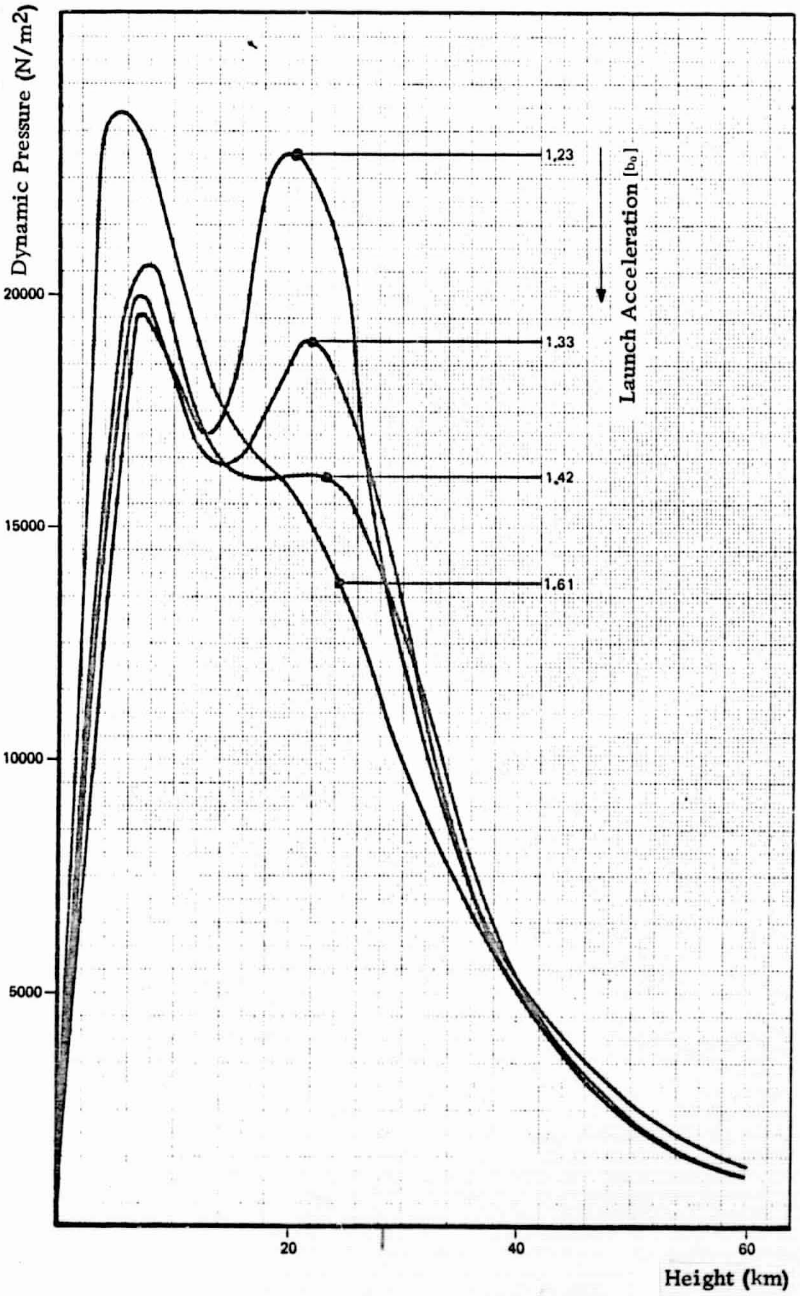

Figure 2-7. Variation of Dynamic Pressure with Height for Launch Accelerations of 1.23 - 1.33 - 1.42 and 1.61 g.

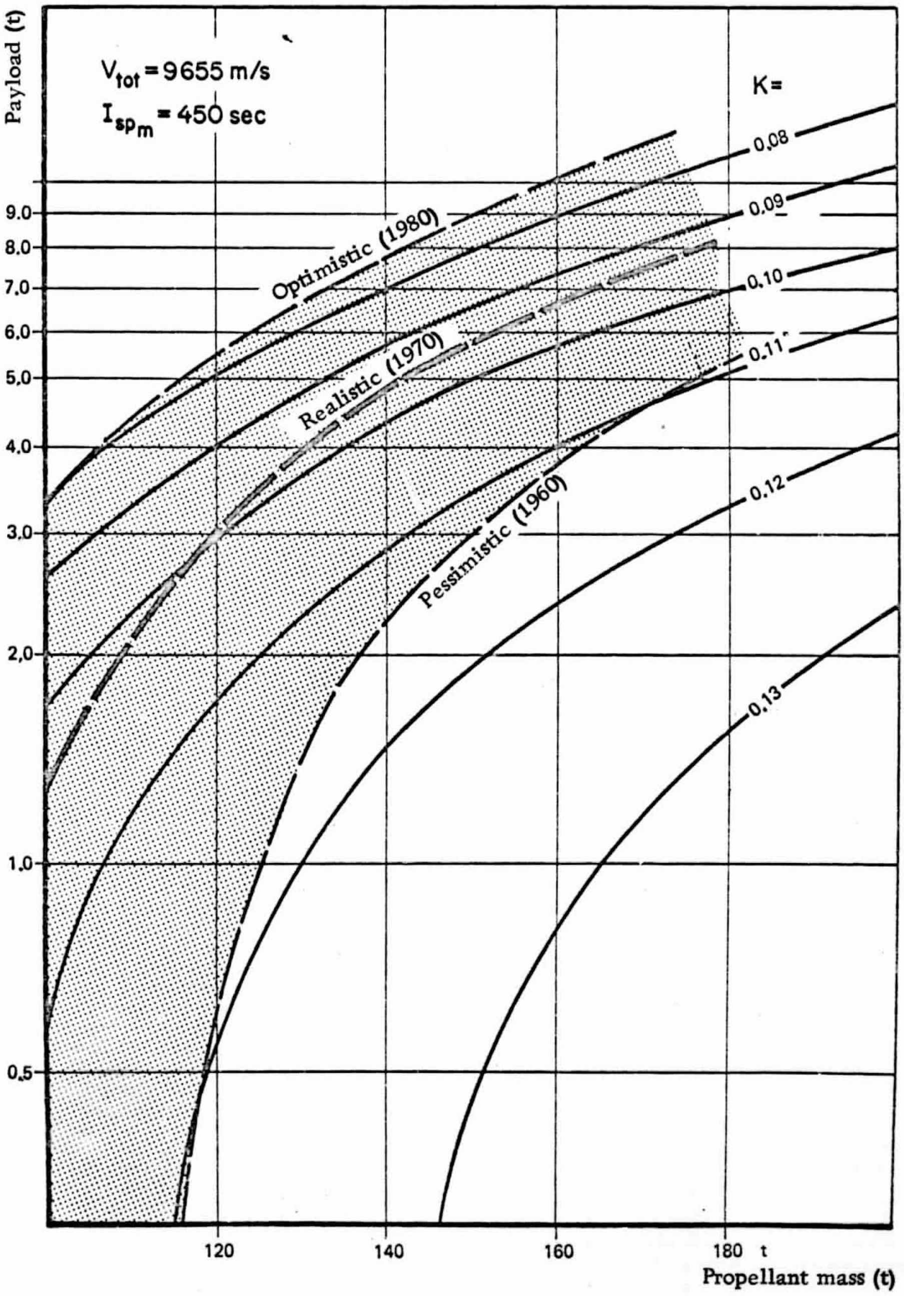

Figure 2-8. Payload for a Single-stage Booster as a Function of the K-factor (Net Weight) for a Mean Specific Impulse of 450 sec. (Equatorial 200 km Orbit).

The decisive effect on the high air drag losses results from the fact that the cross-sectional area F enters the calculation as a linear factor, and at 64 m^2 is relatively large in the BETA design. Figure 2-6 shows further that gravitational losses depend only slightly on the launch acceleration.

2.33. Performance diagram

When the effective impulse requirement has been established to a first approximation and estimates or assumptions are on hand for engine performance (see chap. 3.1), it is possible to survey the range of booster payloads in relation to the size. The payload is here understood as including the payload enclosure; for the time being, the net weight was set at 0.1 M_P.

The results are shown in Fig. 2-8: they show the strong effect of the net weight on the payload. These payloads hold for devices including return; if recovery equipment, braking propellant, etc. are deleted, the payload is increased by more than 1600 kg.

Further technical investigations, which among other things are to provide a more exact determination of the net weight, were carried out on the basis of the performance diagram with 115 t of propellant mass. This allows a payload range of 2.5 to 4.5 t to be expected.

Performance improvements over the theoretical results of this chapter for the mass at the end of burning can still be attained by introducing a variation of the propellant mixing ratio from 1 : 5.5 at launch to 1 : 8 before the end of burning, as well as by further optimizing the ascent program and the thrust program.